THE AEROSPACE PLANE: TECHNOLOGICAL FEASIBILITY AND POLICY IMPLICATIONS

by

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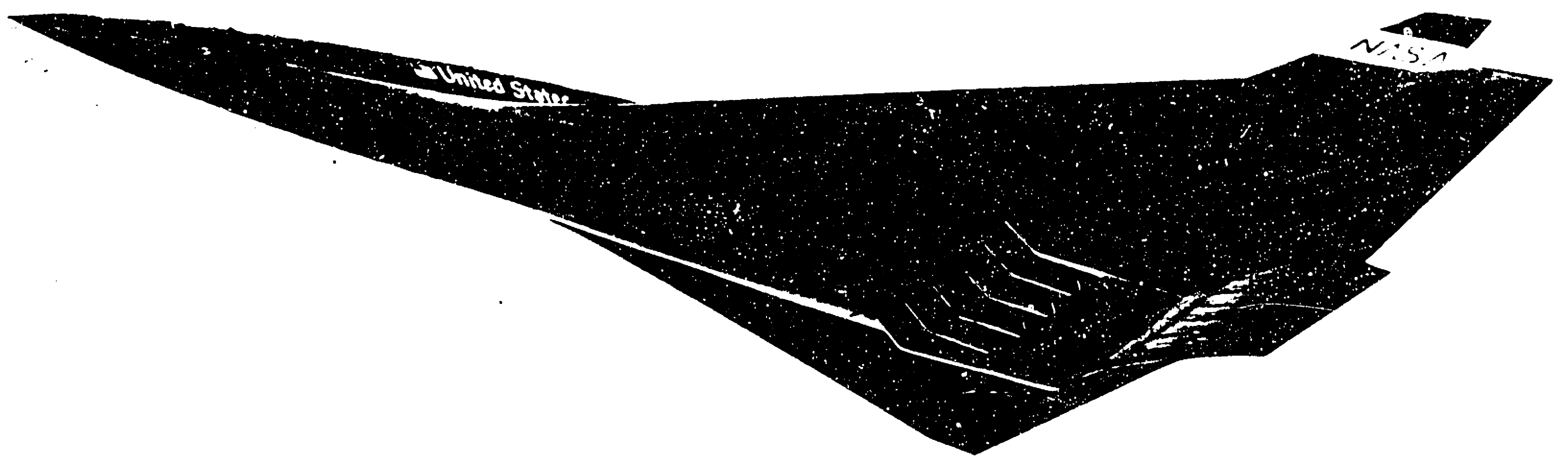

Artist's Conception of the Aerospace Plane (ASP)

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ABSTRACT

Recently, national attention has focused on the aerospace plane (ASP) concept to fulfill a variety of civilian and military goals. The envisioned vehicle would be a fully reusable, single-stage-to-orbit, horizontal take-off and landing, scramjet-powered, manned vehicle.

In this thesis, the technical feasibility of propulsion and thermal protection at high Mach number flight was explored. A mass-estimating program was also written and run for different values of the currently unknown maximum scramjet operating velocity (from Mach 12 to Mach 23). The results showed that ASP gross weight, and hence costs, are highly dependent on the operating range of scramjet engines. Dramatic improvements in vehicle performance occur if the operating regime of scramjets extends to about Mach 17. Beyond Mach 17 the gains are marginal.

ASP costs were estimated using Transcost, an empirical space transportation system cost model, modified to take into account the advanced air-breathing engine system an ASP would require. Development, fabrication, and operations costs were calculated for the range of scramjet performance limitations. Operating costs were estimated to be approximately \$5 million, while development and fabrication costs were estimated to be approximately \$17 billion and \$1 billion, respectively.

Since the ASP concept is being considered for several uses, a variety of potential ASP applications were surveyed. The examination showed that for civilian purposes, the ASP is probably unsuitable as a hypersonic transport, but has promise as a low-cost space launch vehicle. The only cost-effective, non-destabilizing military application seemed to be strategic reconnaissance. For each major application, the institutional context of ASP development was examined, focusing on the interests of the various governmental groups involved in the effort. Finally, several policy recommendations were made.

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Finally, for her cheerful support and encouragement, I thank my wife, Barbara, to whom this thesis is dedicated.

GLOSSARY OF ACRONYMS AND ABBREVIATIONS

ABES	air breathing engine systems
ACC	advanced carbon-carbon
AIAA	American Institute of Aeronautics and Astronautics
APRC	Aeronautical Policy Review Committee (White House)
ASAT	anti-satellite weapon
ASP	aerospace plane
ATR	air turbo-ramjet
CBO	Congressional Budget Office
CER	cost-estimating relationship
CFD	computational fluid dynamics
DARPA	Defense Advanced Research Projects Agency
DOD	Department of Defense
ELV	expendable launch vehicle
ESA	European Space Agency
ET	external tank
FY	fiscal year
g	gravity
GASL	General Applied Sciences Laboratory
GEO	geostationary orbit
GLOW	gross lift-off weight
GTAT	ground turn-around time
HOTOL	Horizontal Take-off and Landing
HRE	hypersonic research engine
HST	Hypersonic Transport
IOC	initial operational capability
kg	kilogram
L/D	lift over drag ratio
LEO	low earth orbit (200 to 600 km)
LH2	liquid hydrogen
LOX	liquid oxygen
MMC	metal matrix composite
MY	man-year
NASA	National Aeronautics and Space Administration
NCOS	National Commission on Space
OSTP	Office of Science and Technology Policy
OTA	Office of Technology Assessment
psf	pounds per square foot
R&D	research and development
RCC	reinforced carbon-carbon
RFP	request for proposals
RSR	rapid solidification rate
scramjet	supersonic combustion ramjet
SDIO	Strategic Defense Initiative Organization
SRB	solid rocket boosters
SSME	Space Shuttle Main Engines
SST	Supersonic Transport
STS	Space Transportation System (The Space Shuttle)
TPS	thermal protection system
USAF	United States Air Force

LIST OF SYMBOLS

Chapter 2

Ae	Engine area at exit
Cf	Skin friction coefficient
Cp	Specific heat
D	Drag
E	Energy
f	Fuel to air ratio
g_{leo}	Gravitational acceleration at low-earth orbit
g_o	Gravitational acceleration at earth's surface
h	Height of orbit
hf	Heat transfer coefficient
Isp	Specific impulse
$\overline{Isp}$	Average specific impulse
L	Lift
$\dot{m}_a$	Mass flow rate of incoming air
$\dot{m}_e$	Mass flow rate of combustion products at engine exit
$\dot{m}_f$	Mass flow rate of injected fuel
M	Mach number
MO	Free-stream Mach number
M2	Engine Mach number
Matr	ATR Engine mass
Mf	Fuselage mass
Mfe	Fixed Equipment mass
Mo	Initial take-off mass (sum of all component masses)
Mpl	Payload mass
Mpr 1	ATR Propellant mass
Mpr 2	Scramjet Propellant mass
Mpr 3	Rocket Engine Propellant mass
Mroc	Rocket Engine mass
Mscram	Scramjet Engine mass
Mt	Fuel Tank mass
Mtps	Thermal Protection Sytem mass
Mw	Wings, empennage (tail assembly), and landing gear mass
N	Total number of stages and engines
p_a	Ambient pressure
p_e	Exit pressure
p_o	Free-stream pressure
p_{to}	Free-stream stagnation pressure
p_{ta}	Stagnation pressure at engine inlet
$\dot{q}$	Heat flux/unit area
$\dot{Q}$	Total heat input to ASP surface due to aerodynamic heating
$\dot{Q}e$	Heat input to the scramjet engines
$\dot{Q}f$	Heat absorbed by the fuel
$\dot{Q}r$	Radiated heat from the ASP
S	Unit area
St	Stanton number
r1	Mass at ATR shut-off/take-off mass
r2	Mass at scramjet shut-off/mass at ATR shut-off
r3	Final (rocket burn-out) mass/mass at ATR shut-off
rnet	Net mass ratio: on-orbit mass/take-off mass
R1	(1-r1)/r1 = ATR propellant mass/mass at ATR shut-off

R2	(1-r2)/r2 = scramjet propellant mass/mass at scramjet shut-off
R3	(1-r3)/r3 = rocket engine propellant mass/final mass
R(eff)	Effective mass ratio, taking into account a 2 percent fuel reserve
R_{leo}	Distance from earth's center to low-earth orbit
R_o	Earth's radius
t	Time
T	Thrust
T_o	Free-stream temperature
Ts	ASP surface temperature
Tt	Total temperature
Tt_o	Free-stream total temperature
Tw	Skin temperature
u_e	Flow velocity at exit
u_o	Flow velocity at inlet
V_{leo}	Orbital velocity at low-earth orbit
ΔV	Delta-v
$\dot{V}$	Net acceleration
W	Vehicle weight

Greek Symbols

α	Angle of Attack
θ	Angle of Climb
λ	Payload mass/fueled vehicle mass minus payload mass
δ_{atr}	Mass of ATR engines/mass carried
δ_f	Mass of fuselage/total mass contained
δ_{fe}	Variable portion of fixed equipment mass/initial mass
δ_{roc}	Mass of rocket engines/mass propelled
δ_{scram}	Mass of scramjet engines/mass carried
δ_{tps}	Mass of thermal protection system/protected mass
δ_w	Mass of wings, empennage, landing gear/supported mass
ϵ	Mass of fuel tanks/mass of propellant contained
ϵ	Emissivity
γ	Specific heat ratio
μ	Mach angle
π_d	Diffuser pressure recovery ratio
ρ	Density
σ	Stefan-Boltzmann constant

Chapter 3

ALT	Absolute engine operational altiutude (km)
Cc	Pre-launch operations, assembly, and check-out costs
Cdev	ASP development cost
Ce	Launch and mission control costs
Cm	Technical systems management costs
Cops	Total operations costs
Cprod	Total ASP fabrication cost
Cprop	Propellant costs
Crf	Refurbishment costs
f1	Cost correction factor for technical development standard
f2	Cost correction factor for technical quality

f3	Cost correction factor for team experience
f4	Cost reduction factor for series production
Fab	Fabrication cost of an air-breathing engine
Faf	ASP airframe fabrication cost
Faf,n	Fabrication costs for n ASP airframes
Fatr	ATR fabrication cost
Fsj	Scramjet fabrication cost
Fr	Rocket engine fabrication cost
H	Development effort in man-years
Haf	ASP airframe development cost
Hatr	ATR development cost
Hsj	Scramjet development cost
Hr	Rocket engine development cost
Laf	Expected lifetime of ASP airframe
Le	Expected lifetime of ASP engines
TMAX	Maximum rated thrust (kg)
TN/W	Normal rated thrust divided by engine dry weight

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PREFACE

Many different configurations for a space transportation system can be imagined. Various permutations of factors such as the degree of expendability or reusability, the number of stages, horizontal versus vertical take-off, ballistic (water) versus winged (land) landing, payload size, propulsion system, and manned versus unmanned systems lead to strikingly different designs, capabilities, and economies. This thesis presents one such configuration, the aerospace plane (ASP), for consideration as a possible successor to the space shuttle. The ASP would be a fully reusable, single-stage-to-orbit, horizontal take-off and landing, scramjet-powered, manned vehicle.

The aerospace literature discusses other potential follow-ons to the shuttle including the Shuttle Derived Vehicle (SDV), the Heavy Lift Launch Vehicle (HLLV), and the SRB-X, but the ASP warrents special consideration for several reasons. First, recent statements from members of the technological and political communities indicate that the ASP is a front-runner in the advanced launch vehicle competition. Aerospace experts are saying that what was considered fantasy a mere two or three years ago, is appearing to be feasible in light of recent advances in technology. Politicians at the highest levels have been actively discussing the ASP, both for its civilian and military applications. Most notable of all was President Reagan's highlighting the ASP in his 1986 State of the Union address.

Second, while the other launch vehicle design concepts represent the traditional conservative approach of making incremental improvements on already existing systems, the ASP, if carried to fruition, would be a radical departure from existing systems. Outgoing Presidential Science Advisor Dr. George A. Keyworth used the following words to describe the ASP: "...this [the ASP] is not the ramjet technology of the 1960s extrapolated forward. This is not the 30 years of experience we have had in supersonic transport. We are talking about new materials, new design, new propulsion, new avionics that simply let us, if you wish, invent the next generation aircraft (United States Congress, Hearing on High Speed Aeronautics, 1985, p. 23)."

Finally, as its name implies, the proposed "aerospace plane" would require an unprecedented amount of collaboration between the worlds of aeronautics and astronautics. Despite its airplane-like horizontal landing, the space shuttle is still a rocket-propelled vehicle. In contrast, the ASP would require the development of hybrid, dual-mode air-breathing/rocket engines. Only by synergistically drawing on the institutionally and conceptually divided worlds of aeronautics and astronautics could such an aerospace plane be produced.

The central questions of my thesis are: 1) is the proposed aerospace plane technologically feasible, and 2) what are its policy implications? By "technologically feasible," I mean theoretically capable of launching a payload of approximately 10,000 kg into low earth orbit. While a definitive answer to the technical feasibility question will require decades of research, even a tentative assessment such as this has value because it can focus attention on key, leveraged technology areas for intensive research. The technical details of the government's ASP research is strictly classified, but it is no secret that the ASP is considered to be a high-risk, potentially high-payoff project. The primary purpose of this thesis is to show what the technological risks are and what the rewards might be. I have tried to do this through: (1) a performance sensitivity evaluation of the propulsion and thermal protection systems, and (2) an estimation of ASP component masses given certain technology levels.

In exploring the ASP's policy implications, I shall examine how much this system would cost to develop, build, and operate; what missions might the vehicle be used for and are they necessary; what does the vehicle mean to the various government agencies involved in the project; and, in general, does the ASP merit the aggressive funding it is now receiving.

THESIS ORGANIZATION

Before discussing a follow-on to the shuttle, it is necessary to look closely at the problems it faces so that potential pitfalls can be identified and evaluated. In Chapter 1, the current US space transportation system (STS) is discussed with attention to its technical short-comings and resulting diseconomies. The proposed vehicle concept, the ASP, is then presented along with a brief explanation of its technological underpinnings, its history, and its current status.

Chapter 2 contains technical feasibility calculations. The intent here is to address systems-level concerns: interactions and interdependencies, trade-offs, and interfaces between the vehicle and its environment. This is not an attempt to design an ASP. With ASP initial operational capability (IOC) planned for the turn of the century, the minds that will eventually design the vehicle are still in high school.

Chapter 3 presents some ball-park figures for the costs of the launch vehicle recognizing the high degree of uncertainty inherent in such calculations at this early stage. The cost analysis uses a modified version of the European Space Agency's statistically-derived "Transcost" model for the estimation of launch vehicle development, fabrication, and operations costs.

Chapter 4 assesses the utility of an ASP for both civilian and military purposes. Also examined are the interests of the different organizations which may have a stake in the current research effort.

Chapter 5 summarizes the results of the preceeding chapters, presents an overall conclusion, and contains several policy recommendations.

CHAPTER 1: AEROSPACE PLANE CONCEPT AND MOTIVATION

"Space transportation is now at a very early stage in its development, probably corresponding to the period in the evolution of the aircraft when the flying boat appeared as the ultimate answer for long distance travel."

- Prof. Rene Miller, Massachusetts Institute of Technology

"The Columbia is the 'Wright Flyer' of Space Shuttles."

- Prof. John Anderson, University of Maryland

1.1 Space Commercialization

A broad range of businesses are being drawn toward commercial space ventures by the potential for an entirely new class of products and services, based on the space vantage point and the weightless environment. Specific endeavors include space-based production of pharmaceuticals, gallium-arsenide semiconductors, and advanced ceramics, as well as services such as advanced telecommunications, remote sensing, and in-orbit satellite servicing. Estimates of the future profitability of such space-based commercial activity vary greatly. For example, the Cambridge-based Center for Space Policy predicts that revenues from commercial space operations will reach $55 billion by the year 2000 (Covault, 1984, p. 40). Rockwell International has predicted revenues will exceed $30 billion by the mid 1990s (Covault, 1984, p. 40). Less optimistic analysts project much lower figures, citing impediments such as the suspension of space shuttle flights; the tightening space launch insurance market, recoiling after $600 million in losses from 1982-1985; the hazards posed by man-made space debris in low earth orbit (LEO); the environmental pollution that may result from a vastly increased launch rate using present vehicles and propellants; and the daunting complexity and red-tape that companies face when attempting to engage in commercial space research and development.

All aerospace analysts manage to agree on one point however: the cost of transporting mass to orbit is the most critical factor in determining the rate at which space resources are exploited. History offers several analogies. Just as the develoment of sea-ports, inter-continental railroads, interstate highways, and civil aviation routes was crucial to the economic growth of the regions they served, so too is the development of a low-cost, reliable space transportation system a prerequisite for extensive commercial operations in space.

1.2 Launch Costs

Space transportation costs are high, especially when compared to transportation costs within the atmosphere. Today's aircraft can transport a given mass ten times as far at one-thousandth of the cost of space transport (United States Congress, OTA Study Civilian Space Stations and the U.S. Future in Space, 1984, p. 120). Admittedly, air is a more hospitable operating environment than the vacuum of space, but the energy costs of going from earth to LEO (approximately 250 kilometers high) are only four times greater than the energy costs of going round-trip from Los Angeles to London (Miller and Akin, 1980, p. 195). This latter transportation is available for about $8/kg. One reason for this disparity between air and space transportation costs is that aircraft are used much more regularly than spacecraft. Jet aircraft make many flights a day and carry a passenger and cargo complement as close to full as is practical. Worldwide about 20 million airline flights are made per year (Miller and Akin, 1980, p. 195). Only a few hundred space launches are made yearly, meaning the capital expenditures supporting space activities must be amortized over a much longer time. A second reason for the high cost of space transportation is that all current space transportation systems have some degree of expendability, whereas airplanes can be reused for 40 years or more.

Figure 1-1 on the following page shows the historical trend in launch costs (adapted from Koelle, 1985). The specific launch cost is given in units of 1985 $ per kilogram (kg) of payload placed in LEO. Remarkable progress has been made in two decades. Costs have been reduced from over $70,000/kg to around $5,000/kg using expendable launch vehicles (ELVs). The levelling off shown by the chart indicates that ELV technology has matured. Further cost reduction using ELVs does not appear to be feasible. The space shuttle, however, is a different breed of launch vehicle, the first step towards total reusablity.

1.3 NASA's Space Transportation System (STS)

The explosion of the space shuttle Challenger in January has put the shuttle program on hold pending a thorough investigation of the causes of the mishap. While NASA and its Congressional overseers probe the shuttle's reliability, an examination of the shuttle's operational performance is also warranted. Over the past five years the space

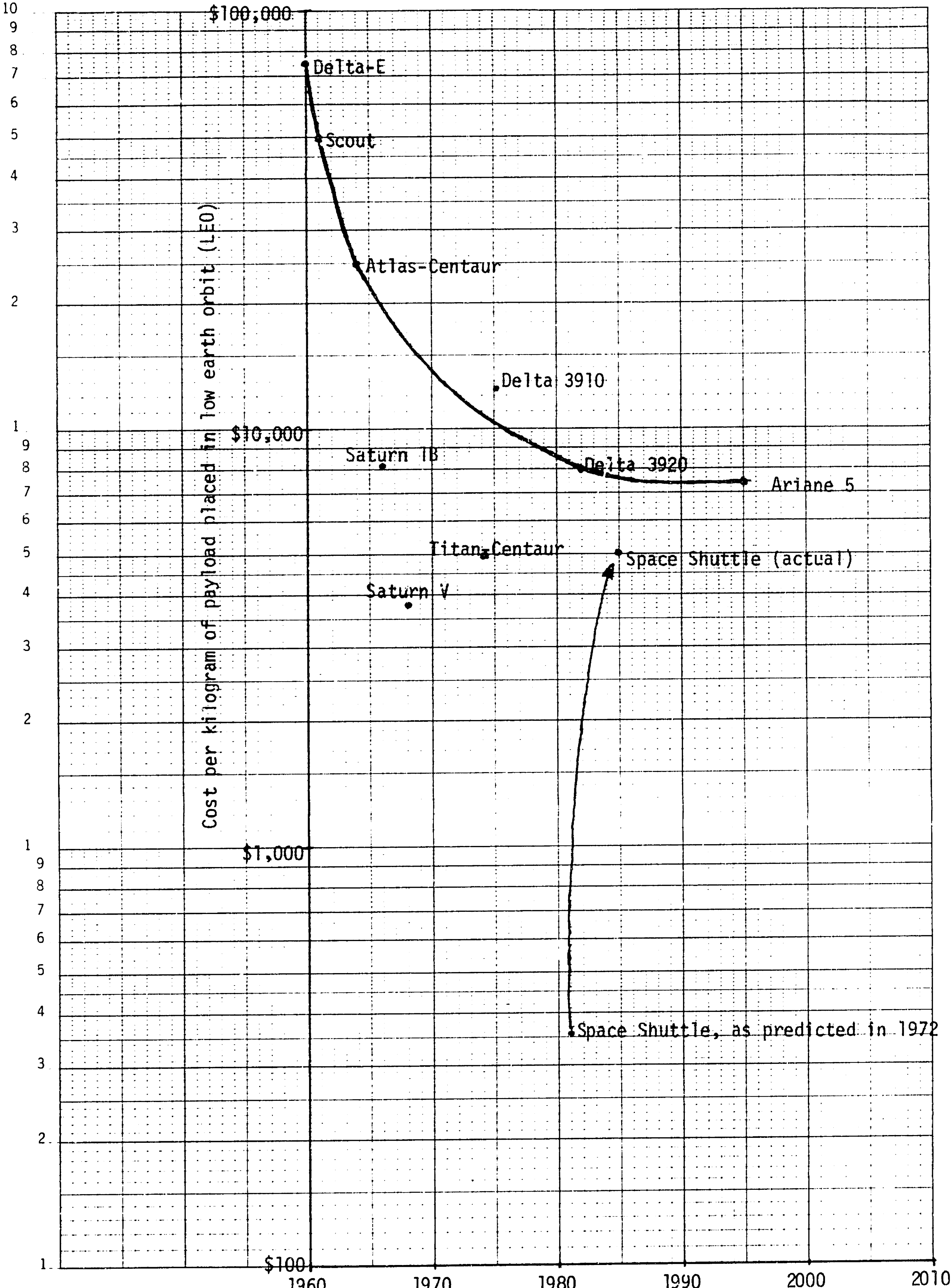


Figure 1-1: Historical Trend of Launch Costs to LEO

shuttle fleet has flown 24 missions, sent over 100 people aloft, conducted over 320 scientific experiments, and logged over 50 million miles in space. There are two sides to a review of this performance. "As it was originally conceived, the shuttle was supposed to do two things for us," says Dr. Jerry Grey, publisher of Aerospace America, "It was supposed to provide us with a flexible way of operating in space. In that respect it has been an overwhelming success. It was also supposed to give us an economical space transportation system. In that respect it has been a major disappointment." (Toner, 1985, p. 88).

The Cost of a Shuttle Flight. Putting aside any non-economic justifications of the shuttle, from the standpoint of reducing launch costs to LEO the shuttle is a "giant leap" backward. In 1985 the Congressional Budget Office (CBO) calculated the cost of a shuttle flight using a variety of accounting methods (Space Business News, 1985, p. 5):

Accounting Method	Cost per Launch	Cost per kg*
Short-run marginal cost	$42 million	$1,424 / $1,969
Long-run marginal cost	$76 million	$2,577 / $3,565
Average full operational cost	$84 million	$2,848 / $3,940
Average full cost less development	$108 million	$3,664 / $5,066
Average full cost	$150 million	$5,088 / $7,035

* for ideal 65,000 lbs payload/ for actual maximum 47,000 lbs payload

Figure 1-2: The Cost of a Shuttle Flight

In 1972, NASA Administrator James Fletcher estimated that the cost of launching payloads to LEO using the Saturn V was $3,689 per kilogram (in 1985 dollars). Therefore, when the shuttle finally achieves its paylaod design goal of 65,000 lbs, using the most reasonable accounting method--average full cost less development--it will have equalled the performance of the Apollo launch vehicle of fourteen years ago! Nor is this situation apt to improve in the near future. In 1973, Mathematica, Inc. determined that in order for the shuttle to break-even, it would have to fly 30 flights per year (Pace, 1982, p. 47). Before the Challenger disaster, NASA did not expect to reach that goal until 1990 at least. This goal is receding even farther into the future now, with the likelihood that additional time-consuming pre-flight check-out procedures will be required for future shuttle launches.

Shuttle Development Costs. In 1972, NASA estimated STS research, development, test, and engineering (RDT&E) costs to be about $11 billion (1985 $) (Toner, 1985, p. 47). In the end, the shuttle's RDT&E amounted to $14 billion. It is not at all surprising that a major high technology space system would exhibit development cost growth. Compared to cost overruns in the Defense Department, which average about 50 percent, NASA did quite well (United States Congress, Hearings on Inaccuracy of Department of Defense Weapons Acquisition Cost Estimates, 1979, p. 4). Unfortunately, this error was compounded by NASA's gross overestimation of the to-orbit traffic that the shuttle would service. Fewer flights means that the shuttle investment must be amortized over a longer period of time at a higher total cost. The NASA baseline model of 30 flights per year was crucial to the shuttle's paying for itself. Before the Challenger explosion, the shuttle had made 24 successful flights (2 in 1981, 3 in 1982, 4 in 1983, 5 in 1984, 9 in 1985, and 1 in 1986), only a fraction of the projected launch rate. The problem is self-perpetuating as long as the government seeks to recoup its R&D money: raising user charges would cause customers to look elsewhere for launch services, reduce the shuttle's traffic, and further damage the shuttle's economics.

Shuttle Operations Costs. As of January 1986, the shuttle operational cost (orbiter operational and fabrication costs, minus development costs) was $108 million per launch, yet the rate charged for a dedicated shuttle payload bay was $74 million. Thus, every shuttle launch was being subsidized at least $34 million in order to compete with alternative launch vehicles such as the French Ariane. Beginning in 1989, NASA had planned to charge $87 million per launch in accordance with its "full cost recovery" policy. NASA officials said that this rate of $87 million would have enabled the shuttle to pay its operating costs--but only at a rate of 24 launches a year.

Before considering the next generation of launch vehicles, one must understand where the current generation's high operations costs come from. Figure 1-3 presents a NASA break-down of the predicted direct operating costs (that is, not including shuttle R&D or amortization of the orbiter production) for the space shuttle program, if and when it reaches its mature operating level of 30 flights per year with the full 30 ton payload. (Eldred, 1984, p. 5).

Expendables		
External Tank (ET)	19%	$15 million
Liquid Propellants	2%	$ 2 million
Reusables		
Solid Rocket Boosters (SRB)	23%	$19 million
Refurbishment	10%	$ 8 million
Manpower		
Launch Manpower	35%	$28 million
Flight Manpower	11%	$ 9 million
Total	100%	$81 million

Figure 1-3: Breakdown of Space Shuttle Operations Costs

This chart illustrates a number of significant points which will bear on the later analysis of the ASP. First, a space shuttle launch is very labor-intensive, with launch manpower, launch facilities maintenance, and crew training accounting for 46 percent of the direct operating costs. These are fixed costs. In other words, if NASA flies five flights when it had planned to fly ten, its fixed costs are doubled for each launch. Some strategies being considered for simplifying ground operations and reducing required launch manpower include: introducing highly automated systems for checkout, diagnostics, and flight planning; incorporating more long-life or maintenance-free subsystems; and using horizontal processing facilities for vehicle servicing, check-out, and component mating.

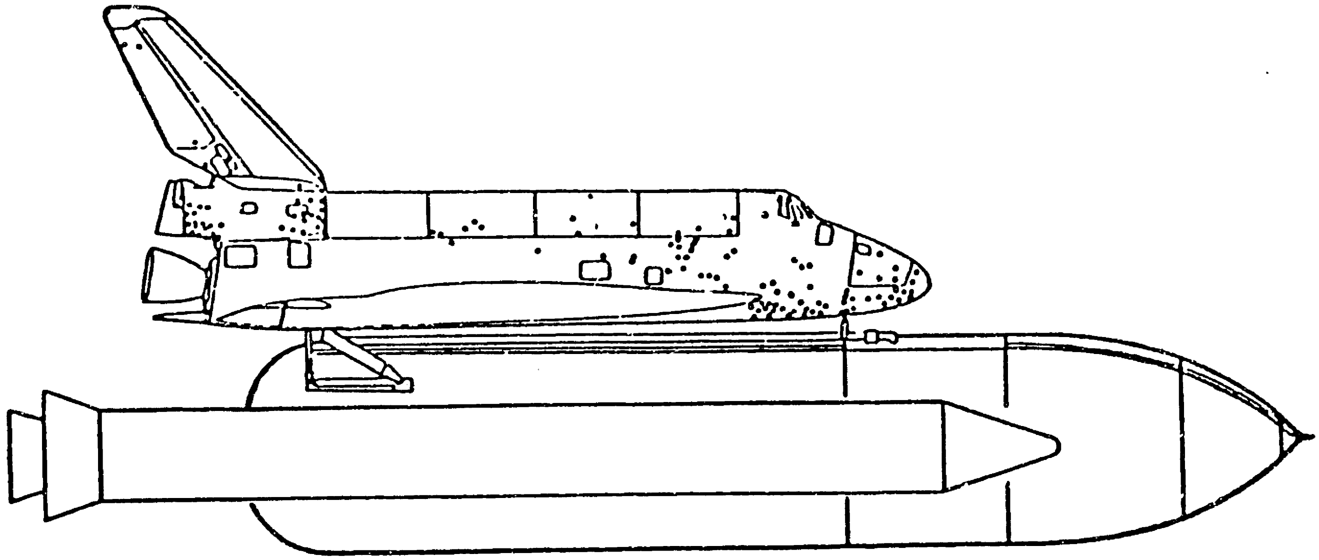
Second, even though the most massive piece of shuttle hardware, the ET, is discarded, its unit costs represents only 19 percent of recurring launch costs. Thus, expendability hurts the shuttle's balance sheet, but it does not ruin it. However, the costs of the ET together with the manpower costs account for fully two-thirds of the operations costs.

Third, even though the shuttle requires thousands of tons of liquid propellants (LOX/LH2), their costs are virtually insignificant. Thus, reducing the amount of propellant a launch vehicle needs pays off not in terms of lower fuel bills which lower operations costs, but as a reduction in vehicle gross lift-off weight (GLOW) which lowers development and fabrication costs.

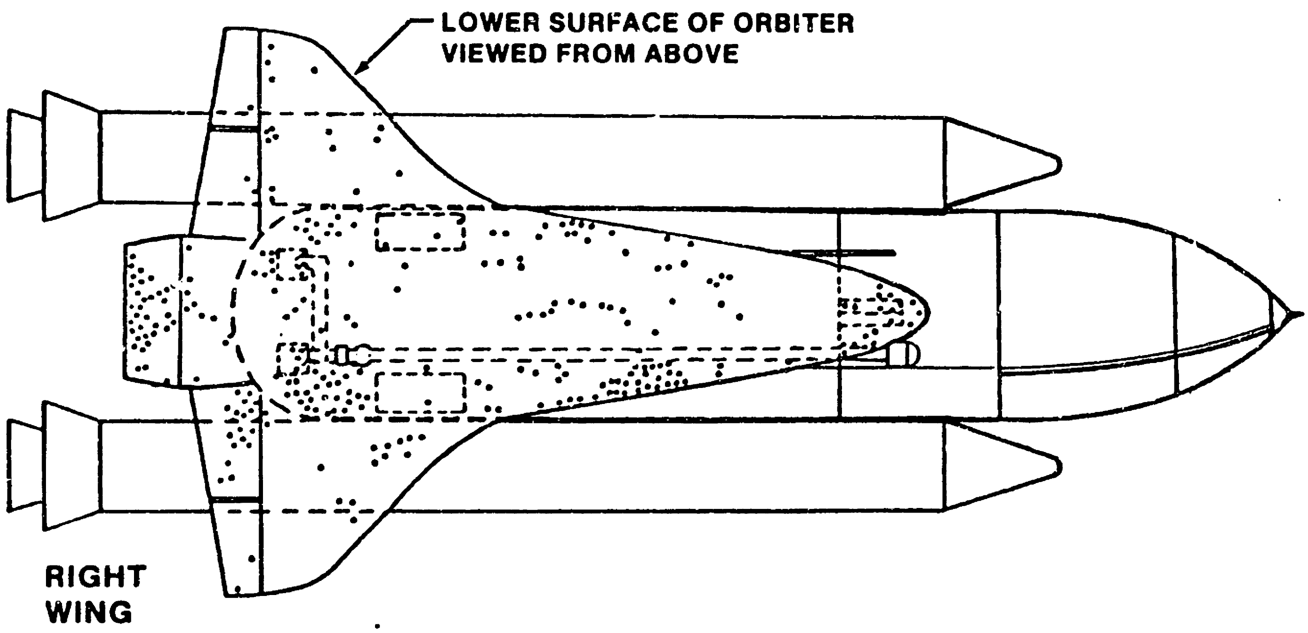
Shuttle Complexity and Design Deficiencies. Some of today's operational problems and expenses can be traced back to design compromises made in the early 1970s. These compromises arose from the need to satisfy three classes of users simultaneously: the commercial sector, the DOD, and the scientific community. The 60 foot cargo bay length was driven by the DOD's reconnaissance satellites, four or five times larger than typical commercial satellites. The result is that the payload bay often contains a diverse assortment of satellites, making payload integration and launch preparation complex and expensive.

The requirement that the orbiter be manned turned the shuttle into a miniature space station, necessitating a heavy crew compartment and attendant life support systems. The inclusion of a crew cabin also eliminated the possibility of integrating the cryogenic fuel tanks into the recoverable structure. Unfortunately, the solution to that problem, the external tank (ET), created more problems. First, as noted above, the expendable nature of the ET adds to the recurring operational costs of every launch, and second, by discarding the tankage, the shuttle orbiter is made more dense upon re-entry. Higher density during re-entry results in higher skin temperatures. To protect the orbiter skin from these high re-entry temperatures (up to 1,925 degrees K), designers turned to ceramic tiles, possessing a low heat transfer coefficient but a high mechanical fragility. These tiles, while managing to protect the shuttle from the heat of re-entry, have been extremely troublesome. The tiles must conform to the irregular shape of the shuttle orbiter's skin with a gap of no more than 0.025 to 0.075 inches between tiles. Tiling the first shuttle orbiter, Columbia, required 31,000 tiles and 335 man-years to complete. Furthermore, during lift-off, ice debris from frozen condensation on the ET have been known to damage the tiles, adding to the ground turn-around time (GTAT) and refurbishment costs. GTAT was averaging two months, far from the original projection of two weeks. Figure 1-4 (Craig, p. 171) shows a composite profile of tile damage on the first two shuttle flights.

The shuttle's cost-effectiveness has also been weakened by its not being up to specs in two crucial areas, engine performance and orbiter weight. The 12 million horsepower Space Shuttle Main Engines (SSMEs) have failed to meet their design goals. The SSMEs, composed of 70,000 parts, are the most sophisticated and most complicated liquid propellant rocket engines ever attempted. A new rocket engine commonly takes more time and money to develop than was planned. Usually, however, the engines deliver more thrust than expected. To the distress of the DOD and space scientists, the SSMEs are not providing the 109 percent full power level required for more demanding missions like the launch of strategic reconnaissance spacecraft and Spacelab into polar orbit. They are also wearing out faster than anticipated. The SSMEs were intended to provide about 55 flights before being overhauled and to require only



STS-1 and STS-2 right-side debris damage composite.



STS-1 and STS-2 lower surface debris damage composite.

Figure 1-4: STS-1 and STS-2 Debris Damage Composite minimum maintenance between flights. Figure 1-5 (from Toner, 1985, p. 67) presents a status report on the SSMEs showing that Kennedy Space Center (KSC) ground crews are spending more time repairing the complex engines than performing scheduled routine engine maintenance. The high pressure turbopumps have been particularly troublesome.

The SSME problems are exaccerbated by the shuttle orbiters' weight being 18,000 pounds heavier than the design goal, restricting payload capacity to 47,000 pounds, instead of the intended 65,000 pounds. To compensate for the short-falls in thrust and payload capability, NASA is developing a liquid fueled propulsion system to be mounted underneath the ET.

To be fair, one must acknowledge that the shuttle is still young and the problems identified above are being gradually resolved. The orbiter, the ET, and the SRBs are all getting lighter as composite materials are incorporated into their structures. Furthermore, the shuttle has unique capabilities for on-orbit satellite repair and servicing which have only begun to be exploited. It may also be argued that the investment in the shuttle was necessary to maintain the United States' lead in advanced space technology.

International Competition. This technology lead, however, has done nothing to stave off international competition in the launch vehicle market. Competition compounds the shuttle's problems by maintaining pressure on the U.S. to keep shuttle user fees low. With the shuttle fleet temporarily out of service, satellite owners are looking at the

Major unexpected work on main engines	Engine Maintenance (man-days) Routine	Repair
STS-1: Columbia, 4/81	maiden flight	
STS-2: Columbia, 11/81 High pressure fuel pumps removed, inspected; main fuel valve replaced; fuel burner and engine nozzle repaired.	600	840
STS-3: Columbia, 3/82 High pressure oxidizer pump removed and repaired.	207	363
STS-4: Columbia, 6/82 High pressure fuel pumps inspected, two replaced; failed engine controller and low pressure oxidizer pump replaced.	195	465
STS-5: Columbia, 11/82 High pressure fuel pumps inspected, one replaced; high-pressure oxidizer pump and main fuel valve replaced; engine nozzles repaired.	192	438
STS-6: Challenger, 3/83 Although engines had previously been used only in ground tests, all three had to be repaired or replaced due to cracks in the hydrogen fuel system.	maiden flight	
STS-7: Challenger, 6/83 High pressure fuel pumps removed, two repaired, one replaced.	200	280
STS-8: Challenger, 8/83 Cracked high pressure fuel pump repaired.	170	70
STS-9: Columbia, 11/83 Low pressure fuel pump and three engine controllers replaced.	270	470
STS-10: cancelled at the request of the US Air Force.		
STS-11: Challenger, 2/84 High pressure fuel pump removed, inspected; one engine and one engine controller replaced; fuel preburner and nozzles repaired.	200	400

Figure 1-5: SSME Maintenance Record
(continued on next page)

Major unexpected work on main engines	Engine Maintenance (man-days) Routine	Repair
STS-12: cancelled due to changes in the cargo manifest.		
STS-13: Challenger, 4/84 One engine replaced due to preburner erosion; one engine controller and two gaseous oxygen-control valves replaced.	150	155
41D: Discovery, 8/84 Two engines replaced.	370	910
41G: Challenger, 10/84 Three engines removed for modification.	150	1,250
51A: Discovery, 11/84 High pressure fuel pump and main-engine controller replaced.	170	340
Total man-days	2,874	5,981
Average per flight	205	427

Figure 1-5: SSME Maintenance Record (cont.)

French Ariane for launch services. Not too far away are upgraded Arianes, Japan's N-2, H-1 and H-2, and even commercial versions of the Soviet Proton, and China's Long March 2 launch vehicle.

Lessons Learned. To summarize, the space shuttle has taught us five important lessons in launch vehicle design. If space transport is to achieve the efficiency and cost-effectiveness that air transport has, our next-generation launch vehicle must:

(1) Sharply reduce required launch manpower. This goal must be unequivocally articulated in the vehicle design philosophy. Current design philosophy has been characterized as, "We'll build the damn thing; *you* figure out a way to launch it!"

(2) Have a higher launch frequency. This would amortize fixed costs over a greater number of launches. The implication is that payload bays must be sized to be more in line with the actual sizes of satellites.

(3) Have excellent technical performance. A fragile thermal protection system and an unreliable engine system has greatly increased the shuttle's maintenance and repair requirements and increased its ground turn-around time (GTAT).

(4) Be fully reusable. It was demonstrated in this analysis that expendable systems seem to have reached the limits of their performance and that further reductions in launch costs to LEO will require fully reusable vehicles.

(5) Specialize as a low-cost launch vehicle. Perhaps the most important lesson the shuttle has taught us is that a space vehicle designed to perform many functions is optimized for none.

1.4 A "New" Idea in Space Transportation

It may seem premature to be discussing the prospects for a next generation launch vehicle. Certainly, it makes sense to get the most mileage possible out of existing launch vehicles. Nevertheless, a revolutionary idea in launch vehicle technology is stirring in certain government agencies, congressional committees, and among several aerospace contractors. The idea is to propel a hybrid aircraft-spacecraft at very high speeds through the fringes of the atmosphere, using air-breathing scramjet engines, instead of rocket engines. It is not a new idea, but it is being looked at anew in light of recent technological advances. This concept is attracting widespread attention because it may have applications to a broad range of civilian and military aerospace objectives. Air Force officials have stated that in a launch vehicle configuration, a scramjet-powered "aerospace plane" could have the potential to reduce launch costs to LEO by a factor of 100, to the vicinity of $50 per kilogram (Of course, such predictions are difficult to evaluate because they have imbedded in them many unstated assumptions regarding vehicle fleet size, launch rate, payload bay size, and so on. In the cost analysis in chapter 3, my assumptions regarding these parameters may not be the same ones used by the government in arriving at $50 per kilogram, but at least they are explicitly stated and the analysis provides a framework for further cost calculations based on updated estimates of the various system parameters).

The proposed vehicle has been on the drawing board in one form or another for over twenty years under a variety of names, but whatever the name, the technology requirements have been very similiar. In its military configuration the proposed vehicle concept has gone by such names as Military Aerospace Vehicle (MAV), Orbit-on-Demand Vehicle (OODV), Advanced Military Spaceflight Capacity (AMSC), Advanced Aerospace Vehicle (AAV), Project "Copper Canyon," and most recently, as the Transatmospheric Vehicle (TAV). Civilian versions include the Hypersonic Transport (HST) or "Orient Express" and the Single-Stage-to-Orbit (SSTO) vehicle. The exact meanings of these terms are somewhat fluid as the concepts get fleshed out, so unless specified otherwise, throughout this thesis "aerospace plane" (ASP) shall refer to the generic technological concept of a scramjet-powered hypersonic vehicle, though the focus of this thesis shall be on the ASP as a launch vehicle. The following sections present an overview of the ASP concept and a brief history of its development.

1.5 ASP Concept

Seventy-five percent of the space shuttle's 4.4 million pounds gross lift-off weight (GLOW) is propellant mass, and of this propellant mass, 83 percent is due to the oxygen. From an efficiency stand-point, a launch vehicle carrying oxygen on-board while travelling through an atmosphere rich in oxygen, is like a fish in the ocean carrying a canteen of drinking water with him. Twenty percent of the atmosphere is oxygen, and that can be had for free. The essense of the ASP is to rely on the oxygen in the air to oxidize the fuel while in the atmosphere, and then switch to rocket propulsion when the air becomes too sparse to support combustion. Small rockets will, of course, still be necessary for the final acceleration, on-orbit maneuvers, and for re-entry initiation. Although some on-board oxygen would still be necessary for the rocket mode, the weight savings would be enormous.

The weight savings in oxygen would result in a number of other improvements. Single-stage-to-orbit, horizontal take-off and landing would allow a much greater flexibility in basing. In a two stage system, the first stage must either incorporate a propulsion system for fly-back to the launch site, or ballistically land in an ocean. That is why our two space launch sites, Kennedy Space Center and Vandenberg Air Force Base, are located on coasts. Freed of the burden of heavy oxygen tanks, the ASP could be designed to take-off and land at large airports, coast to coast, like a regular plane. These airports would have to have special facilities for cryogenic fuel production and storage.

Being an aerodynamic vehicle, the ASP should have a much better abort capability than the shuttle and, unlike the shuttle, which must be mated to a special Boeing 747 for transportation, the ASP would have an inherent ferry capability. Horizontal take-off has another inherent advantage. If the engines fall slightly short of their design thrust level, instead of sacrificing payload to reach orbit or strapping on extra boosters, one could simply extend the runway a little. Horizontal payload integration and launch processing might also be cheaper than the current vertical processing arrangment. These capabilities lead to ASP operational scenarios like launch on demand from conventional airports, delivery of high priority cargos to space, extensive maneuvering in the upper atmosphere, and return to an airport of choice anywhere on the earth.

The concept does, however, impose some design penalties. Designed to take advantage of atmospheric oxygen, the more complex air-breathing engine system (ABES) is likely to be heavier per pound of thrust than conventional rocket engines. Also, heavily strengthened landing gear would be necessary to support the aircraft at launch when its weight is nearly five times as much as at landing. These and other design issues will examined in chapter 2.

1.6 Propulsion Technology Overview

ASP development would challenge engineers in all aerospace disciplines, including propulsion, aerodynamics, materials, structures, avionics, guidance, control, navigation, and systems. This section will explain the technological underpinnings of the most critical technology to ASP development--propulsion. Propulsion is the factor that will make or break the plane because as currently envisioned, the entire underside of the airframe would be part of the engine.

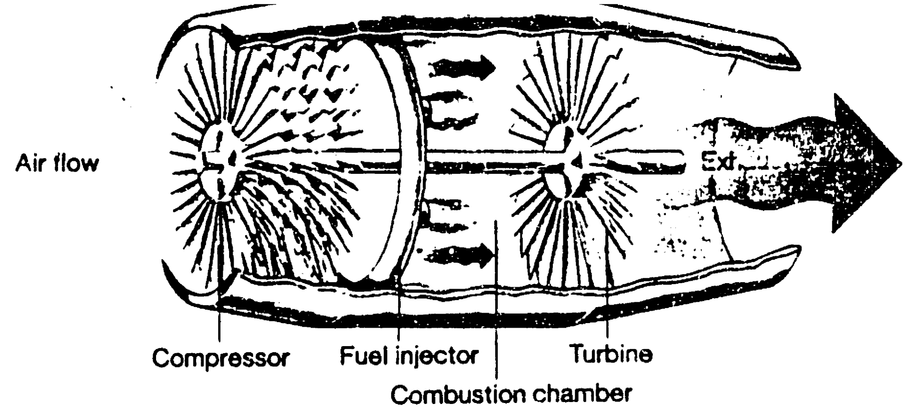
Research at NASA-Langley has centered around the airframe-integrated vehicle concept shown on page 2. The reason for this blending of the engine modules with the underside of the vehicle is that at hypersonic speeds very large engine airflows are required for adequate thrust. Under these circumstances, major sections of the vehicle must serve as extensions of the engine inlet and nozzle. Prior to entering the engine module inlet, the airflow is slowed down and compressed by the shock produced by the bow of the ASP. Engine modules are mounted side by side on the underside of the vehicle towards the aft end in order to swallow the shock. The aft end of the ASP would provide more area for combustion product expansion, which would increase engine efficiency and reduce drag. At high speeds, an aircraft experiences a pressure differential between the front and rear surfaces that effectively creates drag. The engine exhaust in the integrated design would partially equalize this pressure imbalance. A new problem facing aircraft designers is that with the integrated engine-airframe approach, design changes in any single part of the engine or airframe result in changes in all the other parts.

ASP development is dependent on the refinement of two relatively new ABESs: the air turbo-ramjet (ATR) and the supersonic combustion ramjet (scramjet). Because of the complexity of these engines, it is necessary to review the fundementals of their predecessors, the turbojet and the ramjet. Schematics of these engines are presented in figure 1-6.

Turbojet. These engines are found on most airliners. A turbojet uses a compressor to raise the temperature and pressure of the air prior

TURBOJET

As the air enters, it's compressed, mixed with fuel, and ignited. The burning gases expa. , powering the turbine, which drives the compressor. Escaping at high speed, the gases push the plane on. Most airliners have turbojets.



RAMJET

In this type of jet engine, the incoming air is traveling so fast it doesn't require a compressor: "ram pressure" alone can drive the aircraft. But ramjets don't develop enough thrust until the plane reaches supersonic speeds.

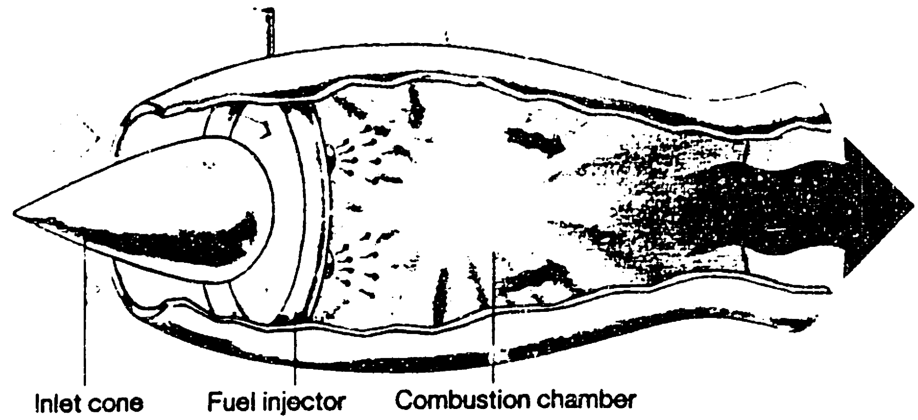


Figure 1-6: Turbojet and Ramjet Engines

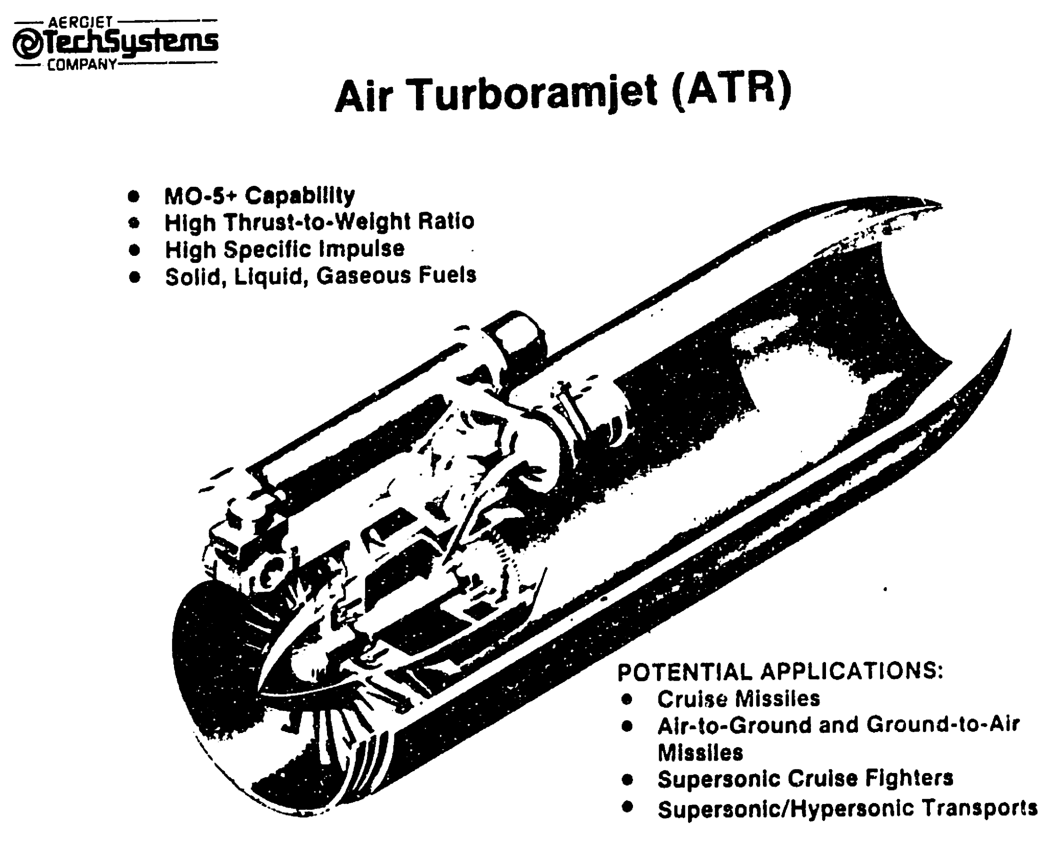


Figure 1-7: Air Turboramjet Engine

to combustion to improve its Brayton cycle efficiency. The burning gases then drive a turbine which is connected by a shaft to the compressor. At speeds of about Mach 3, the combined effects of aerodynamic heating and combustion raise the temperature of the gases which turn the turbine to about 1500 degrees K. This is the maximum temperature sustainable by today's turbines employing state-of-the-art materials and cooling techniques. Thus, Mach 3 is the practical limit for conventional turbo-jet engines.

These temperature limitations restrict the turbine inlet temperature to values below that corresponding to the stoichiometric mixture of fuel and air in the combustor. The exhaust gas contains a significant amount of residual oxygen. This oxygen can be used to provide additional thrust through the use of an after-burner. An after-burner is a second combustion chamber, downstream of the turbine, where additional combustion and expansion occurs. This fuel is used less efficiently than fuel in the primary chamber, because it is burned at lower temperatures and pressures. After-burners are therefore only used for brief bursts of speed. At Mach numbers of 2.5 or more, after-burners become much more efficient than at subsonic speeds.

Ramjet. The ramjet is conceptually the simplest of aircraft engines. In a ramjet, air enters the diffuser, is slowed down to subsonic speeds (typically $M = 0.2$), mixes with fuel in the burner, is ignited in the combustion chamber, and escapes through the rear nozzle, imparting forward momentum to the aircraft. A ramjet dispenses with the turbo-machinery used by a turbo-jet to compress the incoming air flow, relying instead on the ram-pressure of the incoming air stream for compression. Consequently, ramjets cannot produce static thrust and are unable to operate until the aircraft reaches high subsonic speeds. Some other form of propulsion is required to bring the vehicle up to speed. Once at these high speeds however, the so-called ram-pressure alone can sustain the ramjet. Like turbo-jets, ramjets have an upper speed limit. At Mach numbers above about 6, ramjets lose their effectiveness because the combustion temperature in the subsonic combustion chamber becomes so high, nearly 1800 degrees K, that the combustion products form a variety of partially burned fragments which do not contribute their full energy to the engine cycle.

Air turbo-ramjet. The ATR was invented by Aerojet TechSystems Company of Sacramento, California 37 years ago. Because of the ATR's possible defense applications, the US Patent Office issued a Secrecy Order on the patent application. In 1963, that order was removed and patent 3,110,153, entitled "Gas Generator Turbojet Motor" was issued to Mr. William C. House of Aerojet.

The ATR, shown in figure 1-7, is a continuous-flow air-breathing engine which involves air compression, heat addition, and expansion through a thrust nozzle, like any other engine. It is unique in that from Mach 0 to Mach 2 the engine uses the compressors to mechanically compress the incoming air, but between Mach 2 and Mach 6, it relies on ram pressure.

The turbine of the turbocompressor is driven by high pressure/high temperature fuel-rich gas from a gas generator. The crucial design feature of the ATR is that air coming from the compressor does not go to the turbine, but instead gets mixed with the fuel rich gas coming from the turbine, and burns in a ramjet style combustion chamber. By keeping the inlet airstream away from the turbine and having primary combustion take place downstream of the turbine, the turbine temperature is made independent of the flight Mach number. The temperature of the fuel-rich gas is about 1100 degrees K no matter how fast the airplane flies. This is commonly referred to as "cold turbine" technology.

These engines reach their performance limit at about Mach 5 or 6 when the temperature rise resulting from the incoming air being abruptly slowed down in the combustion chamber reaches about 1900 degrees K. At these temperatures, even the non-moving parts of the engine will overheat and fail.

ATRs have only been tested in the laboratory but show high efficiencies and a high thrust-to-weight ratio. Three ATR demonstrator engines have already been built and tested. In 1955 a 26 inch diameter ATR was built for the USAF. This engine developed 3750 lbs of thrust at sea level conditions. As of 1964 the ATR began to win acceptance as the powerplant of choice for the SST. The Navy has been interested in ATRs for a long time as a means of propelling target vehicles at low cost. In 1982, a 15 inch demonstrator engine providing 1000 lbs of thrust was built and tested for the Navy. In 1983 a flight-weight, low-cost version of the ATR was built for the Navy for missile applications. These ground-test results still need to be verified at simulated ASP altitudes and Mach numbers, however. Currently, DARPA, NASA-Lewis, and the Aerojet Corporation are working on demonstrating a full-scale, flight-weight ATR for ASP applications.

Scramjet. Like the ATR, scramjets have only been tested in laboratories. For speeds above Mach 6, the scramjet is the only possible ABES. A scramjet is a variant of the ramjet, the difference being that in a ramjet combustion occurs at subsonic speeds, while a scramjet maintains supersonic speeds throughout the engine. By maintaining the airflow at supersonic speeds within the engine, the ram-down effect is reduced and aerodynamic heating is minimized.

The scramjet's operating range is potentially the widest of all jet engines. It only begins to work at speeds above Mach 4, twice as fast as the Concorde travels. Laboratory experiments have not determined an upper limit on the flight speeds capable with scramjets, however some propulsion specialists say they should operate up to orbital speeds, around Mach 24 or about 7.5 km/sec. The current NASA-Langley fixed-geometry, airframe-integrated scramjet is shown in figure 1-8, and a close-up view of the world's first working scramjet model is shown in figure 1-9 (courtesy of the General Applied Sciences Laboratory).

For a long time, the greatest impediment to scramjet development was achieving combustion in a supersonic flow. The task was compared to "lighting a match in a hurricane." At high speeds, such as Mach 7, a fuel particle stays in the combustor for less than a millisecond.

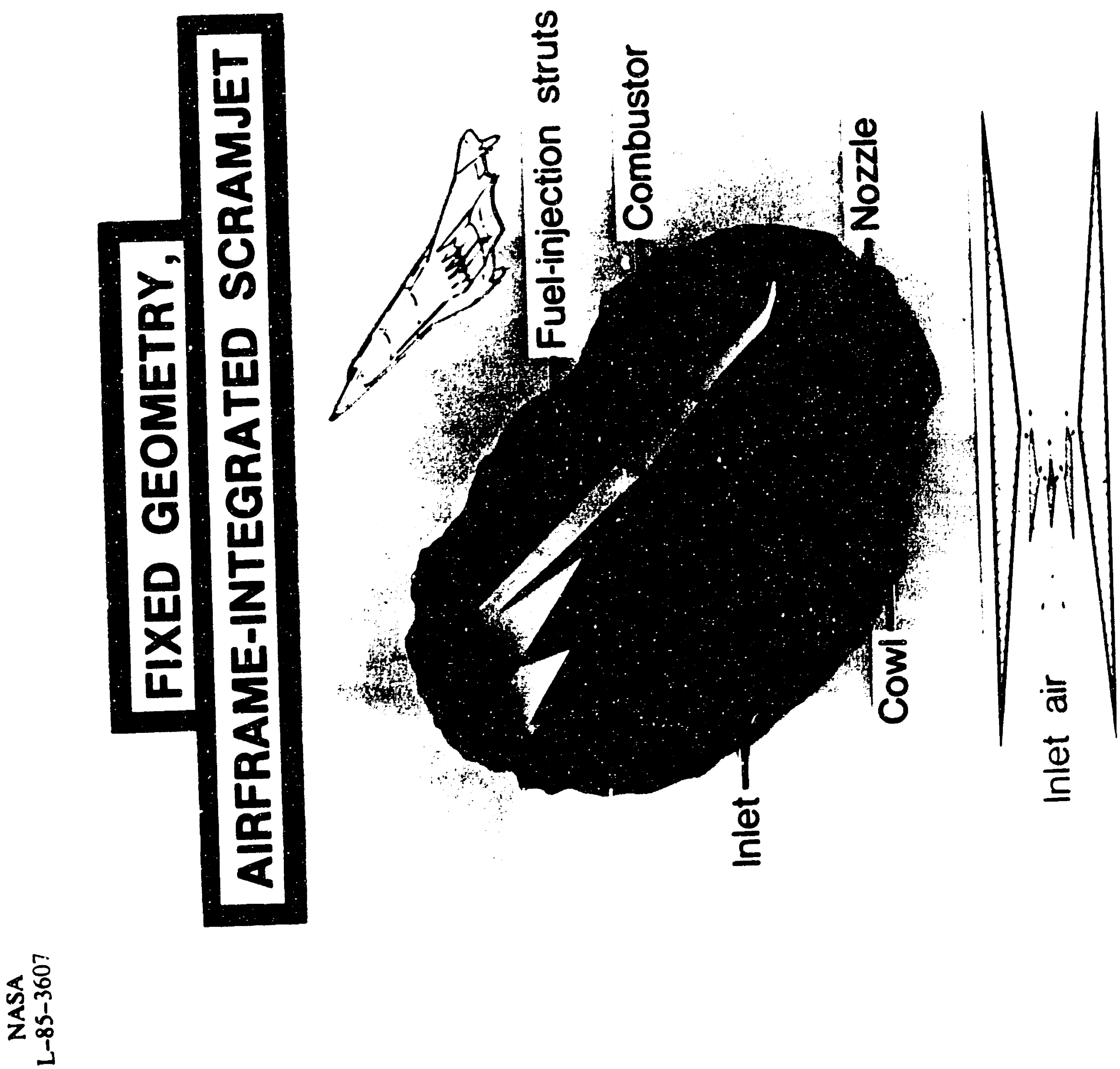


Figure 1-8: NASA Airframe-Integrated Scramjet

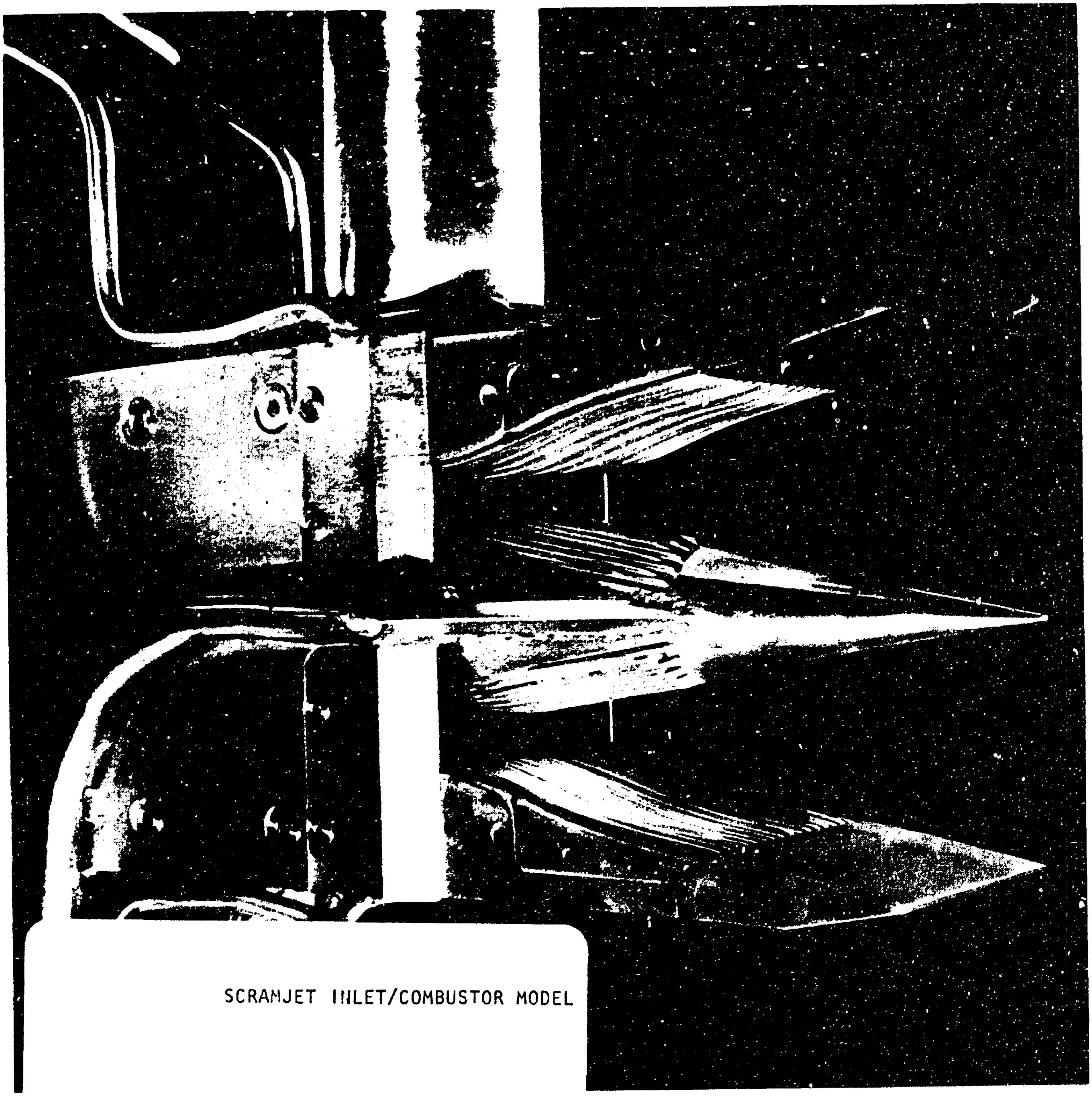


Figure 1-9: First Working Scramjet Model

Traditional hydrocarbon fuels do not burn quickly enough, resulting in engine "flame out."

The answer appears to be to use hydrogen, which burns much faster than hydrocarbon fuels. In addition to its rapid combustion rate, hydrogen has other advantages over alternative fuels. First, hydrogen has the highest specific energy content of any fuel. For a given mass, hydrogen has 2.3 times the energy content of conventional jet fuel, kerosine. Second, hydrogen has excellent heat absorption, an important characteristic at high temperature, high Mach number flight. Other considerations are: hydrogen combustion is non-polluting--its product is water; the Hindenburg notwithstanding, hydrogen burns with a non-propagating flame unlike kerosene; and hydrogen can be made from natural gas or water, without increasing our dependence on foreign fuel supplies.

The primary drawback of using hydrogen fuel is that it is less dense than hydrocarbons and consequently requires larger tanks. Kerosine is 11.6 times as dense as liquid hydrogen, but only 1/2.3 times as much hydrogen is needed to provide the same amount of energy. Therefore, hydrogen requires fuel tanks that are 11.6/2.3 or 5 times as large as equivalent hydrocarbon fuel tanks.

The hydrogen would be stored cryogenically in liquid form, circulated through the engine and airframe to keep them cool, and injected into the airstream in gaseous form to speed combustion. Conventional ramjets inject fuel from the engine walls, which remain relatively cool. To achieve better fuel mixing, NASA plans to mount fuel injectors on struts spanning the airflow. These struts would be subject to very high temperatures and would be made of high-temperature metal, and cooled by the circulating hydrogen.

NASA researchers say that the scramjet fuel injection struts simultaneously present the most formidable cooling and structural problems of the ASP (Dixon, 1985, p. 53). At the same time, the struts must "support a large side load, contain high-pressure hydrogen at two temperature extremes, and withstand the high thermal stresses resulting from complex aerodynamic heating as well as convective heating from the hot hydrogen in the internal manifolds. To compound these problems the cross-sectional area and contour cannot be altered without significantly changing the engine propulsion performance."

So far, the NASA scramjet test engines have developed sufficient thrust to accelerate a large aircraft to Mach 7--the maximum speed of the NASA-Langley wind tunnel. Modifications to wind tunnels will permit ground testing of scramjets up to about Mach 9. Beyond that, flight tests will be necessary. And what is the likelihood that scramjets will work in flight? The manager of DARPA's hypersonic technology program, Robert Williams has said, "While a thousand tests of scramjets give me a lot of good feelings, it isn't until we actually build that thing and test it over a couple of years that we'll be able to say it's mature enough to go ahead. Still, we don't see any show-stoppers (Grey, 1986, p. 79)."

1.7 ASP Historical Background

Some of the theory behind the ASP has been around for a long time. NASA's exploration of hypersonics began in the early 1950s with some simple experiments to show shock waves in a supersonic combustion stream. Hypersonic flight research began in the late 1950s with the X-15 program. Twenty-eight years ago, the X-15 set the still-unbroken world aircraft records for high-altitude flight at 108 km and speed at Mach 6.7. At this time, supersonic combustion ramjets were called hypersonic ramjets. At a 1963 AIAA Conference, a Johns Hopkins University Scientist, Dr. William Avery, coined the term "scramjet."

From 1965 to 1974, NASA's Hypersonic Research Engine (HRE) program at NASA-Langley, a $40 million effort, conducted ground tests on hypersonic ABES. A prototype scramjet was scheduled to be mounted on an X-15 and flight-tested, but the X-15 program was scrapped and the flight test abandoned. Since then, NASA-Langley has demonstrated practical scramjet operation up to Mach 7, the testing limit of their facilities.

Another project which never materialized, but nonetheless contributed to the ASP, was the Dyna-Soar boost-glide space vehicle studies of the early 1960s. The Dyna-Soar was to be a manned vehicle capable of orbital flight and manoeuvrable re-entry and landing with a cross-range capability of several thousand miles. Like today's ASP, the Dyna-Soar was intended as a quick-response, surveillance, strike, and logistics system. However, critics complained the program lacked a clear definition of its ultimate goal, so the USAF redesignated the

Dynasoar as the X-20A to emphasize its research functions. In 1963 Secretary of Defense McNamara cancelled the X-20A, favoring the Manned Orbiting Laboratory. Still, the Dynasoar/X-20A program provided a $783 million technology base on large, hot structures.

In the mid-1970s a cooperative effort between NASA and the USAF almost produced a Mach 8 research aircraft for testing propulsion, aerodynamics, and structures technology. This National Hypersonic Flight Research Facility (NHFRF) was terminated because of a lack of clearly defined mission requirements within the USAF.

The US Navy has also conducted a great deal of missile-oriented hypersonic research in conjunction with the Applied Physics Laboratory of Johns Hopkins University and the China Lake Weapons Center. These efforts have been shrouded by security classification, however.

The space shuttle has provided some data on hypersonic flight, but its launch and reentry paths are very different from those for a vehicle powered by an ABES. Just prior to the suspension of shuttle flights, a more intensive program, the Shuttle Orbiter Experiments (OEX), was underway to gather data on hypersonic flight using the shuttle.

Overall, U.S. involvment in hypersonic research has been characterized by a cycle of ambitious but short-lived starts and a series of paper studies. Figure 1-10 presents this history as seen by Patrick Johnston, an engineer at NASA-Langley.

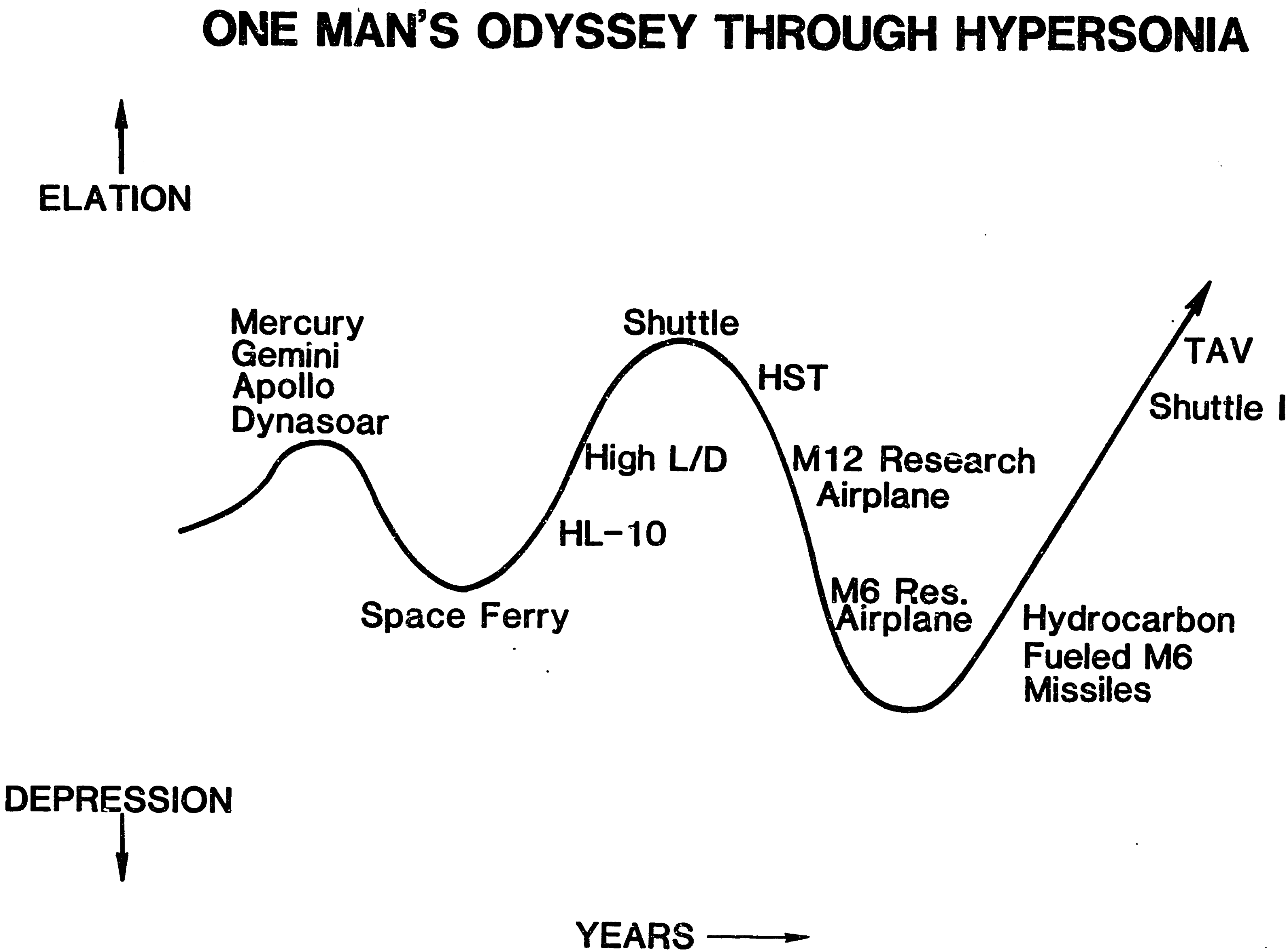


Figure 1-10: Vicissitudes of US Hypersonics Research

Recent Developments. In his State of the Union address on February 4, 1986, President Reagan singled out development of the ASP as a high priority national goal. "We are going forward with research [on an aircraft] that could by the end of the next decade, take off from Dulles Airport and accelerate up to 25 times the speed of sound, attaining a low-earth orbit [and] fly to Tokyo within two hours," the President declared.

According to Lana Couch, NASA's ASP program manager, the ASP re-examination is warranted by advances in three crucial technology areas: air-breathing propulsion (scramjet engines), the creation of light-weight, high-temperature materials for use in the airframe and engine, and dramatic improvements in computational power that makes it possible to integrate components in the design stage and simulate three-dimensional high-speed aerodynamic fluid flow (computational fluid dynamics).

In January 1986, a $600 million, coordinated effort to design a ASP in detail began. Government agencies participating in the three year "technological development" program include the the Defense Advanced Research Projects Agency (DARPA), NASA, the USAF, the Navy, and the Strategic Defense Initiative Organization (SDIO). Funding for the program is split 80/20 between the DOD and NASA. The USAF has been assigned overall DOD responsibility for the ASP research program and has established a joint program office at Wright-Patterson Air Force Base in Dayton, Ohio. Currently, the USAF plans to make their decision on whether to proceed with ASP development in 1991, at which time studies of various configurations will be completed and key technology areas will be demonstrated.

1.8 Summary

The space shuttle is likely to be the United States' premier launch vehicle through the end of the century. Unfortunately, the shuttle has failed to meet its economic objectives for many reasons--chiefly, the degree of technical complexity and redundancy necessary to insure crew safety was underestimated and the space traffic the shuttle would encounter was overestimated. Furthermore, cut-backs in development funding and the requirement that the shuttle serve multiple roles precluded its optimization as a low cost space transportation system. Regardless of what caused the problems, the net result is that today it costs nearly as much to launch a satellite as to construct it.

The prospects for the next generation advanced launch vehicle system may be brighter, however. Recently, interest has been revived in the aerospace plane (ASP), a vehicle concept which has looked at in one form or another for more than two decades. Enthusiasts claim that the major technological obstacles to ASP development have been overcome and that the vehicle has the potential to reduce launch costs to LEO by two orders of magnitude. Also, because of its extremely fast speeds, the ASP could have important civilian air transport and military implications. The ASP concept, therefore, deserves a detailed examination of its technological feasibility and utility for both civilian and military purposes.

CHAPTER 2: AEROSPACE PLANE SYSTEMS ANALYSIS

"The technology is here. It is not waiting to be developed. It is simply waiting to be used."
-Dr. George Keyworth, former Presidential Science Advisor

"There must be no misunderstanding--this will be the most complex vehicle ever built."
-Raymond S. Colladay, NASA deputy associate administrator

The first two sections of this chapter concern the performance of two ASP subsystems, propulsion and thermal protection. Both sets of calculations indicate that when designing for hypersonic speeds, the margin for error is slim. Acceptable and unacceptable performance levels are separated by a fine line. In section three, the ASP is examined from a macro perspective to estimate the vehicle mass fractions for the three different propulsion phases: ATR, scramjet, and rocket. In section four, these mass fractions are incorporated into an algorithm which produces estimates of the masses of the major ASP subsystems. These masses are then used in the cost estimating model of chapter 3.

2.1 ASP Propulsion System Performance

The net thrust produced by the scramjet will be the difference between the rate at which momentum leaves the vehicle and the rate at which momentum enters the vehicle as shown in figure 2-1.

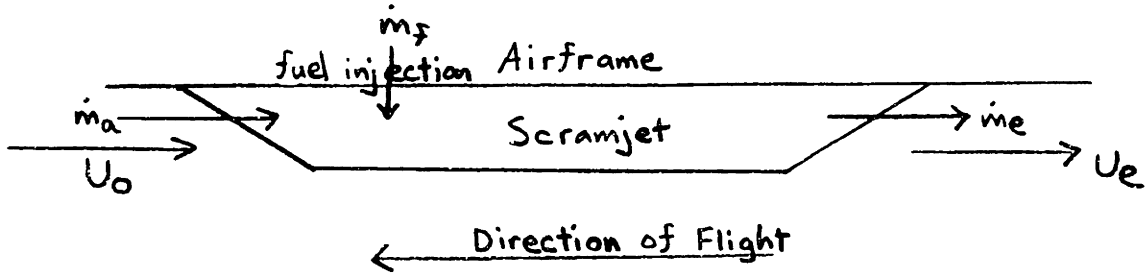


Figure 2-1: Scramjet Schematic

This can be expressed as:

(2.1) $$T = \dot{m}_e u_e - \dot{m}_a u_o + A_e(p_e - p_a)$$

where:

T = net thrust
$\dot{m}_e$ = mass flow rate of combustion products at exit
u_e = flow velocity at exit
$\dot{m}_a$ = mass flow rate of incoming air
u_o = flow velocity at inlet
A_e = area at exit
p_e = exit pressure
p_a = ambient pressure

The mass flow rate leaving the engine is merely the sum of the mass flow rates of the incoming air and the injected fuel, hence:

(2.2) $$\dot{m}_e = \dot{m}_a + \dot{m}_f$$

Expressed in terms of the fuel to air ratio, $f = \dot{m}_f / \dot{m}_a$, the thrust equation becomes:

(2.3) $$T = \dot{m}_a[(1 + f)\, u_e - u_o] + A_e(p_e - p_a)$$

Specific impulse is defined as the thrust output per unit weight flow rate of propellant. Thus:

$$\text{(2.4)} \qquad Isp = T/\dot{m}_f g = T/\dot{m}_a f g$$

In general, in equation (2.3), the contribution of the net pressure force term to the thrust is much smaller than that of the momentum-flux term and can be ignored. Substituting in equation (2.4) for T in equation (2.3) yields:

$$\text{(2.5)} \qquad Isp = [(1 + f)\cdot u_e - u_o]/fg$$

$$\approx u_o(u_e/u_o - 1)/fg, \text{ since } f \approx 0$$

Above Mach number 5, engines operate stoichiometrically, so the value of f can be calculated from the equilibrium equation:

$$\text{(2.6)} \qquad 2\ H2 + (02 + 79/21\ N2) \rightarrow 2\ H20 + 79/21\ N2$$

from which the fuel to air ratio is found to be 2(2)/[32+(79/21)28] = 0.0291. Equation (2.5) can now be rewritten as

$$\text{(2.7)} \qquad u_e/u_o = 1 + (0.0291\ Isp\ g/a_o M_o)$$

Substituting $g = 9.81$ m/sec^2, and $a_o = 295$ m/sec, yields

$$\text{(2.8)} \qquad u_e/u_o = 1 + (0.00097\ Isp/M_o)$$

Before we can evaluate this equation we need an expression for specific impulse as a function of Mach number. The SSMEs have an Isp of 455 secs, close to the 500 sec theoretical limit for a LOX-LH2 engine. Kerrebrock (1977, p. 251) and others have estimated the Isp of a hydrogen-fueled scramjet engine to be three to seven times as great as that of a LOX-LH2 engine (see figure 2-2). Figure 2-3 shows the specific impulse for the ATR, as measured by the Aerojet Company, the inventors of the ATR. From figure 2-2, a mathematical approximation for scramjet Isp as a function of Mach number can be derived for substitution into (2.8). The following relation was used:

(2.9) $$Isp = 4500 \cdot (1 - M_o/24)$$

Thus,

(2.10) $$u_e/u_o = 4.37/M_o + 0.818$$

Equation (2.10) is graphed in figure 2-4. As noted by Kerrebrock (1977), this graph indicates that the percentage change in airstream velocity from inlet to nozzle becomes very small for M_o greater than 10. The important consequence is that at very high Mach number flight, small unexpected inefficiencies in the inlet, engine, or nozzle could reduce the exhaust velocity to the velocity at inlet. If this occurs, zero net thrust would be produced and flight beyond this Mach number would be impossible with air-breathing engines.

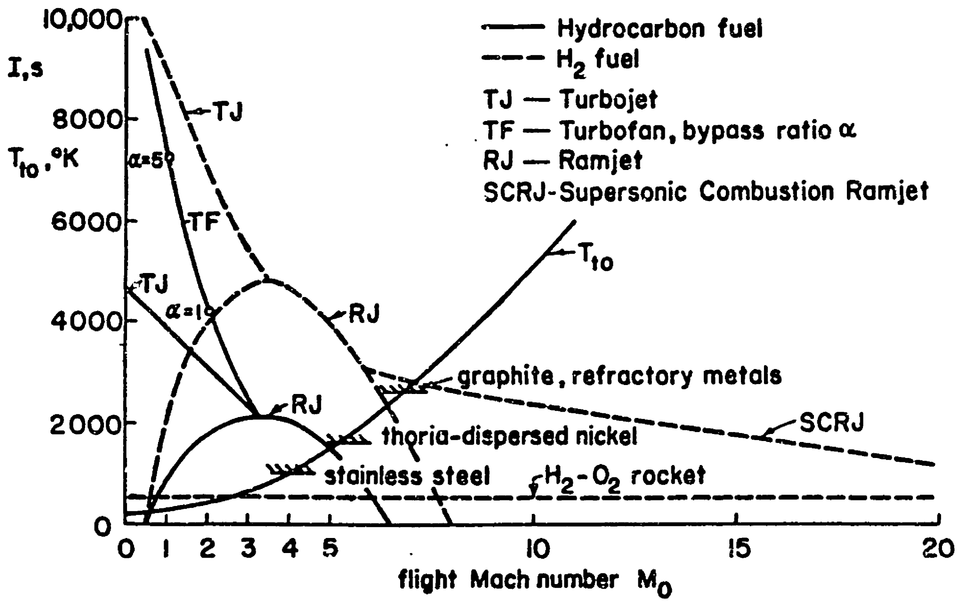


Figure 2-2: Specific Impulse vs. Mach Number for Various Engines

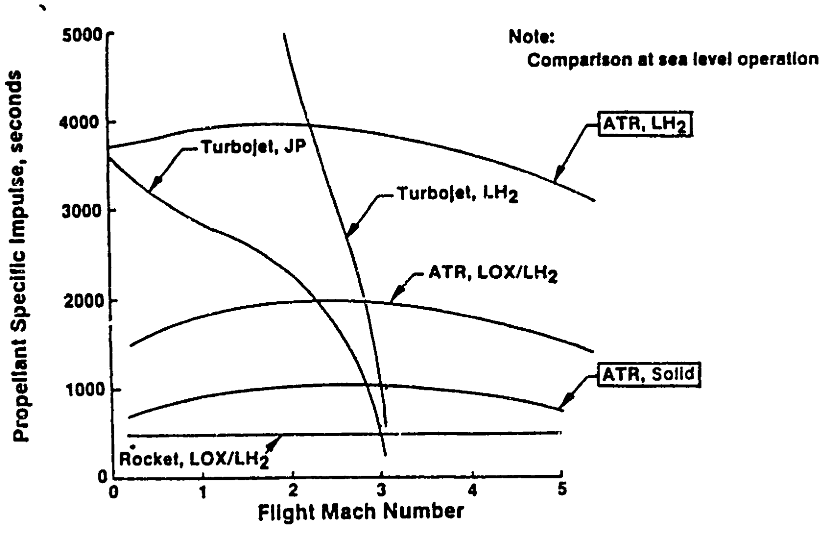


Figure 2-3: Specific Impulse vs. Mach Number for the ATR

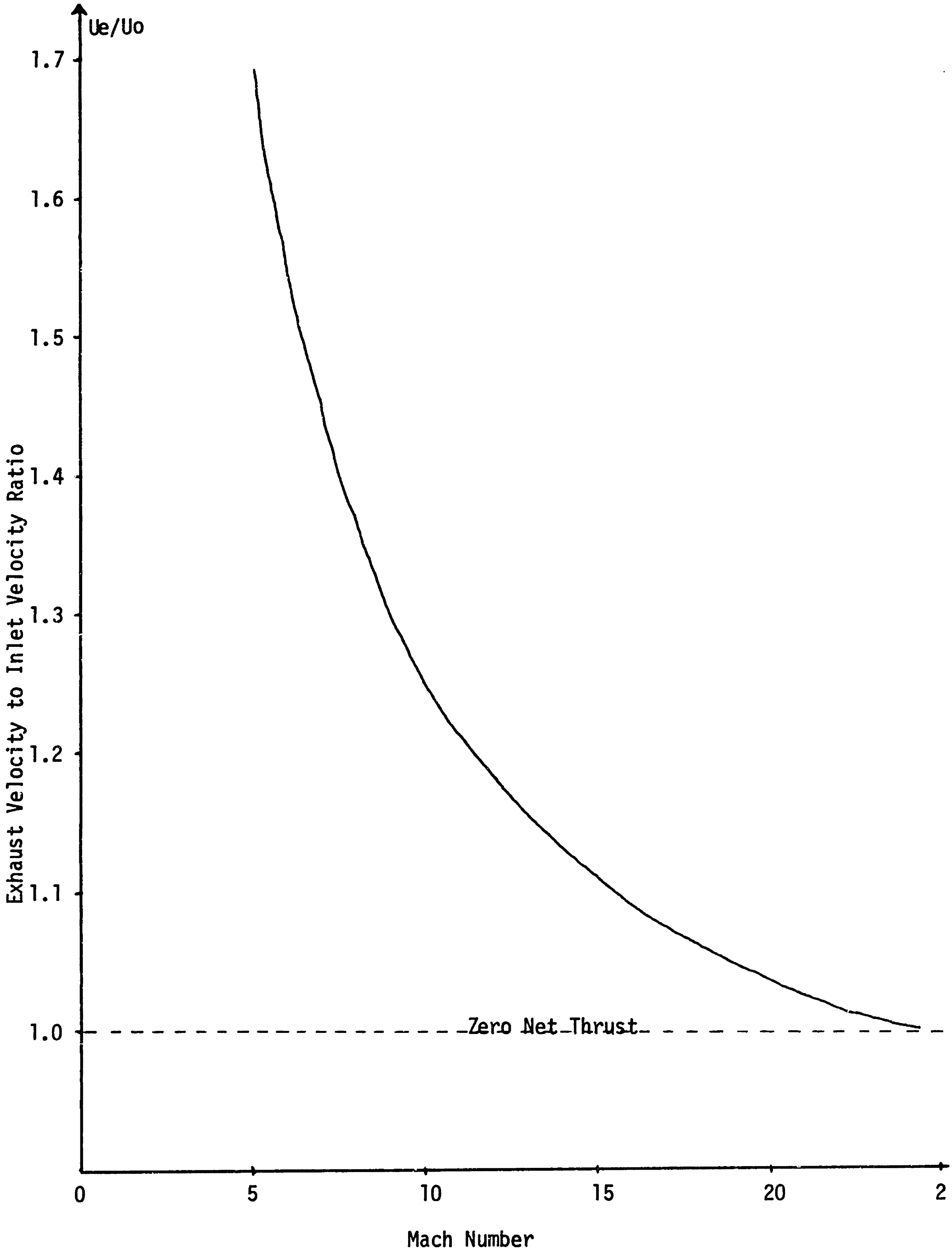


Figure 2-4: Reduction of Exhaust to Inlet Velocity Ratio at High Mach Numbers

2.2 ASP Thermal Protection System

The Problem. During launch and re-entry into the earth's atmosphere, high temperatures are generated by aerodynamic heating. The magnitude of the problem is indicated by the fact that most of the meteors that enter the earth's atmosphere burn up. Consider a vehicle in space moving at orbital speed, 7.5 km/sec. The vehicle has a kinetic energy per kilogram of mass of approximately 25 million Joules, which is 10 percent more than the amount of energy required to vaporize an equal mass of aluminum. Clearly, when this kinetic energy is converted to heat, only a small fraction of it can be allowed to heat the vehicle. The rest must be radiated away or absorbed by the cryogenic fuel.

Approximation. Using basic thermodynamic relations, a rough order of magnitude estimate of the thermal protection problem can be made. Thermodynamic equilibrium is maintained as long as the heating of the vehicle skin due to aerodynamic drag plus the heating of the scramjet engine is less than the cooling provided by heat radiated from the vehicle and the heat absorbed by the cryogenic hydrogen fuel. Thus, we require that:

$$\dot{Q} + \dot{Q}e < \dot{Q}r + \dot{Q}f \qquad (2.11)$$

where: $\dot{Q}$ = heat input to the vehicle surface due to drag

$\dot{Q}e$ = heat input to the scramjet engines

$\dot{Q}r$ = radiated heat from the ASP

$\dot{Q}f$ = heat absorbed by the fuel

First, we derive the expression for the aerodynamic heating. By definition,

(2.12) $$hf = \dot{q} / (Tw - T_{\infty}) = \dot{q} / Tt$$

where:

hf = heat transfer coefficient

$\dot{q}$ = heat flux/unit area

S = unit area

Tw = adiabatic wall temperature

T_{∞} = free-stream temperature

Tt = total temperature

Therefore, $\dot{q} = hf \cdot Tt$. But since $\dot{Q} = \dot{q} \cdot S$, we have $\dot{Q} = hf \cdot Tt \cdot S$. From the definition of the Stanton number, St, we have:

(2.13) $$hf = St \cdot (\rho \cdot Cp \cdot U)$$

where:

ρ = density

Cp = specific heat

U = free-stream velocity

and the subscript s means evaluated at the vehicle's surface. This gives:

(2.14) $$\dot{Q} = St \cdot (\rho \cdot Cp \cdot U \cdot Tt)_s \cdot S$$

We can approximate St as one half the skin friction coefficient since the

Prandtl number is approximately unity. Furthermore, the skin friction coefficient, Cf, is defined as stress (drag per unit area) divided by dynamic pressure, or $Cf = (D/S) / (1/2 \cdot \rho_o \cdot U_o^2)$. The subscripts o mean local free-stream conditions. Thus, the heating rate due to aerodynamic drag is:

$$\dot{Q} = D\,(\rho \cdot Cp \cdot U \cdot Tt)_s \,/\, \rho_o \cdot U_o^2 \qquad (2.15)$$

Let us set this expression aside for the moment and examine the radiated heat term, $\dot{Q}r$. The Stefan-Boltzmann laws says that heat radiated from a surface scales as temperature to the fourth power:

$$\dot{Q}r = \epsilon \cdot \sigma \cdot Ts^4 \cdot S \qquad (2.16)$$

where:

ϵ = vehicle emissivity

σ = Boltzmann constant = 5.67 x 10 W/m K

T_s = vehicle surface temperature

Now we calculate the heat absorbed by the fuel, $\dot{Q}f$. This can be expressed as the mass of the fuel flow per unit time, times the heat capacity of the fuel per unit mass of fuel. The first term is given by the definition of specific impulse: fuel mass flow rate equals thrust divided by g times Isp. The second term is 1.4×10^7 Joules/kg. Thus:

$$\dot{Q}f = 1.4 \times 10^7 \, (T/g \cdot Isp) \qquad (2.17)$$

The heating rate of the scramjet engines will be approximately determined from figure 2-5 (Henry and Griffin, 1972, p. 47). As one might expect, the engine combustor and nozzle components require the most cooling. One goal of the NASA-Langley research program is to reduce the length of these components and the associated cooling requirements. At a flight speed of Mach 6, the chart says that the scramjets require about 25 percent of the heat sink provided by the fuel.

We now have expressions for the four heating rates as functions of many variables. It is preferrable to reduce the number of variables through substitutions and approximations in order to facilitate calculations. To do this, we begin by dividing the heating rates by $D \cdot u_o$. For the first term, this gives:

$$\frac{\dot{Q}}{DU_o} = \left(\frac{\rho_s}{\rho_o}\right)\left(\frac{U_s}{U_o}\right)\left(\frac{C_p T_{ts}}{U_o^2}\right) \tag{2.18}$$

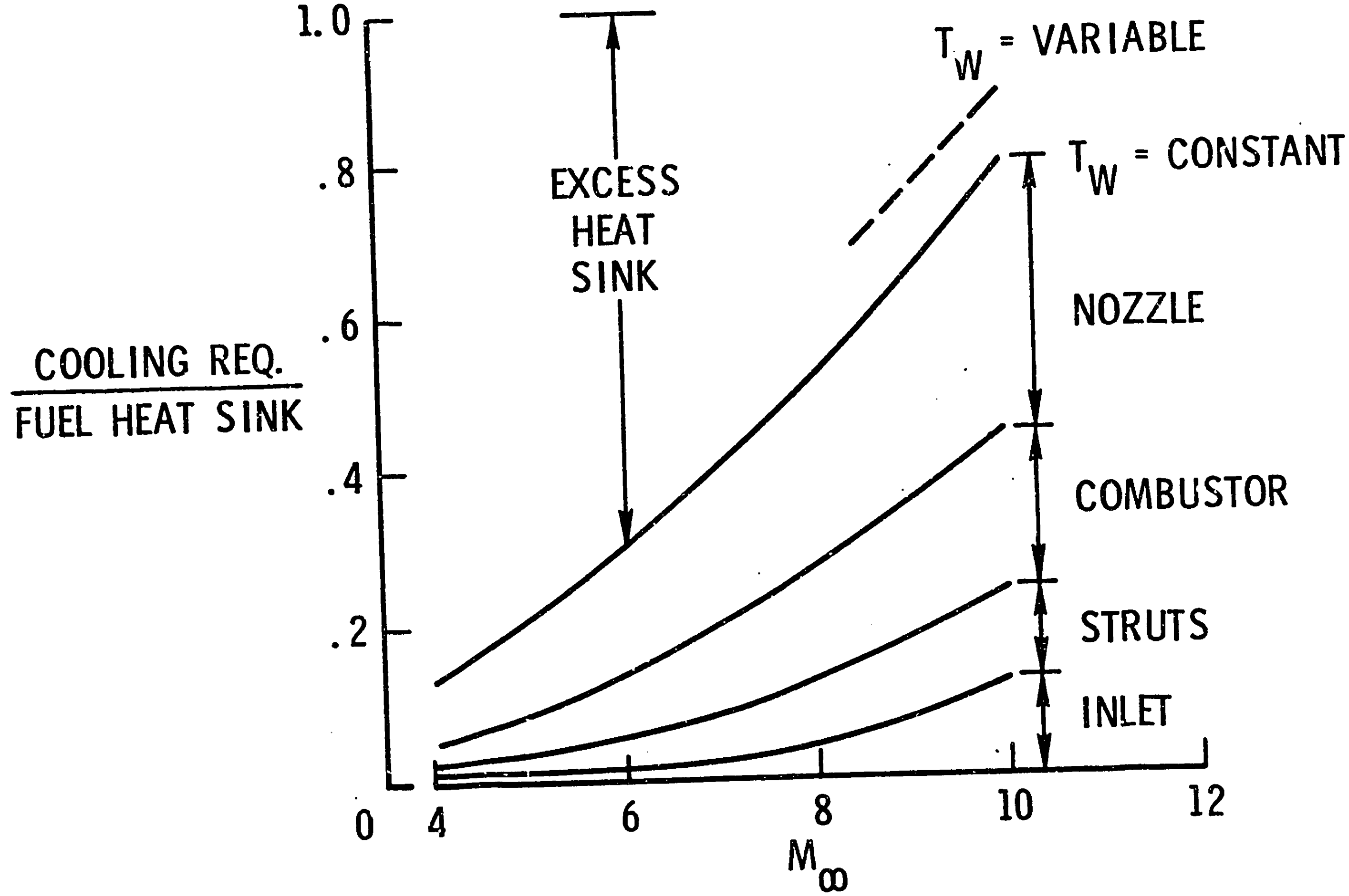
But,

$$\left(\frac{\rho_s}{\rho_o}\right) = \left(\frac{p_s}{p_o}\right)\left(\frac{T_o}{T_s}\right) \tag{2.19}$$

Following Kerrebrock's (1977, p. 22, 262) assumption that the aerodynamic conditions at the vehicle surface can be approximated by the conditions after inlet diffusion, we can substitute the following equations into the above equation:

$$\frac{p_s}{p_o} = \frac{p_{t2}}{p_o\left(1+\frac{1}{2}(\gamma-1)M_2^2\right)^{\gamma/\gamma-1}} \tag{2.20}$$

$$\frac{p_{t2}}{p_{to}} = \pi_d \tag{2.21}$$



Langley Scramjet cooling required. $q_\infty = 47.9\ kN/m^2$ (1000 psf), $\phi = 1.0$.

Figure 2-5: Proportion of Fuel Heat Sink Required to Cool Scramjet Engines

(2.22) $$\frac{p_{t_o}}{p_o} = \left(1+\tfrac{1}{2}(\gamma-1)M_o^2\right)^{\gamma/\gamma-1}$$

Thus,

(2.23) $$\frac{p_s}{p_o} = \pi_d \left(\frac{1+\frac{1}{2}(\gamma-1)M_o^2}{1+\frac{1}{2}(\gamma-1)M_2^2}\right)^{\gamma/\gamma-1}$$

And

(2.24) $$\frac{T_o}{T_s} = \frac{T_o\left(1+\frac{1}{2}(\gamma-1)M_2^2\right)}{T_{t_o}}$$

(2.25) $$\frac{T_o}{T_{t_o}} = \frac{1}{\left(1+\frac{1}{2}(\gamma-1)M_o^2\right)}$$

Thus,

(2.26) $$\frac{T_o}{T_s} = \frac{\left(1+\frac{1}{2}(\gamma-1)M_2^2\right)}{\left(1+\frac{1}{2}(\gamma-1)M_o^2\right)}$$

(2.27) $$\left(\frac{\rho_s}{\rho_o}\right) = \pi_d \left(\frac{1+\frac{1}{2}(\gamma-1)M_o^2}{1+\frac{1}{2}(\gamma-1)M_2^2}\right)^{1/\gamma-1}$$

(2.28) $$\left(\frac{U_s}{U_o}\right) = \frac{M_2}{M_o}\left(\frac{T_2}{T_o}\right)^{1/2} = \frac{M_2}{M_o}\left(\frac{1}{1+\frac{1}{2}(\gamma-1)M_2^2}\right)^{1/2}$$

(2.29) $$\left(\frac{C_p T_{t_s}}{U_o^2}\right) = \frac{1+\frac{1}{2}(\gamma-1)M_o^2}{(\gamma-1)M_o^2}$$

where:

γ = specific heat ratio

p_o = free-stream pressure

p_{t_o} = free-stream stagnation pressure

p_{t_2} = stagnation pressure at engine inlet

π_d = diffuser pressure recovery ratio

T_o = free-stream temperature

Tt_o = free-stream stagnation temperature

M2 = engine Mach number

MO = free-stream Mach number

Therefore,

$$(2.30)\qquad \frac{\dot{Q}}{Du_o} = \pi_d \left(\frac{1+\frac{1}{2}(\gamma-1)M_o^2}{1+\frac{1}{2}(\gamma-1)M_2^2}\right)^{\frac{1}{\gamma-1}} \left(\frac{M_2}{M_o}\right) \left(\frac{1}{1+\frac{1}{2}(\gamma-1)M_2^2}\right)^{1/2} \left(\frac{1+\frac{1}{2}(\gamma-1)M_o^2}{(\gamma-1)M_o^2}\right)$$

But since M2 >> 1, and MO >> 1, this can be approximated as:

$$(2.31)\qquad \frac{\dot{Q}}{DU_o} = \frac{1}{2}\pi_d \frac{1}{\sqrt{\frac{1}{2}(\gamma-1)}} \frac{1}{M_2} \left(\frac{M_o}{M_2}\right)^{\frac{3-\gamma}{\gamma-1}}$$

For the third term, the radiated heat term, dividing by Du_o yields:

$$(2.32)\qquad \frac{\dot{Q}_r}{DU_o} = \frac{\epsilon\sigma T_s^4 S}{DU_o} = \frac{\epsilon\sigma T_s^4}{\frac{1}{2}\rho_o U_o^2 C_f} = \frac{2\epsilon\sigma T_s^4}{\rho_o a_o^3 M_o^3 C_f} = \frac{2\epsilon\sigma T_s^4}{a_o M_o^3 C_f \gamma R T \rho_o} = \frac{2\epsilon\sigma T_s^4}{a_o \gamma p_o M_o^3 C_f}$$

For the fourth term, the fuel cooling term, dividing by Du_o yields:

$$(2.33)\qquad \frac{\dot{Q}_f}{DU_o} = 1.4\times10^7 \left(\frac{T}{D}\right)\left(\frac{1}{u_o g I_{sp}}\right) = 1.4\times10^6 \left(\frac{T}{D}\right)\left(\frac{1}{a_o M_o I_{sp}}\right)$$

Sustituting the following "typical" values for the variables in equations (2.31), (2.32), and (2.33):

π_d = 0.5
γ = 1.4
ϵ = 0.85
M2 = 3
MO = 6
Ts = 1500° K
Cf = 0.002
a_o = 295 m/sec
Isp = 2500 sec
T/D = 1.0, for straight and level flight

yields:

$$\dot{Q}/Du_o = 2.981$$

$$\dot{Q}e/Du_o = 0.25\ \dot{Q}f$$

$$\dot{Q}r/Du_o = 2.735/\ p_o$$

$$\dot{Q}f/Du_o = 0.322$$

Under these particular conditions, then, in order for the aerodynamic heating rate plus the engine heating rate to be less than the sum of the heat radiated away plus the heat absorbed by the hydrogen, p_o must be less than 1.00 N/m^3, corresponding to an altitude higher than 16 km. This is a reasonable result.

The problem becomes much more acute at higher Mach numbers, however. Clearly, the heating due to aerodynamic drag increases as the Mach number increases. Unfortunately, the rate of heat radiated away from the vehicle decreases as the cube of the Mach number as shown by equation 2.32. Furthermore, the excess heat sink available from the fuel decreases dramatically with increasing Mach number as shown by figure 2-5.

This calculation was not intended to demonstrate definitive feasibility or infeasibility, but rather to show that ASP thermal protection is an extremely sensitive problem, with a fine line between the heat load and available heat sink.

2.3 Overall Vehicle Performance: Mass Ratios

Energy Required to Reach LEO--Idealized. Neglecting atmospheric drag, the energy required to raise a kilogram of mass to a height corresponding to that of a low earth orbit and impart to it a circular velocity sufficient so that its centrifugal force will balance the earth's gravitational attraction can be calculated by summing the kinetic and potential energies of the object, namely:

$$E = 1/2\ m\ V_{leo}^2 + \int_{R_o}^{R_{leo}} m\ g\ dR \tag{2.34}$$

When the centifugal force balances the force of gravity,

$$m\ V_{leo}^2 / R_{leo} = m\ g_{leo} \tag{2.35}$$

or,

$$V_{leo} = \sqrt{g_{leo} R_{leo}} \tag{2.36}$$

We still need to know g as a function of height. Neglecting the variation of gravity with geographical latitude, the acceleration of gravity at any distance from the earth's center is given by the following equation:

$$g = g_o (R_o/R)^2 \tag{2.37}$$

where g_o= gravitational acceleration at the earth's surface, R_o = radius of the earth, and h = height of the orbit. Thus, at low earth orbit, (2.38) $$g_{leo} = g_o (R_o / R_{leo})^2 = g_o \, (R_o / R_o + h)^2$$

The velocity of the orbiting body is found using (2.36) and (2.38):

(2.39) $$V_{leo} = R_o \, (g_o / R_o + h)^2$$

Substituting this equation into (2.34) yields the equation:

(2.40) $$E = 1/2 \; m \; R_o^2 \, (g_o / R_o + h) \quad + \int_{R_o}^{R_{leo}} m \; g_o (R_o / R)^2 \, dR$$

At altitudes of 80 km or higher, the effect of air resistance is negligible in equation (2.40) and can be safely ignored. Using $m = 1$ kg, $g_o = 9.81$ m/sec^2, $R_o = 6371$ km, $h = 250$ km, gives $E = 3.2 \times 10^7$ Joules. This is approximately equal to the amount of energy released from burning one gallon of gasoline or 0.26 kg of hydrogen. Of course, this is the theoretical limit for a perfectly efficient system. In any physical system, more energy would be needed to overcome drag and to carry the fuel which will be used along the way.

As an example of a physical system, the space shuttle requires about 67 kg of propellant (56 kg of oxygen and 11 kg of fuel) for each kg placed into LEO, or 40 times the theoretical minimum energy requirement. Certainly, the objective in space transportation is minimization of cost to orbit, not energy to orbit. This calculation is intended to provide an upper bound on the energy efficiency of a space transportation system. Clearly, today's system is far from the theoretical performance limit.

Energy Required to Reach LEO--Actual. While air-breathing engines provide an advantage over rocket engines by reducing the amount of on-board oxygen required, they have a negative side effect. Atmospheric drag affects a horizontal take-off vehicle such as the ASP far more than a vertically launched rocket. It has been long established that "...for rockets of a size sufficient to accomplish the flight [to low earth orbit], the deceleration due to drag is so small that the velocity loss over the interval of transverse of the atmospheric layer can be neglected (Malina and Summerfield, 1947, p. 472)." That the size effect is favorable can be seen from the fact that air resistance is proportional to the projected area of the rocket, whereas mass varies as the cube of the linear dimensions. Thus, the deceleration due to drag is inversely proportional to the linear size of the rocket. In the case of rockets large enough to reach LEO, the neglect of drag is justified and the total velocity change during the burning period is given by the familiar "rocket equation" (Hill and Peterson, 1970, p. 323):

$$Mb/Mo = \exp[-\Delta V/g \cdot Isp] \tag{2.41}$$

where Mb = burn-out mass, or mass at the end of the thrust period, Mo = initial vehicle take-off mass, ΔV = the change in vehicle velocity, and Isp = rocket engine specific impulse.

However, in the case of a vehicle flying horizontally through the atmosphere, the effects of drag cannot be neglected. As will be shown below, drag reduces the vehicle's net acceleration for a given amount of thrust. Thrust is related to specific impulse, the usual measure of performance for an engine or rocket. Specific impulse is defined as the thrust output per unit mass flow of propellant input. From this definition, we have:

$$\text{(2.42)} \qquad Isp = -T/g\cdot(dMprop/dt)$$

where T = thrust and Mprop = propellant mass. Dividing by Mo, rearranging terms and integrating gives:

$$\text{(2.43)} \qquad \int_{M_{initial}=M_o}^{M_{burn\text{-}out}=M_b} (dMprop/Mo) = -\,(T/Mo\cdot g)\int_{t_o}^{t_{burn\text{-}out}} dt/\overline{Isp}$$

where $\overline{Isp}$ = average specific impulse. Recognizing that the differential of the fuel mass is the same as the differential of the vehicle mass since the fuel is the only mass that is changing gives the substitution: dMprop = dMo. Also, Mo·g = W. Thus:

$$\text{(2.44)} \qquad \int_{M_o}^{M_b} (dMo/Mo) = -\,(T/W)\int_{t_o}^{t_b} dt/\overline{Isp}$$

Integrating yields,

$$\text{(2.45)} \qquad \ln\,(Mb/Mo) = -\,(T/W)\cdot(t/\overline{Isp})$$

where t is the flight time of a particular propulsion phase. Assuming a constant acceleration, t can also be expressed as velocity divided by net acceleration, or $\Delta V/\dot{V}$. Thus we have the result:

$$\text{(2.46)} \qquad Mb/Mo = \exp[(T/W)\cdot(-\Delta V/\dot{V}\cdot\overline{Isp})]$$

This is really just the "rocket equation", substituting $(T/W)\cdot\dot{V}$ in place of g, allowing the inclusion of lift and drag effects in the vehicle's net acceleration. Before we can make estimates of the value of Mb/Mo for the different propulsion phases, we need to have values for (T/W), $\dot{V}, \Delta V$, and $\overline{Isp}$ to substitute into the above equation. The following analysis will provide these values.

Net Acceleration. To identify the net acceleration of an ASP in horizontal flight, begin with a free-body diagram, shown in figure 2-6.

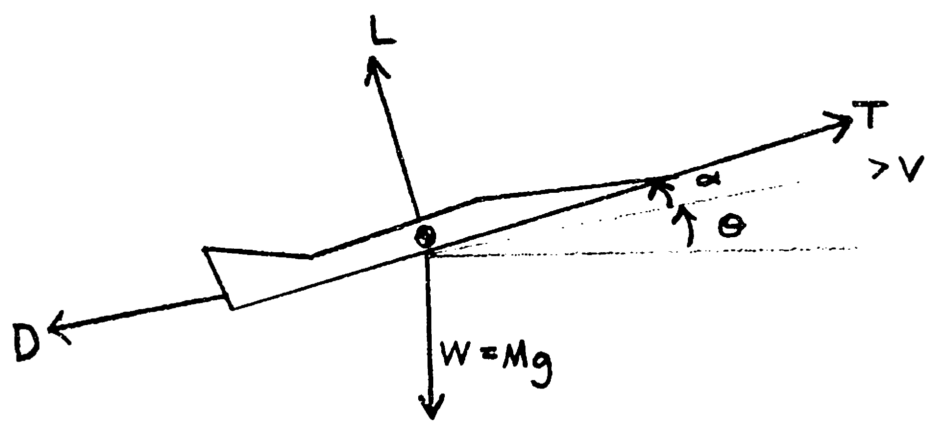


Figure 2-6: Free-Body Diagram of an ASP in Flight

This diagram holds for the first two propulsion phases, the ATR and the scramjet. During the rocket powered phase the ASP would follow a traditional vertical ascent path since the atmospheric oxygen would be gone. Assuming: (1) a constant vehicle acceleration, $\dot{V}$, and (2) a constant climb angle for the first two propulsion phases, θ, the net force on the ASP is given by:

(2.47) $$M\cdot\dot{V} = T\cdot\cos\alpha - D - W\cdot\sin\theta$$

where D = drag, T = thrust, and W = weight. Dividing by M·g and assuming that W = L since Θ is small yields,

$$\dot{V}/g = T/W - D/L - \Theta \tag{2.48}$$

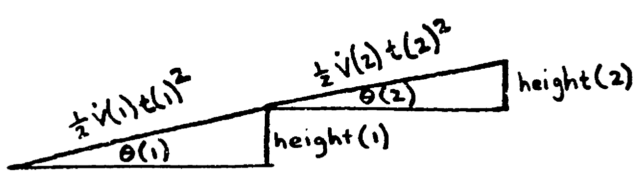
The optimum thrust of an ABES is generally agreed to be 0.5 of the vehicle's weight at take-off based on trade-offs between engine mass and fuel consumed during the vehicle's acceleration phase (see for example Miller, 1985, p. 5 or Cooper, 1985). Thus, T/W will be taken to be 0.5. The lift-to-drag ratio for the ASP will probably be around 5, based on the delta wing configurations shown in preliminary sketches (see for example Reed, 1979, p. 5). The D/L term in equation (2.48) will therefore be assumed to be a constant 0.2. Thus, we now have:

$$\dot{V}/g = 0.3 - \Theta \tag{2.49}$$

The climb angle for the two ABES phases can be determined from geometry (see figure 2-7) using the fact that:

$$\sin\Theta(i) = \text{height}(i) \;/\; 0.5\;\dot{V}(i)\cdot t(i)^2 \tag{2.50}$$

where the subscripts (i) indicate which air-breathing propulsion stage is being considered: (1) ATR, or (2) scramjet.

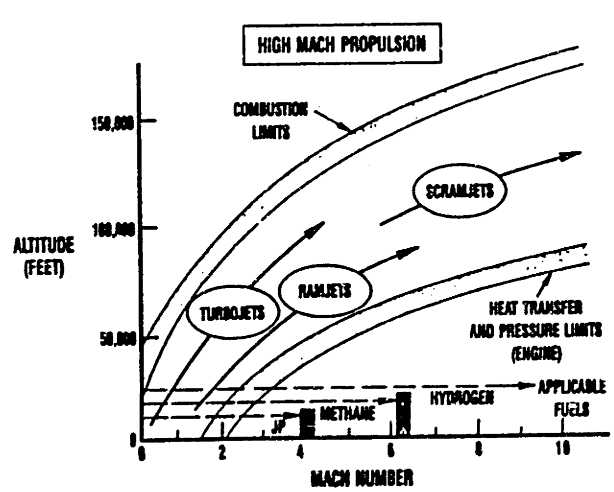


As before, t can be expressed as $\Delta V/\dot{V}$, thus equation (2.50) can be rewritten as:

$$\sin\theta(i) = 2 \cdot height(i) \cdot \dot{V}/(\Delta V)^2 \qquad (2.51)$$

Now we have $\dot{V}$ as a function of θ in equation (2.49), and θ as a function of $\dot{V}$, the maximum height of the ASP for each propulsion phase, and the ΔV. If we could substitute values for the heights and ΔV's, we could solve (2.51) and (2.49) for the climb angles and net accelerations. So far however, we have said nothing about how high the ATRs and the scramjets might operate or what the ΔV requirements for each propulsion phase are. Thus, an examination of these issues is necessary before the analysis can proceed.

Engine Operating Regimes. Air-breathing engine systems, unlike rocket engines, must operate within an altitude/Mach number corridor as shown in figure 2-8 (from Cooper and Escher, 1985, p. 76). This corridor is bounded on the high side by low-pressure combustion limits and inefficient thrust-production. The lower boundary is the result of heat-transfer and vehicle structural limitations.



The ATR engines would be used to take the vehicle from standing on the runway to the point where scramjets becomes more fuel efficient, i.e., have a higher specific impulse. Figures 2-2 and 2-3 indicate that around M = 5.5, the scramjet and the ATR have approximately equivalent performances, around 3000 sec of specific impulse. Mach 5.5, then will be taken as the transition from ATR propulsion to scramjet propulsion in this analysis.

We may now estimate the altitude up to which ATR engines will be used. This operating altitude is based on the maximum allowable external dynamic pressure, qmax, the vehicle can structurally withstand. Using a value of qmax = 1 atmosphere, the dynamic pressure equation qmax = $1/2\ \rho\ V^2$, and the velocity of transition from ATR to scramjet power (Mach 5.5), gives atmospheric density of 0.077 kg/m^3, corresponding to an altitude of about 21 km. Thus, 21 km will be used as the altitude of transition from ATR to scramjet propulsion; in equation (2.51) this will be height(1).

An operating ceiling for scramjets based on engine capture area considerations has yet to be determined. Estimates range from 33 km (Pickney, 1977) to 65 km (Grey, 1986, p. 79). Using the same dynamic pressure limit as above, at Pickney's estimate of 33 km (0.016 kg/m^3), one finds a maximum allowable vehicle velocity at this altitude of about 3560 m/sec, or M = 12. On the other hand, if one assumes that scramjets will function up to an altitude of 45 km or higher, the same dynamic pressure limit will give a maximum allowable vehicle velocity of about Mach 24, or orbital velocity.

Thus, I have chosen an altitude of 49 km as the operating ceiling for scramjets for three reasons: (1) in and of itself, it does not place a constraint on the maximum allowable scramjet operating velocity, a topic that will be addressed later, (2) it is mid-way between the two estimates cited above, and (3) it is not critical to this analysis to have an accurate value for the heights, since they only determine the climb angle which has a very small effect on the net acceleration as shown by equation (2.49). Therefore, height(2) in equation (2.51) will be 49 km - height(1) = 28 km.

Velocity Change Requirements. Overall velocity change requirements, known as Δ V's, can now be calculated. The ATR has to bring the vehicle from zero velocity to Mach 5.5, giving a velocity change of 1620 m/sec. Assuming the scramjets have to increase this velocity to Mach 18 (5330 m/sec), gives a velocity change of 3710 m/sec, from Mach 5.5 to Mach 18. The rocket engines would have to make up the velocity difference between the velocity at which the scramjets shut-off and the velocity required to maintain orbit at 250 km (155 miles). The required orbital velocity at this altitude is given by equation (2.39) and is 7755 m/sec, but propellant allowances must be added to permit on-orbit maneuvering and the de-orbit burn. I have assumed 500 m/sec will be sufficient for this purpose. Therefore, the rocket engines would have to impart a ΔV of (7755 - 5330 + 500) = 2925 m/sec.

To summarize: Assuming scramjets cut-off at Mach 18, the Δ V requirements are:

$$
\begin{aligned}
&\text{ATR} & \Delta V &= 1620 \text{ m/sec}\\
+ &\text{ scramjet} & \Delta V &= 3710 \text{ m/sec}\\
+ &\text{ rocket} & \Delta V &= 2925 \text{ m/sec}
\end{aligned}
$$

$$
\begin{aligned}
= &\text{ orbital velocity} & \Delta V &= 7755 \text{ m/sec}\\
+ &\text{ maneuvering} & \Delta V &= 500 \text{ m/sec}
\end{aligned}
$$

However, since the operating limit of scramjet engines is not known yet with any certainty, it would be misleading to assume any single value. Therefore, I have performed the preceeding ΔV calculations and the following analysis for thirteen different values of the maximum scramjet operating velocity (from Mach 12 to Mach 23). Throughout this thesis, I present the case for Mach 18 as a sample calculation.

We are now able to solve equations (2.49) and (2.51) for $\dot{V}/g$ and . Solving for the two climb angles gives:

(2.52)
$$
\begin{aligned}
\theta(1) &= 0.0406 \text{ rad} = 2.3^\circ\\
\theta(2) &= 0.0176 \text{ rad} = 1.0^\circ
\end{aligned}
$$

Substituting equation (2.51) into equation (2.49) gives the following values for the net accelerations:

(2.53)
$$
\begin{aligned}
\dot{V}(1) &= 2.54 \text{ m/sec}^2\\
\dot{V}(2) &= 2.77 \text{ m/sec}^2
\end{aligned}
$$

Thus, we now have values for the thrust to weight ratio, the velocity change requirements, and the net acceleration for substitution into equation (2.46). Determination of the average scramjet specific impulse is all that remains before we can calculate the ASP mass fractions for the different propulsion phases. Using equation (2.9), the average scramjet specific impulse between Mach 5.5 and different values of the cut-off Mach number can be calculated. The velocity change requirements and average specific impulse for the range of maximum scramjet operating velocities is presented below:

Mach No. at Scramjet Cut-off	ΔVsj (m/sec)	ΔVroc (m/sec)	Isp (sec)
12	1940	4695	2860
13	2235	4400	2766
14	2530	4105	2672
15	2825	3810	2578
16	3120	3515	2484
17	3415	3220	2390
18	3710	2925	2297
19	4005	2630	2203
20	4300	2335	2109
21	4595	2040	2016
22	4890	1745	1922
23	5185	1450	1828

Figure 2-9: Velocity Change Requirements vs. Maximum Scramjet Operating Velocity

ASP Mass Fractions. It is now possible to calculate the fuel requirements of an ASP. Based on figures 2-2 and 2-3, we can approximate the average Isp of the ATR as about 3500 sec over the range M=0 to M=5.5, and the average Isp of the scramjet as about 2297 sec from M=5.5 to M=18. At this point we know the values for the terms in the right hand side of equation (2.46), so the mass ratios and consequently, the fuel ratios, can be calulated:

(2.54) $$Mb(1)/Mo(1) = r1 = \exp[(T/W)\cdot(-\Delta V/\dot{V}\cdot\overline{Isp})]$$

$$= \exp[(0.5)\cdot(-1620/2.54\cdot3500)] = 0.913$$

and

(2.55) $$Mb(2)/Mo(2) = r2 = \exp[(T/W)\cdot(-\Delta V/\dot{V}\cdot\overline{Isp})]$$

$$= \exp[(0.5)\cdot(-3710/2.77\cdot2297)] = 0.747$$

where:

r1 = ASP mass at ATR shut-off/initial ASP mass (GLOW)

r2 = ASP mass at scramjet shut-off/ASP mass at ATR shut-off

r3 = ASP final (on-orbit) mass/ASP mass at scramjet shut-off

The fraction (1-r)/r is then the propellant mass divided by the final mass for each propulsion phase. This will be termed the propellant ratio, R.

(2.56) $$(1-r)/r = R$$

From equations (2.54) and (2.55) then, R1 = 0.0954, and R2 = 0.338. However, these strict calculations unrealistically leave no room for error, no safety margin. I shall assume a fuel reserve factor for all propulsion phases of 2 percent. Thus, the effective propellant ratios are really:

(2.57) $$R1 = 1.02(0.0954) = 0.0973$$

$$R2 = 1.02(0.338) = 0.345$$

Working backwards, using the formula:

(2.58) $r = 1/(R+1)$

we get effective values for r1 and r2. Thus,

(2.59) $r1 = 0.911$

$r2 = 0.743$

For the rocket powered leg of the flight, we use the rocket equation in its standard form:

(2.60) $Mb(3)/Mo(3) = r3 = \exp[-\Delta V/g\cdot Isp]$

where ΔV = 2925 m/sec and Isp = 480 sec for LOX/LH2 engines, using projected year 2000 technology. Thus, Mb(3)/Mo(3) = r3 = 0.537. From equation (2.56), R3 = 0.861. Allowing for a 2 percent fuel reserve margin gives an actual value for R3 = 1.02(0.861) = 0.878, and thus,

(2.61) $Mb(3)/Mo(3) = r3 = 0.532$

We can now get an overall mass ratio for the ASP by multiplying the three individual mass ratios for each propulsion stage since the final mass of the first phase is the intial mass of the second stage and the final mass of the second stage is the initial mass of the third stage. Thus,

(2.62) r_{net} = ASP final mass/ASP initial mass = $r1\cdot r2\cdot r3 = 0.36$

In other words, the mass of the vehicle when it reaches LEO will be about 36 percent of the mass of the vehicle when it took-off from the runway. The other 64 percent of the initial vehicle mass was propellant mass. This is a substantial improvement over the shuttle, which has a total propellant mass equal to 75 percent of its initial take-off weight. Of course, the propellant mass fraction is not as good an indicator of space transportation system efficiency as is payload mass fraction, the payload mass divided by the GLOW, is better.

Clearly, this calculation was dominated by the assumption that scramjets have a maximum operating velocity of Mach 18, and hence, that rocket engines must make up the ΔV deficit of 2925 m/sec. The performance potential of the ASP depends very heavily on the relative proportions of each propulsion phase. In figure 2-10, I have shown the effect of varying the maximum operating velocity of the scramjet engines from Mach 12 to Mach 23. The overall mass ratio increases from 29 percent to a maximum of 40 percent. Since the operating ceiling of scramjet engines is subject to debate, I will examine the effects of different scramjet ceilings on the vehicle mass calculations of the next section.

Mach No. at Scramjet Cut-off	ASP on-orbit mass/take-off mass
12	0.29
13	0.31
14	0.32
15	0.33
16	0.34
17	0.35
18	0.36
19	0.37
20	0.38
21	0.39
22	0.39
23	0.40

Figure 2-10: Overall ASP Mass Ratio vs. Maximum Scramjet Operating Velocity

2.4 ASP Mass Estimation

Importance. Ultimately, the feasibility of the ASP concept depends on the total vehicle mass. Development and fabrication costs scale as the vehicle's dry mass. The desired operational flexibility depends on the ASP's mass being low enough to permit the use of conventional runways. This fact is not lost on the engineers at the Langley Research Center, where most of NASA's ASP work is being conducted. When asked what aspect of the ASP they are working on, engineers involved in the project, regardless of their disciplines, (thermal, structural, avionics, guidance, etc) quip, "I'm a weights-person."

Method. Without a specific vehicle design from which to calculate the masses of the various components, a systems engineer must rely on parametric methods as described in Akin (1981) and Miller and Akin (1980). Using these methods, the desired payload mass is selected and the other components of the launch vehicle--including the fuselage, the wings, the fuel tanks--are then expressed as a linear function of each other. The coefficients used in these linear equations are determined by reviewing previous designs, and adjusted for the incorporation of new technology. The system of linear equations is then solved and the values for the masses of the various components are determined.

Definition of Terms. The following symbols shall represent the masses and mass ratios of the various launch vehicle elements:

Mpl	Payload
Mt	Fuel Tanks

M_f	Fuselage
M_w	Wings, empennage (tail assembly), and landing gear
M_{tps}	Thermal Protection Sytem .
M_{fe}	Fixed Equipment
M_{atr}	ATR Engines
M_{sj}	Scramjet Engines
M_{roc}	Rocket Engines
M_{pr1}	ATR Propellant
M_{pr2}	Scramjet Propellant
M_{pr3}	Rocket Engine Propellant
M_o	initial take-off mass (sum of all component masses)
λ	payload mass/fueled vehicle mass minus payload mass
ϵ	mass of fuel tanks/mass of propellant contained
δ_f	mass of fuselage/total mass contained
δ_w	mass of wings, empennage, landing gear/supported mass
δ_{tps}	mass of thermal protection system/protected mass
δ_{fe}	variable portion of fixed equipment mass/initial mass
δ_{atr}	mass of ATR engines/mass carried
δ_{sj}	mass of scramjet engines/mass carried
δ_{roc}	mass of rocket engines/mass propelled
r_1	mass at ATR shut-off/initial mass (GLOW)
r_2	mass at scramjet shut-off/mass at ATR shut-off
r_3	final (rocket burn-out) mass/mass at scramjet shut-off
R_1	$(1-r_1)/r_1$ = ATR prop. mass/ASP mass at ATR shut-off
R_2	$(1-r_2)/r_2$ = scramjet prop./ASP mass at scramjet shut-off
R_3	$(1-r_3)/r_3$ = rocket eng. prop. mass/ASP final mass

Determination of Mass Element Parameters

Payload Mass: A consensus appears to be forming around 10,000 kg as the reference ASP payload mass (Military Space, 1984). In this analysis therefore, a value of 10,000 kg is used for the payload mass. Using this reference ASP payload mass, it is possible to estimate what the dimensions of the payload bay might be. This calculation will be useful in the ASP applications analysis contained in chapter 4.

The space shuttle has a payload bay 18.3 m long and 4.57 m in diameter, for a total volume of 300 m^3. The maximum payload the shuttle is designed for is 29,500 kg, yielding a packing density of 98 kg/m^3 or 6.12 lbs/ft^3. Applying the shuttle's payload density to the ASP reference payload weight yields payload bay volume requirements of 3268 ft^3 or 92 m^3. Maintaining the same length/diameter ratio as the orbiter (l/d = 4) gives ASP payload bay dimensions of 3.08 m for the diameter and 12.33 m in length.

Fuel Tank Mass: Akin (1981) gives the following non-linear equation for determining fuel tank mass as a function of propellant mass:

$$Mt = 0.2\,(Mpr1 + Mpr2 + Mpr3)^{0.9} \qquad (2.63)$$

Fuselage Mass: As defined above, δf is the ratio of the mass of the fuselage to the mass it contains. In the case of the ASP, this would consist of the propellant fuel tankage, the fixed equipment, the payload, and the engines. The propellant mass is left out of the equation in this model because the structure necessary to carry the fuel is really the fuel tankage, and the fuel tankage is included in the equation, which is

$$(2.64) \quad Mf = \delta f \,(Mpl + Mt + Mfe + Matr + Msj + Mroc)$$

The task then remains to select an appropriate value for δf. Akin (1981) suggests a value of 0.5 based on current technology. John Steiner, chairman of the White House Aeronautical Policy Review Committee, has said that in the future, these structural weights could be reduced by 25 to 35 percent through the incorporation of advanced composite materials (United States Congress, Hearing on The Future of Aeronautics, 1983, p. 31). Thus, a value of 0.40 will be used for δf.

Wing, Empennage, and Landing Gear Mass: Likewise, the mass of the wings, empennage, and landing gear is δw times the mass supported, or

$$(2.65) \quad Mw = \delta w \,(Mpl + Mt + Mf + Mw + Mtps + Mfe + Matr + Msj + Mroc)$$

Here Akin suggests a value of approximately 0.125. Similar weight reductions through advanced composites may be expected to reduce δw to about 0.10, the value assumed in the following analysis.

Thermal Protection System Mass: The thermal protection system mass is given by δtps times the mass protected. Again, the mass of the propellants is left out of this equation because the fuel tankage mass, which contains the propellant, is included. The TPS mass can therefore be written as

$$\text{(2.66)} \quad M_{tps} = \delta_{tps} (M_{pl} + M_t + M_f + M_w + M_{fe} + M_{atr} + M_{sj} + M_{roc})$$

Based on current reusable surface insulation (RSI) technology as used on the shuttle, Akin (1981) cites a value of 0.25 for δ_{tps}. Based on research at NASA-Langley, a 400 percent reduction of the TPS areal density is considered possible, suggesting that tps may be reduced to below 0.10 (Cooper, 1985). For this analysis, I chose a conservative estimate of tps: 0.20.

<u>Fixed Equipment Mass</u>: Miller (1985, p. 5) found, using shuttle orbiter data, that the fixed equipment mass scales linearly with the initial take-off mass of a space vehicle plus a certain fixed amount that is needed regardless of the size of the vehicle. His equation is presented below in general form and as applied to the specific example of an ASP:

$$\text{(2.67)} \quad M_{fe} = \delta_{fe} M_o + 4240 = \delta_{fe} (M_{pl} + M_t + M_f + M_w + M_{tps} + M_{fe} + M_{atr} + M_{sj} + M_{roc} + M_{pr1} + M_{pr2} + M_{pr3}) + 4240$$

where δ_{fe} has been determined to be approximately 0.04. The fixed equipment includes prime power, electrical and hydraulic conversion and distribution, surface controls, avionics, environmental control, furnishings, personnel provision, OMS/RCS fuel, and the mass of two pilots.

ATR Mass: The Aerojet Company provides an estimate of the thrust to engine weight ratio of ATRs as approximately 25 (United States Congress, Hearing on High Speed Aeronautics, 1985, p. 148). Dividing the thrust to weight ratio of the vehicle (assumed to be 0.5) by this value gives the ATR engine weight as a proportion of the total vehicle weight. Thus,

$$\text{(2.68)} \qquad Matr = \delta atr\ Mo = 0.02\ Mo$$

or,

$$\text{(2.69)} \qquad Matr = \delta atr\ (Mpl + Mt + Mf + Mw + Mtps + Mfe + Matr + Msj + Mroc + Mpr1 + Mpr2 + Mpr3)$$

Scramjet Mass: Miller (1985, p. 5) conservatively estimates the weight of a scramjet engine to be 10 percent of the developed thrust. The same reasoning as above gives,

$$\text{(2.70)} \qquad Msj = \delta sj\ (Mo - Mpr1) = 0.05\ (Mo - Mpr1)$$

The ATR propellant mass must be subtracted out of the equation first since this is no longer part of the vehicle's weight at the time of scramjet ignition. In terms of component masses,

$$\text{(2.71)} \qquad Mscram = \delta sj\ (Mpl + Mt + Mf + Mw + Mtps + Mfe + Matr + Msj + Mroc + Mpr2 + Mpr3)$$

Rocket Engine Mass: For LOX/LH2 engines, Miller (1985, p. 5) gives a value of 0.037 for the ratio of the engine weight to the vehicle dry weight. Thus,

$$Mroc = 0.037\,(Mo - Mpr1 - Mpr2 - Mpr3) \tag{2.72}$$

Again, the propellant masses of the two previous propulsion stages is subtracted out of the equation. Letting 0.037 = σroc, the rocket engine mass is σroc times the mass of the vehicle at the time the rocket engines begin operation, or

$$Mroc = \sigma roc\,(Mpl + Mt + Mf + Mw + Mtps + Mfe + Matr + Msj + Mroc) \tag{2.73}$$

Propellant Masses: The propellant mass required for each propulsion phase is a function of the burn-out mass of the vehicle at the end of that propulsion phase and R, which is a function of the vehicle thrust-to-weight ratio, the ΔV required, the net acceleration, and the engine's specific impulse. The propellant mass equations are then:

$$Mpr1 = R1\cdot(Mpl + Mt + Mf + Mw + Mtps + Mfe + Matr + Msj + Mroc + Mpr2 + Mpr3) \tag{2.74}$$

$$Mpr2 = R2\cdot(Mpl + Mt + Mf + Mw + Mtps + Mfe + Matr + Msj + Mroc + Mpr3) \tag{2.75}$$

$$Mpr3 = R3\cdot(Mpl + Mt + Mf + Mw + Mtps + Mfe + Matr + Msj + Mroc) \tag{2.76}$$

From the above relations we see that only the tank mass scales non-linearly. The other nine component masses, fuselage, three propellant masses, wings, three engines, and thermal protection system, can be expressed as a system of linear equations.

In one very important manner, these equations represent conservative estimates, that is, estimates that err on the side of excess weight. The way in which this comes about is that there is a difference between treating all three individual propulsion systems separately, as I have done here, and treating the engines as a single combined-cycle engine. A combined-cycle engine is one which provides synergistically integrated, multi-mode propulsion. Well-known examples are the ATR, the variable-cycle turbofan/jet, and the dual-mode ramjet, which is capable of either subsonic or supersonic combustion operation. The ASP program is searching for innovative ways to combine the operation of the ATR, scramjets, and rocket engines into a single, closely-coupled engine. It is likely that the weight of the integrated engine will be less than the sum of the weights of the individual engines; however, the extent of this weight reduction is difficult to predict, so I have ignored it. Thus, by taking engine mass fractions as if they were separate engines, conservatism is built into the vehicle mass estimates.

Solution Algorithm. This system of equations is presented in figure 2-11, showing that solving this system requires finding x where Ax = b, and A is a ten by ten matrix. The problem is solved using a program adapted from Akin (1981). A listing of the program is contained in Appendix A. The solution algorithm is:

$$
\begin{bmatrix}
1 & 0 & 0 & -\delta_f & -\delta_f & -\delta_f & -\delta_f & 0 & 0 & 0 \\
-\delta_w & 1-\delta_w & -\delta_w & -\delta_w & -\delta_w & -\delta_w & -\delta_w & 0 & 0 & 0 \\
-\delta_{tps} & -\delta_{tps} & 1 & -\delta_{tps} & -\delta_{tps} & -\delta_{tps} & -\delta_{tps} & 0 & 0 & 0 \\
-\delta_{fe} & -\delta_{fe} & -\delta_{fe} & 1-\delta_{fe} & -\delta_{fe} & -\delta_{fe} & -\delta_{fe} & -\delta_{fe} & -\delta_{fe} & -\delta_{fe} \\
-\delta_{atr} & -\delta_{atr} & -\delta_{atr} & -\delta_{atr} & 1-\delta_{atr} & -\delta_{atr} & -\delta_{atr} & -\delta_{atr} & -\delta_{atr} & -\delta_{atr} \\
-\delta_{sj} & -\delta_{sj} & -\delta_{sj} & -\delta_{sj} & -\delta_{sj} & 1-\delta_{sj} & -\delta_{sj} & 0 & -\delta_{sj} & -\delta_{sj} \\
-\delta_r & -\delta_r & -\delta_r & -\delta_r & -\delta_r & -\delta_r & 1-\delta_r & 0 & 0 & 0 \\
-R1 & -R1 & -R1 & -R1 & -R1 & -R1 & -R1 & 1 & -R1 & -R1 \\
-R2 & -R2 & -R2 & -R2 & -R2 & -R2 & -R2 & 0 & 1 & -R2 \\
-R3 & -R3 & -R3 & -R3 & -R3 & -R3 & -R3 & 0 & 0 & 1
\end{bmatrix}
\begin{bmatrix}
M_f \\ M_w \\ M_{tps} \\ M_{fe} \\ M_{atr} \\ M_{sj} \\ M_r \\ M_{prop1} \\ M_{prop2} \\ M_{prop3}
\end{bmatrix}
=
\begin{bmatrix}
\delta_f(M_t + M_{pl}) \\
\delta_w(M_t + M_{pl}) \\
\delta_{tps}(M_t + M_{pl}) \\
\delta_{fe}(M_t + M_{pl}) + 4240 \\
\delta_{atr}(M_t + M_{pl}) \\
\delta_{sj}(M_t + M_{pl}) \\
\delta_r(M_t + M_{pl}) \\
R1(M_t + M_{pl}) \\
R2(M_t + M_{pl}) \\
R3(M_t + M_{pl})
\end{bmatrix}
$$

Figure 2-11: Matrix of Linear Equations Representing ASP Component Masses

1) Select values for the parametric coefficients, the payload mass, the delta-v's, and the Isp's.

2) Set the initial estimate of the propellant tank mass equal to zero.

3) Solve the set of linear equations for the ten masses in the column vector x.

4) Using the derived propellant tank mass, update the initial estimate.

5) Repeat steps 3 and 4 until the values of tank mass converge.

Results. The output of the mass properties computer program are presented in Appendix B. Figure 2-12 shows the effect an increased scramjet operating limit has on total vehicle weight. The ASP GLOW drops an order of magnitude from a maximum of 3.25 million kg at M = 12 to just over 300,000 kg at M = 21. Compared with the shuttle, at M = 12 the ASP GLOW is 50 percent greater than the shuttle's; at M = 21 the ASP GLOW is about one-sixth that of the shuttle. The sharp decline in vehicle GLOW is due to the extension of the scramjet propulsion phase over a wider range, meaning that less fuel is required. A velocity change achieved with scramjet engines requires only LH2; the same velocity change using rocket engines requires both LOX and LH2. Figure 2-13 expresses the information as a function of lambda, the payload mass fraction. Not until scramjets operate up to Mach 16 does the ASP payload mass fraction exceed the shuttle's.

Sensitivity Analysis. The preceeding mass properties calculations were based on assumptions of certain technology levels, reflected in the choice of mass fraction deltas. Since these values are uncertain, it is useful to examine how the vehicle mass is af cted by small changes in these values. The baseline design used was   ase in which scramjets

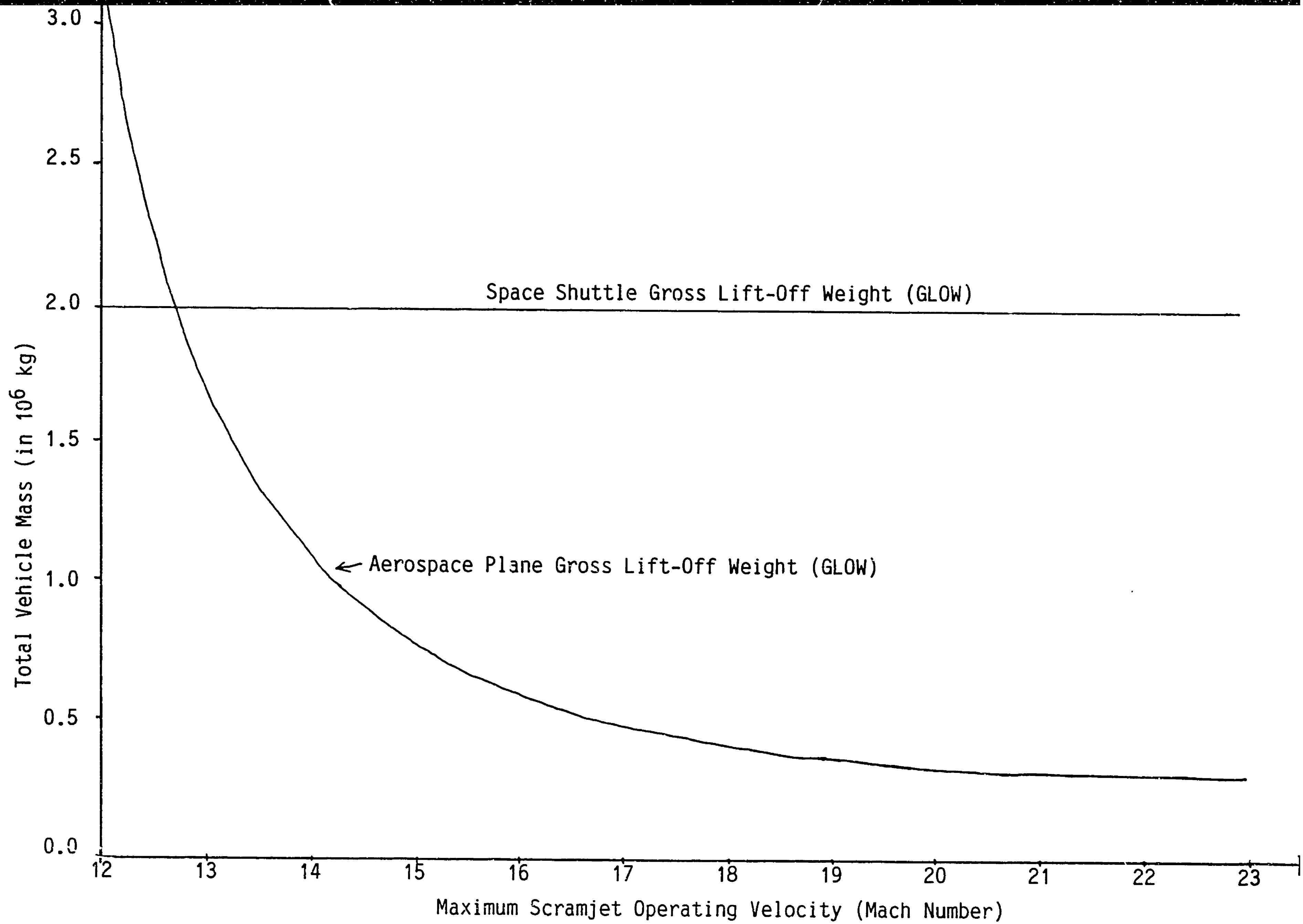


Figure 2-12: ASP Gross Lift-Off Weight vs. Maximum Scramjet Operating Velocity

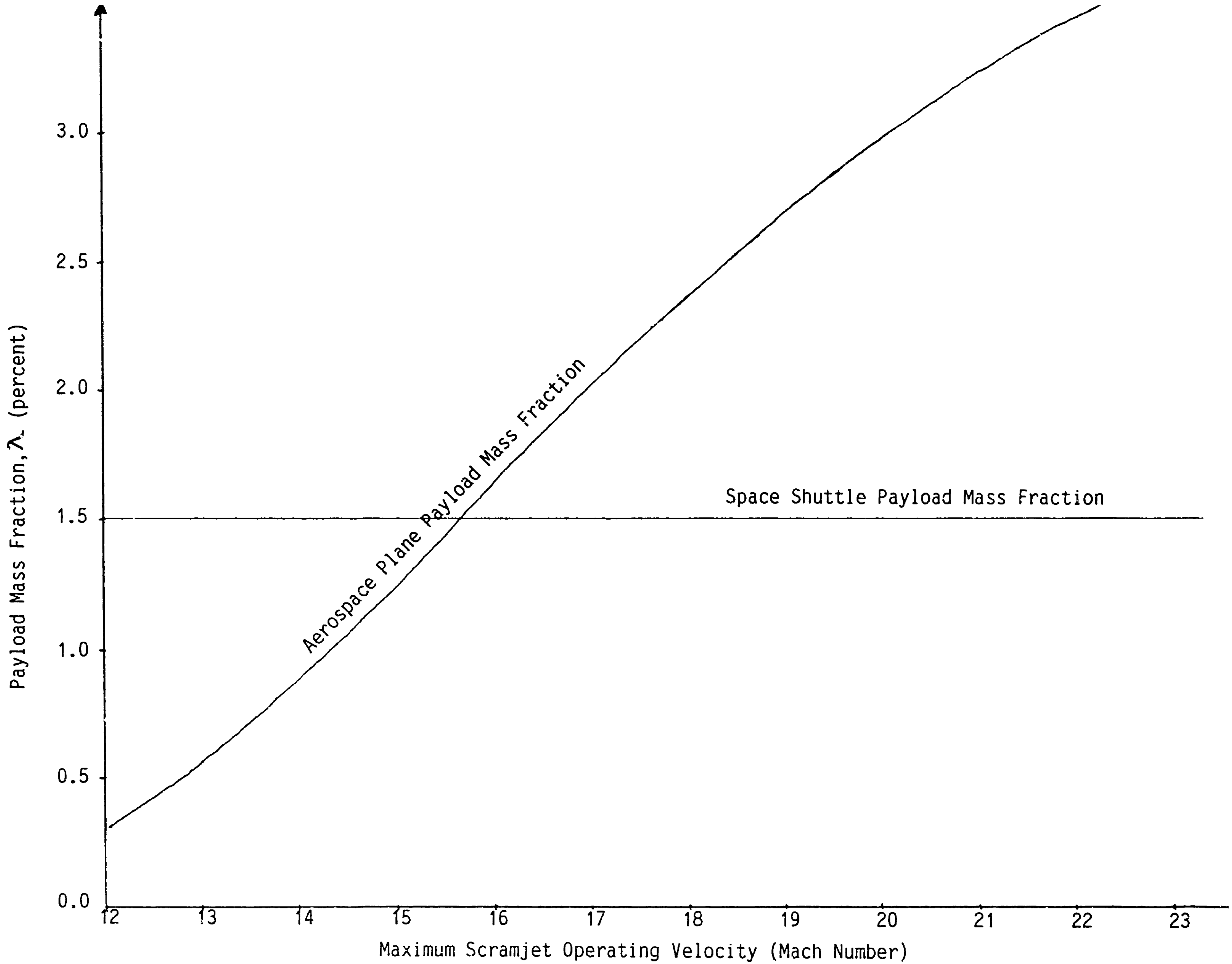


Figure 2-13: ASP Payload Mass Fraction vs. Maximum Scramjet Operating Velocity

operate up to Mach 17. Each mass fraction ratio was varied individually, holding the others constant, and the component masses were calculated.

Figure 2-14 presents the results of this sensitivity analysis. The vertical axis lists the mass fractions and the horizontal axis lists the components. The numbers in the boxes represent the percent increase in the mass of that component or subsystem resulting from a one percent increase in the mass ratio listed in the left-most column. For example, increasing δf by one percent (from 0.4 to 0.404) results in an increase in the total vehicle mass of 1.65 percent (from 500,315 kg to 508,570 kg). The mass fraction ratios are listed in order of decreasing effect on total vehicle mass. This table suggests that the parameters which affect the vehicle's total mass the most are: the rocket engine propellant mass ratio, the fuselage mass ratio, the scramjet engine mass ratio, and the scramjet propellant mass ratio.

Conclusions. These graphs illustrate three main points:

(1) If scramjets prove workable up to Mach 17 or beyond, the ASP may well live up to its supporters expectations for a launch vehicle the size and weight of a conventional aircraft.

(2) If scramjets fall short of these performance goals, perhaps because of unexpected propulsive inefficiencies as described in section 2.1, then the vehicle weight rises exponentially, making the concept untenable. In other words, scramjets are a very unforgiving technology. In contrast with rocket engines, which fail "gracefully," scramjet

	Fuselage	Wing	TPS	Fixed Equipment	Tank	Total Engine	ATR	Scramjets	Rocket Engines	Total Propellant	ATR Propellant	Scramjet Propellant	Rocket Propellant	TOTAL VEHICLE	Payload Mass Fraction
R3	2.49	2.49	2.49	2.47	2.94	2.91	3.00	3.00	2.48	3.28	3.00	3.00	3.51	3.00	-2.91
δ_f	2.38	1.65	1.65	1.36	1.48	1.65	1.65	1.65	1.65	1.65	1.65	1.65	1.65	1.65	-1.63
δ_{sj}	1.44	1.44	1.44	1.19	1.29	2.03	1.44	2.45	1.44	1.44	1.44	1.44	1.44	1.44	-1.42
R2	1.14	1.14	1.07	1.13	1.34	1.33	1.37	1.39	1.13	1.49	1.37	2.14	1.13	1.37	-1.35
δ_{fe}	1.26	1.26	1.26	1.88	1.13	1.26	1.26	1.26	1.26	1.26	1.26	1.26	1.26	1.26	-1.24
δ_{tps}	0.90	1.10	1.93	0.90	0.98	1.09	1.09	1.09	1.09	1.09	1.09	1.09	1.09	1.09	-1.08
δ_w	0.65	1.80	0.79	0.65	0.71	0.78	0.78	0.79	0.79	0.78	0.79	0.79	0.78	0.78	-0.78
δ_{atr}	0.63	0.63	0.63	0.52	0.56	0.88	1.63	0.63	0.62	0.63	0.63	0.63	0.63	0.63	-0.62
δ_{roc}	0.41	0.41	0.41	0.33	0.36	0.57	0.40	0.40	1.40	0.41	0.41	0.40	0.41	0.41	-0.40
R1	0.31	0.31	0.31	0.33	0.40	0.33	0.39	0.31	0.31	0.44	1.31	0.31	0.31	0.39	-0.39

A one percent increase in the mass fraction delta will result in the number in the box percent increase in the component mass. E.g., a one percent increase in δf results in a 1.65% increase in the wing mass.

Figure 2-14: Sensitivity Analysis of ASP Mass Estimating Algorithm performance falls off precipitously. The overall technological riskiness of the ASP is primarily a result of this, the extreme sensitivity of scramjets to engine inefficiencies.

(3) Given (1) and (2), scramjets are the most crucial "make-it-or-break-it" technology of the ASP program. ASP research must concentrate on scramjets in order to remove this large uncertainty about the ultimate feasiblity of the ASP.

CHAPTER 3: AEROSPACE PLANE COSTS

"If there was an OMB in Columbus' time, we'd all still be Europeans."
- Dr. George Low, former Associate Administrator of NASA

3.1 Costing Space Transportation Systems

The ASP may be worth developing for reasons other than strictly cost-effectiveness. Nevertheless, since the major selling point of the ASP is its potential to reduce costs to low-earth orbit, cost should be an important measure of merit.

The total cost of a launch vehicle system has two components: the non-recurring costs, such as R&D, tooling, and vehicle construction; and recurring costs, the costs associated with each launch. Two approaches are available for reducing total costs. An expendable launch vehicle (ELV), being less sophisticated and less risky, minimizes development costs. Operations costs are higher however, because the hardware is discarded. A reusable vehicle has a higher development cost, but reduces operations costs since it ideally only expends propellant with each launch. Consequently, the greater the number of projected flights, the more economic considerations favor reusable vehicles as opposed to expendable ones. As shown in chapter one, the space shuttle fell in between: neither a bargain in terms of development costs, nor a cheap way to get to orbit.

3.2 Methods

Experience has shown that the preliminary cost estimate for a new space system is, at best, an order of magnitude figure. In most instances the cost is not known with any precision until after the project is completed. Nevertheless, preliminary cost estimates can be useful as long as their uncertainty is recognized.

The most accurate cost estimating method would be to actually design an ASP and sum the costs of its components. However, design of an ASP is beyond the scope of this thesis, given the highly proprietary nature of scramjet technology and the security classification which clouds ASP R&D. In lieu of this, one cost estimating strategy is to use empirical analysis, establishing statistical relationships between cost and relevant aggregate system parameters, and extrapolating these costs from previous vehicles of the same technical class to the proposed vehicle. Another approach would be to review cost estimates made by insiders. These experts include ASP program managers, aerospace analysts at major financial institutions, and aerospace industry personnel. Both strategies have flaws.

The problem with the use of cost-estimating relationships (CERs), is that much of the previous statistical data gives cost as a function of GLOW or dry weight. Vehicle gross lift-off weight and dry weight do have strong correlations with total system cost, but they are not the only factors. The ASP has an unusally low GLOW and dry weight relative to a rocket vehicle, but requires the development of additional propulsion system(s), the combined-cycle scramjet engine. The costs associated with propulsion system development might offset the advantages resulting from lower weight. Also, there is no historical cost data specifically for a) a single-stage-to-orbit vehicle, b) a scramjet-powered vehicle, c) a fully reusable vehicle, or d) a horizontal take-off and landing space vehicle from which to extrapolate data.

Two dangers are associated with the second method, soliciting expert opinion. One is that since we do not know the thought process that led one expert to a particular conclusion, assessing the validity of a particular estimate is difficult. The second problem relates to the axiom, "never ask a barber if you need a haircut." The person may have an incentive for providing a biased estimate. Still, these are the only cost estimating strategies available for a first-cut. Both methods are presented here, as a check on each other. First are cost estimates made by various organizations. Next is a modified version of the Transcost Model, which uses statistically derived cost estimating relationships where appropriate, and estimates of ASP-specific components where historical data is lacking.

3.3 Air Force Estimates

DARPA issued a request for proposals (RFP) last fall to cover engines, airframe, and test systems. Multiple contracts worth a total of $300 to $400 million were awarded in early April 1986 for development of a full-scale engine demonstrator and airframe work. Each winning contractor is expected to submit components and systems for ground testing in 1988-89 as technology demonstrators. The USAF estimates the cost of Phase II to be $600 to $700 million (North, 1986). If the aircraft and engine technology prove successful, Phase III will involve the building of test vehicles, now designated the X-30A. The USAF estimates that it will take two and a half vehicles to complete the ground and flight testing of the ASP in the 1990s. There would be at least one flyable prototype, and the other vehicle and aircraft sections would be used for ground testing of the systems and the durability of the aircraft and engines.

The first flight test is envisioned for the mid-1990s. The Air Force is considering wind tunnel testing of the aircraft only up to Mach 8, the capability of existing facilities, and then flying the prototype in gradual stages to its expected Mach 25 capability. The USAF has cited cost estimates of $3 billion to bring the ASP to the flight test stage (Hotz, 1985). Official estimates of the program's cost, if taken beyond the flight test stage, are not readily available. Thus, the most we can say about official estimates is that they predict development costs will be at least $3 billion.

There is disagreement within the USAF about the extent to which the ASP might reduce operational costs. USAF Undersecretary Edward Aldridge enthusiatically predicts a factor of 100 reduction (Military Space, 1985a). Gen. Lawrence Skance has said that his Air Force Systems Command is projecting launch costs will be cut by only a factor of 10-20 (Military Space, 1985b). This uncertainty is understandable. With time, the ASP technologies will be understood in greater detail and more accurate cost estimates made. In this vein, Scott Pace has written:

> It is important to remember how difficult it is to predict the detailed nature of systems to fulfill requirements that had never existed in the past. Particularly in spacecraft development the approach itself is invented, not just the techniques and equipment needed to carry out the mission. Even after this there is the need to develop new manufacturing methods and materials. As each program activity passes through the various steps from definition to operations, the ability of managers to understand the many ramifications of the project increases. (Pace, 1982, p. 207)

3.4 Other Estimates

The British too, are working on an "aerospace plane." Theirs is called HOTOL, short for horizontal take-off and landing and is projected to carry 7,000 kg into LEO. British Aerospace estimates the development costs for the unmanned HOTOL to be between $5 and $6 billion (Conchie, 1985). It is not clear whether this estimate is for the vehicle up to the flight test stage, or whether it encompasses the entire development phase. Either way, it is significant to note that their estimate of the costs is twice as high as the USAF estimate, even though it is a smaller, unmanned version with, presumably, less complexity.

Christopher C. Demisch, an aerospace analyst at the First Boston Corporation, has estimated the financial requirements for the ASP program to be roughly the same as the shuttle's, i.e. in the vicinity of $14 billion for development (Kristof, 1985).

3.5 Transcost Estimate

My own estimate of the ASP program's costs will be calculated using the Transcost Model of D.E. Koelle (Koelle, 1983). This model provides a method for estimating the costs of future space transportation systems and has proved to be within a range of 15 percent compared to the real costs of past projects. The model incorporates both basic cost/mass relationships derived statistically, and correction factors, f1, f2, f3, and f4, reflecting technical logic and judgement regarding non-quantifiable differences between programs. Figure 3-1 on the following page shows a schematic of the derivation of the CERs.

The cost model is divided into development costs; production costs; and flight operations costs (see figure 3-2). The costs are presented in man-years (MY), and thus are not dependent on inflation rates. For conversion to the actual value of the effort, a cost in man-years may be converted to a dollars cost using the appropriate conversion factor, which is strongly influenced by, but not identical to, the inflation rate. Figure 3-3 (from Koelle, 1983, p. 5) presents this conversion scale.

The latest year for which MY conversion factors are available is 1983. To get \$ 1985 from \$ 1983, this factor of \$117,000 was multiplied by the GNP deflator from 1983 to 1985 (1.092), giving 1 MY = \$128,000 (\$ 1985). All cost cited are in \$ 1985. These costs are gross values, including costs of items like travel, and computer costs. Overall costs for each of the hypothetical scramjet operating regimes from Mach 12 to

FIG. 1 – 2

COST MODEL DERIVATION

Logic and sequence of cost model establishment from reference data

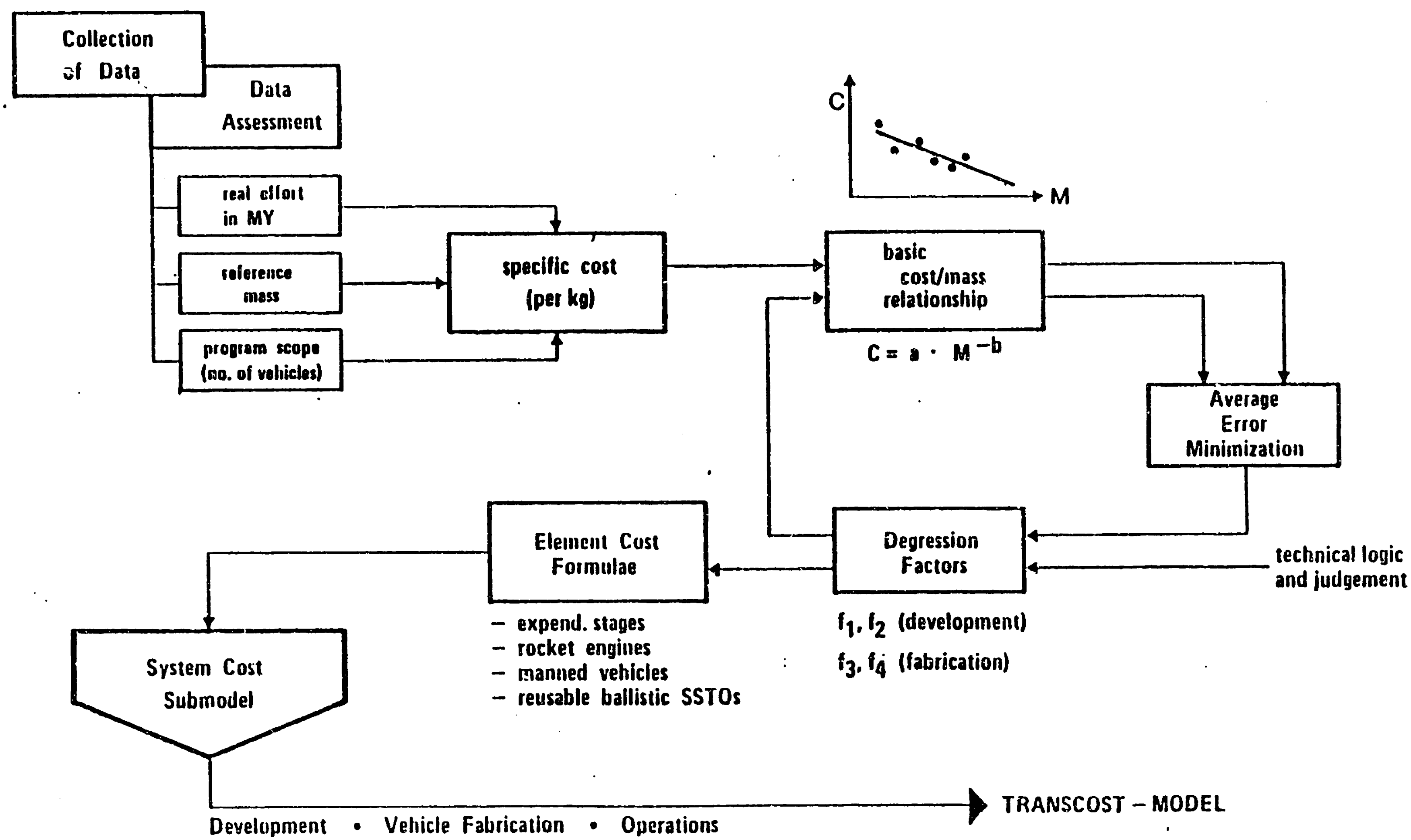


Figure 3-1: Derivation of Transcost Model

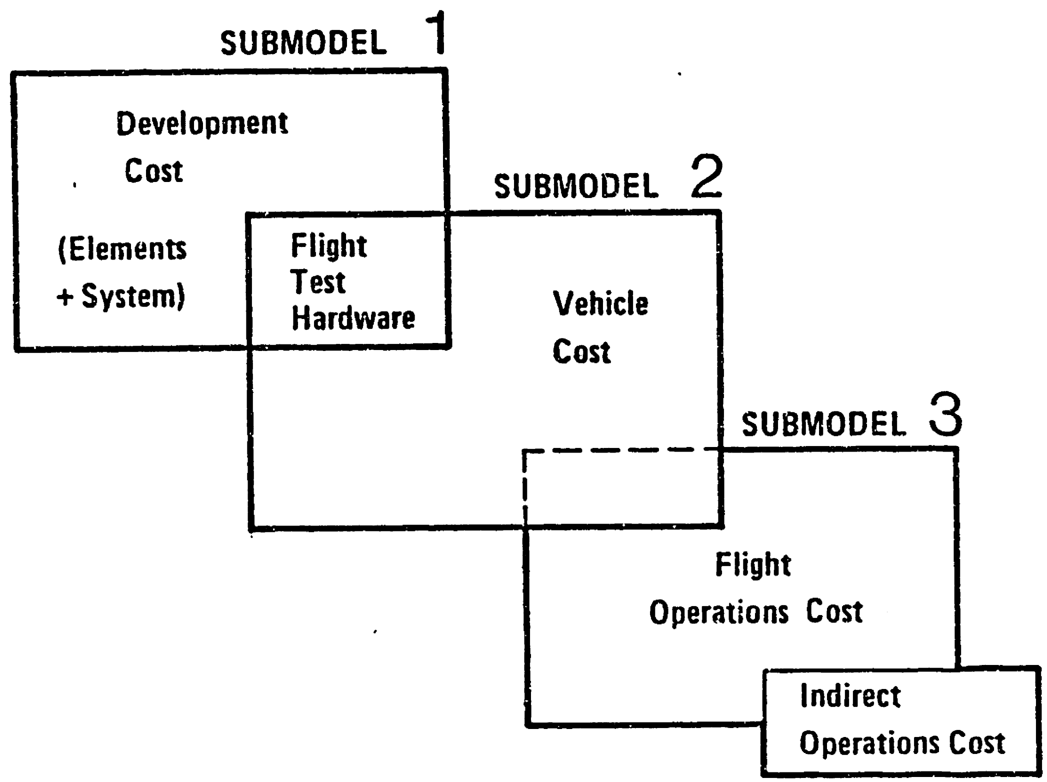


Figure 3-2: Organization of the TRANSCOST-Model in three major Submodels

Mach 23 were calculated and are presented at the end of this chapter. As a sample calculation, costs for a scramjet limit of Mach 18 are presented after each cost equation. Also, where appropriate, comparisons to existing programs are made.

Year	U.S. Aerospace Industry
1970	$ 38,000
1971	$ 40,000
1972	$ 44,000
1973	$ 50,000
1974	$ 55,000
1975	$ 60,000
1976	$ 66,000
1977	$ 72,000
1978	$ 79,500
1979	$ 85,000
1980	$ 92,500
1981	$103,000
1982	$112,000
1983	$117,000

Figure 3-3: Cost of 1 Man-Year (MY) for US Aerospace Industry

Since the inputs to this cost estimating model will be the mass estimates derived in chapter 2, the uncertainty of the cost estimates is compounded by both errors in the vehicle mass projection and errors produced by the cost model. Therefore, the uncertainty of the following cost estimates is likely to be large, as evidenced by the extreme sensitivity of the vehicle mass analysis to different mass fraction values shown in section 2.7. As research proceeds, and the technology becomes better understood, these cost projections will become obsolete rapidly. Nevertheless, the method used to derive them should retain its usefulness since it can be easily modified to incorporate new estimates of key parameters such as specific impulse, engine weight fraction, number of launches per year, ASP fleet size, etc.

DEVELOPMENT COSTS

General Equation. The development cost is modelled as the sum of the individual cost elements plus 10 percent for systems engineering and testing. The essential equations are:

(3.1) $Cdev = 1.1\,(Haf + Hatr + Hsj + Hr)$

where
- Cdev = development cost of the ASP
- Haf = development effort for the airframe
- Hatr = development effort for the ATRs
- Hsj = development effort for the scramjets
- Hr = development effort for the rocket engines

and

(3.2) $H = a \cdot M^{x} \cdot f1 \cdot f2 \cdot f3$ [MY]

where
- H = development effort in man-years
- a = specific value for each type of system
- x = specific value for each type of system
- M = the reference system mass in kg

f1 = correction factor for the technical development standard, i.e.

- first generation system f1 = 1.25
- technology already proven by similar systems f1 = 0.8 - 1.0
- existing system (only minor modifications or size change required) f1 = 0.4 - 0.8

f2 = technical quality factor, different for each system

f3 = team experience factor, i.e.

- new team/company $f3 = 1.1 - 1.3$
- company/industry with some related experience

  $f3 = 0.9 - 1.1$
- company/industry with previous relevant experience

  $f3 = 0.6 - 0.9$

Airframe Development Costs. Figure 3-4 shows the Transcost development cost regression lines for expendable vehicles, reusables, and manned winged airframes. In order to increase the number of sample points, the model also incorporates estimated development costs for three launch systems which were extensively analyzed on paper: the Rockwell Flyback Booster and two Boeing Heavy Lift Launch Vehicles (the white dots). Note that the equations in figure 3-4 are specific cost, that is, cost per unit mass. To get actual cost, the equations in figure 3-4 must be multiplied by M to the first power, yielding an exponent of 0.21 in each case. Thus, the regression indicates that development cost scales less than linearly with the airframe dry weight, a result which seems intuitively logical.

Therefore, for a winged, manned vehicle such as the ASP: a = 6500, and x = 0.21. Since the ASP would clearly be a first generation system, f1 must be taken as 1.25. The technical quality factor was taken to be 1.2, reflecting the incorporation of advanced technology into the vehicle structure. This is to correct for the problem that a more light-weight advanced structure would theoretically reduce the development cost. In reality, the application of advanced materials technology represents a cost driver. The team experience factor is a little harder to judge, but

SPECIFIC DEVELOPMENT COST

FIG. 2–2

of different types of space transportation systems (stages)

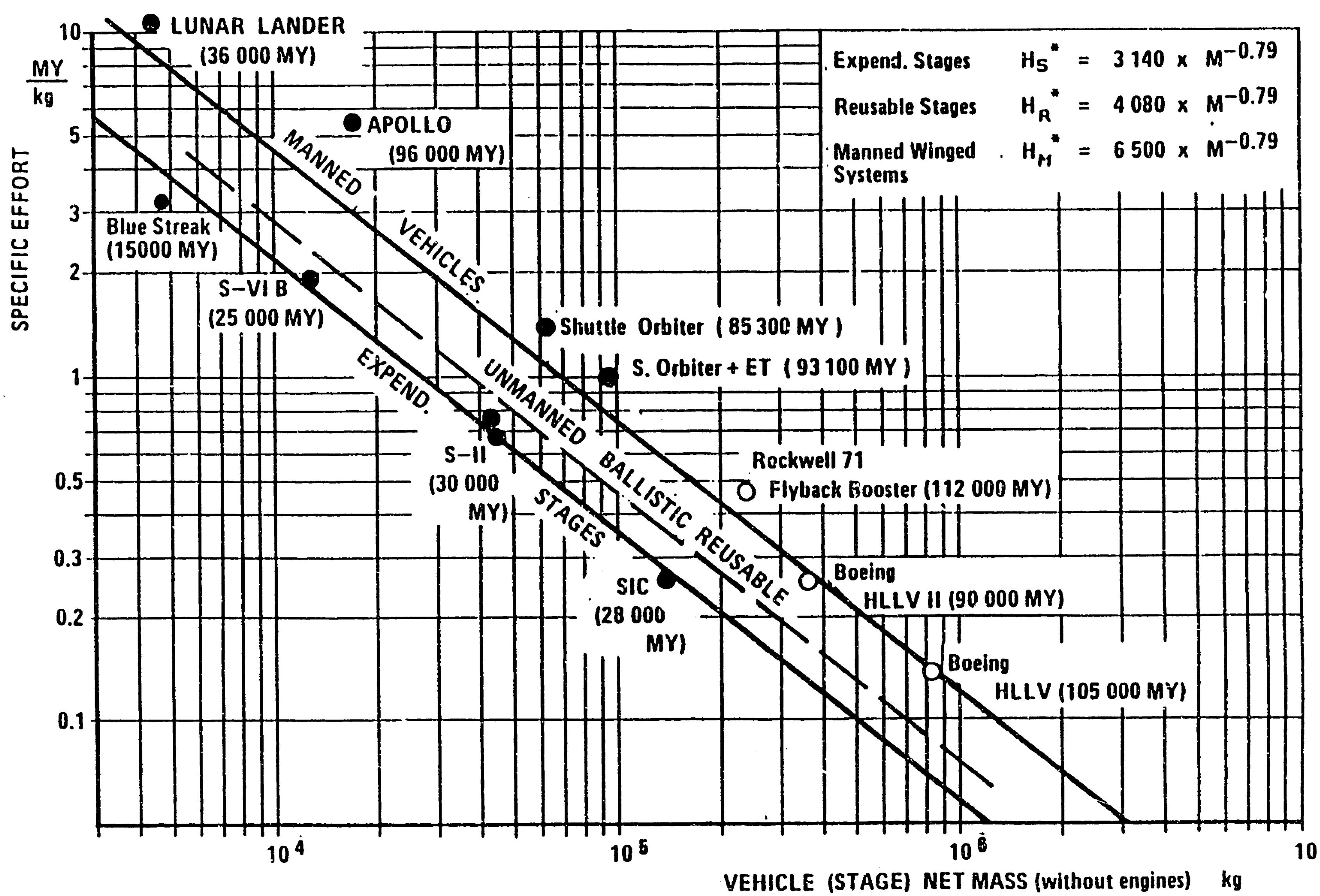


Figure 3-4: Transcost Development Cost Equations for Various Space Transportation Systems

was estimated to be about 0.9 based on the aerospace industry's previous experience with the space shuttle and the SR-71. This gives:

$$(3.3) \qquad Haf = 8775\ Maf^{0.21}\ [MY]$$

$$Haf = 8775\ (Mt + Mf + Mw + Mtps + Mfe)^{0.21}\ [MY]$$

for a Mach number of 18,

$$Haf = 100{,}781\ MY = \$12.9\ \text{billion}$$

<u>ATR Development Costs</u>. As noted in section 2.3, the air turbo-ramjet is a relatively mature technology, though unproven on the scale needed for ASP applications. ATRs have more development risk than the rocket engine stage, but far less than the scramjet engines. Therefore, as a rough estimate of the remaining development costs for the ATR, I shall postulate that they are approximately the arithmetic average of the development costs for the scramjet and the rocket engines.

$$(3.4) \qquad Hatr = 0.5\ (Hsj + Hr)$$

<u>Scramjet Development Costs</u>. At NASA Headquarters, Gregory et al (1971) performed a detailed analysis of scramjet development costs, and arrived at a figure of $488 million ($ 1969). To convert $ 1969 to $ 1985 using the the Federal GNP price deflators, multiply $488 million by 2.76, yielding about $1.3 billion ($ 1985). Figure 3-5 presents Gregory's scramjet development cost estimate, as well as estimates for other vehicle components. Since 1969, the amount of money spent on scramjet research has not been more than a few hundred million dollars, at most. As a result, scramjet technology has progressed steadily, but certainly

Table 2 Vehicle cost statements, millions of dollars

	Air-breather	HTO rocket	VTO rocket	Orbiter
RDT&E				
Airframe design				
Concept formulation	12	12	12	12
Contract definition	70	35	35	35
Design engineering	1391	1659	1481	947
	1473	1706	1528	994
Avionics development	20	20	20	100
Propulsion development				
Rocket	0	373	373	362
Turbojet	694	90	90	0
Ramjet/scramjet	480	0	0	0
	1174	463	463	362
Development support				
1st unit manufacturing				
Airframe	(161)	(150)	(144)	(57)
Avionics	(3)	(3)	(3)	(6)
Rocket	(0)	(35)	(35)	(5)
Turbojet	(25)	(9)	(9)	(0)
Ramjet/scramjet	(8)	(0)	(0)	(0)
	(197)	(197)	(191)	(68)
Flt./grnd. test veh. (3/1)	654	643	623	233
Flight/ground test spares	116	115	112	41
Flight test operations	497	263	274	88
Maint./operat. trainers	46	46	44	81
AGE/tooling	180	200	184	110
Documentation	10	10	10	3
	1503	1277	1247	556
Total RDT&E	4170	3466	3258	2012
Initial acquisition				
Operational vehicles (6)				
Airframe	590	544	523	236
Avionics	13	13	13	25
Propulsion	137	183	183	19
	740	740	719	280
Spares	191	206	202	36
Facilities	100	100	100	100
Sustaining engineering	203	241	215	141
AGE	111	111	108	42
Miscellaneous	34	35	34	89
Total acquisition	1379	1433	1378	688
Recurring operations				
Base support and crew	487	594	583	360
Vehicle maintenance	1071	1163	1115	522
Propellants	138	144	142	24
Facilities/AGE maint.	73	73	72	53
Miscellaneous	44	45	45	23
Total operations (10 yr)	1813	2019	1957	982
Total RDT&E	4170	3466	3258	2012
Total systems cost	7362	6918	6593	3682

Figure 3-5: NASA Estimate of Scramjet Development Costs (in 1969 dollars)

has not made any great "technological leaps forward." Therefore, we can assume that Gregory's estimate is still basically valid. Discounting Gregory's current-day dollar estimate of \$1.3 billion by a few hundred million dollars to take account of progress made between 1969 and 1986 yields an approximate figure of:

$$\text{(3.5)} \qquad Hsj = 7813 \text{ MY} = \$1.0 \text{ billion}$$

Rocket Engine Development Costs. It has been suggested that the ASP rocket engine might be more a derivative of the 150 kg Pratt & Whitney RL-10 engine than the 2880 kg SSMEs (Cooper, 1985). The RL-10 was built in the 1960s and is used to power the Centaur upper stage. In any case, the rocket engines will be based on existing technology and will have the lowest development costs of any ASP engine system. In order to estimate the development costs for the rocket engine, we use the Transcost CER for turbopump engines, shown in figure 3-6:

$$\text{(3.6)} \qquad Hr = 137 \cdot Mroc^{0.64} \cdot f1 \cdot f2 \cdot f3 \quad [\text{MY}]$$

The asterisks in figure 3-6 represent the actual development costs, while the black dots show where the engines lie after the three technology correction factors are incorporated. Thus, it was necessary to estimate values for these correction factors for the ASP rocket engine. The factor which accounts for the novelty of the technology, f1, was taken to be 0.9, since the rocket technology has already been proven by similar systems. The technical quality factor, f2, was taken to be 0.9, the same qualification and reliability standard that was used

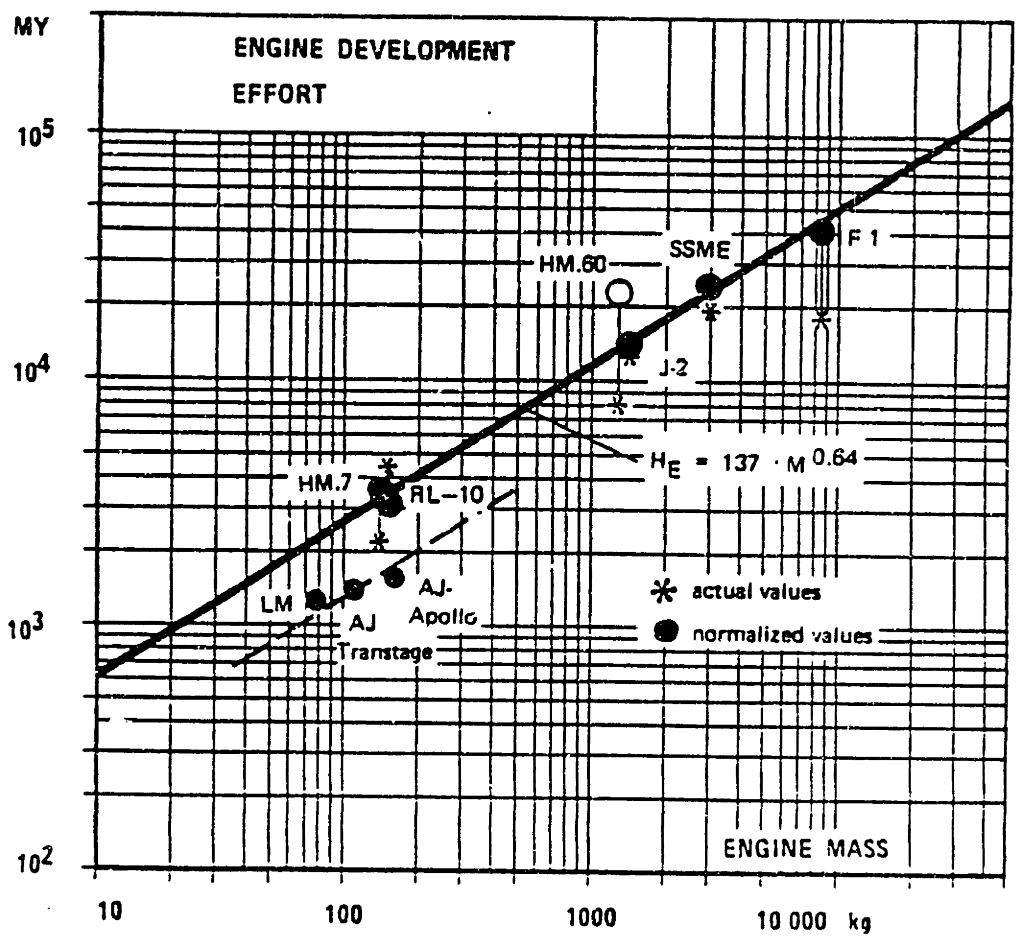


Figure 3-6: Transcost Rocket Engine Development Cost Equation

for the SSMEs in the model (Koelle, 1983, p. 20). The team experience factor, f3, was taken to be 0.75, based on industry's extensive experience with liquid-propellant rocket engines. Thus,

$$(3.7) \qquad Hr = 83 \cdot Mroc^{0.64} \ [MY]$$

The value to be used in this equation is not the "rocket engine mass" calculated in chapter 2. Instead, one-fifth of this value should be used since the ASP is likely to have five separate rocket engines--one for each engine module as shown on page 2. One-fifth of the 5675 kg calculated for M = 18 gives 1135 kg. Thus,

$$(3.8) \qquad Hr = 7507 \text{ MY} = \$961 \text{ million}$$

and by equation (3.4),

$$(3.9) \qquad Hatr = 7660 \text{ MY} = \$980 \text{ million}$$

Total Combined Engine Costs. As mentioned in section 2.6, NASA does not plan to employ three separate engine systems (ATRs, scramjets, and rockets) in the ASP. Instead, the goal is to develop a combined or hybrid system. A combined system integrates the features of two or more airbreathing engines into a single system (e.g., the ATR). A hybrid system integrates a rocket engine into an air-breathing engine system (e.g., the "scram-rocket"). Since the specific nature of the ASP's engine system remains to be determined, this cost analysis treated each individual propulsion mechanism as if it were a separate engine.

It is not clear whether this tends to over-estimate or under-estimate the actual costs of developing and fabricating a combined or hybrid engine system. On the one hand, the cost of the whole may be less than the sum of its parts because program management is consolidated, contract administration is shared, and some engine parts would serve multiple roles. On the other hand, the challenges of integrating such an engine may outweigh the above benefits. For lack of any compelling reason to assume one view over the other, I have assumed that the development and fabrication costs of the combined or hybrid engine is the simple sum of its component parts analyzed separately. Thus,

(3.10) $$He = Hatr + Hsj + Hr = (7660 + 7813 + 7507)\ MY = 22{,}980\ MY$$
$$He = \$2.94 \text{ billion}$$

For comparison, the official NASA budget cost summary for the SSME development FY 1971 to FY 1983 shows 18,842 MY, or $2.41 billion (Koelle, 1983, p. 20). Thus, the above estimate represents an ASP engine development cost about 22 percent higher than that of the SSMEs, a plausible result.

Total Development Costs. By equation (3.1), total development costs are 110 percent of the sum of the development costs of the individual components, or for this case:

(3.11) $$Cdev = 1.1\ (\$12.9 + \$2.94) \text{ billion, or}$$
$$Cdev = \$17.4 \text{ billion}$$

Conclusion. At this point a development cost assessment can be made. I have cited:

(1) British Aerospace's estimate of $6 billion in development costs for their unmanned, smaller HOTOL;

(2) the First Boston estimate of about $14 billion in development costs for the ASP; and

(3) my own Transcost-derived estimate of around $17 billion for the ASP.

Given these three alternative estimates, the official USAF estimate of $3 billion looks rather low. Furthermore, a comparison to less sophisticated aircraft is informative: a new subsonic aircraft today costs between $2 billion and $3 billion to develop (Cross, 1986, p. 52). NASA-Langley has estimated the cost of developing an advanced SST today to be about $5 billion (Cross, 1986, p. 52). As mentioned earlier in the introductory remarks about different cost estimating strategies, neither one of the alternative estimates alone would suffice to challenge the official estimate, but taken together, they do suggest that ASP development costs will be between $10 billion and $20 billion.

Vehicle	Development Cost (billions)
Subsonic aircraft	2
Advanced SST	5
HOTOL	6
Space Shuttle	14
ASP (First Boston)	14
ASP (Transcost)	17
ASP (USAF)	3 ?

Figure 3-7: Aerospace Vehicle Development Cost Comparison

PRODUCTION COSTS

General Equation. In the Transcost Model (Koelle, 1983, p. 21), the fundemental cost estimating relationship (CER) for production costs is the sum of the single elements costs times a factor of 1.02^N for management, integration, and checkout, given by:

(3.12) $$Cprod = 1.02^N (Faf + Fatr + Fsj + Fr) \quad [MY]$$

where Cprod = total vehicle fabrication costs

N = number of stages and engines

Faf = fabrication cost of the airframe

Fatr = fabrication cost of the five modular ATRs

Fsj = fabrication cost of the five modular scramjets

Fr = fabrication cost of the five modular rockets

The basic cost elements have the structure:

(3.13) $$F = n \cdot a \cdot M^x \cdot f4$$

where n = number of units (vehicles or engines) built

a = specific values for each type of system

x = specific values for each type of system

M = reference hardware mass (kg)

f4 = cost reduction factor for series production, depending on the learning factor and number of units built

ASP Fleet Size. A fourth correction factor is introduced in the production cost sub-model. The cost reduction factor due to the economy of serial production, f4, is a function of the number of units built. Thus, the estimate of the number of ASPs that would be needed has a strong influence on the ASP unit cost. Using rough approximations, one can estimate the required number of ASPs to meet demand for launch services in the time period 2000-2010. Assuming that:

1) the original shuttle orbiter fleet prior to the Challenger accident was sufficient to handle today's space traffic demand,

2) that this demand grows at a rate of 10 percent yearly,

3) that the number of ASPs needed is inversely proportional to the size of its payload bay,

4) that the number of ASPs needed is directly proportional to its ground turn-around time (GTAT), and

5) that the ASP fleet will be sized to meet the demand 20 years from now, i.e. 2006,

we get the following relation:

(3.14)

$$\text{number of ASPs needed} = (\text{number of shuttles in present fleet}) \times (\text{STS payload/ASP payload}) \times (\text{ASP GTAT/STS GTAT}) \times (1.1)^{20}$$

Taking the number of shuttles to be 4, the payload bay ratio to be 3, the GTAT ratio to be 1/8 (based on 1 week for the ASP and 8 weeks for the shuttle), we get 10 for the number of ASPs needed just for space launches (that is, excluding any ASPs built for non-space military missions as described in chapter 4). Note that this assumes that current shuttle payload manifests are divisible into thirds for launch using the ASP. As will be shown in section 4.2, "The ASP as Launch Vehicle," this is a plausible assumption for most paylaods.

This is almost certainly a minimum since in that time period, the shuttle's useful life will have expired, the space station will require logistical support, space commercialization will proceed, and there is the possibility that the nation will be in the process of deploying a space-based anti-ballistic missile defense system. The estimate of a 10 percent annual growth in demand for launch services is consistent with a recent joint NASA/DOD mission model update. In the fall of 1984, NASA and the DOD were asked by the President to identify its missions and corresponding launch system requirements for the 1995 to 2010 time frame (White House, "National Security Decision Directive-144, National Space Strategy," 1984). Their report projected that the combined launch requirements of NASA, the DOD (excluding SDI infrastructure deployments), and the commercial sector would grow at an annual rate of about 9 percent during the 1990s (Lucas, 1985, p. 2). Thus, my choice of a 10 percent growth rate is slightly higher than the official estimate, but may be justified on the grounds that development of the ASP would spur some additional demand.

The need for 10 ASPs requires that at least 50 of each type of engine be built since current ASP plans call for each vehicle to have five modular engines as shown in the figure on page 2.

Learning Curve. In 1936 an industrial operations analyst, T.P. Wright, defined the so-called "learning factor," p, which takes into account the fact that the more times a person does something, the easier, faster, more accurate, and consequently, cheaper the task becomes. This factor is defined as the ratio of the unit costs of the

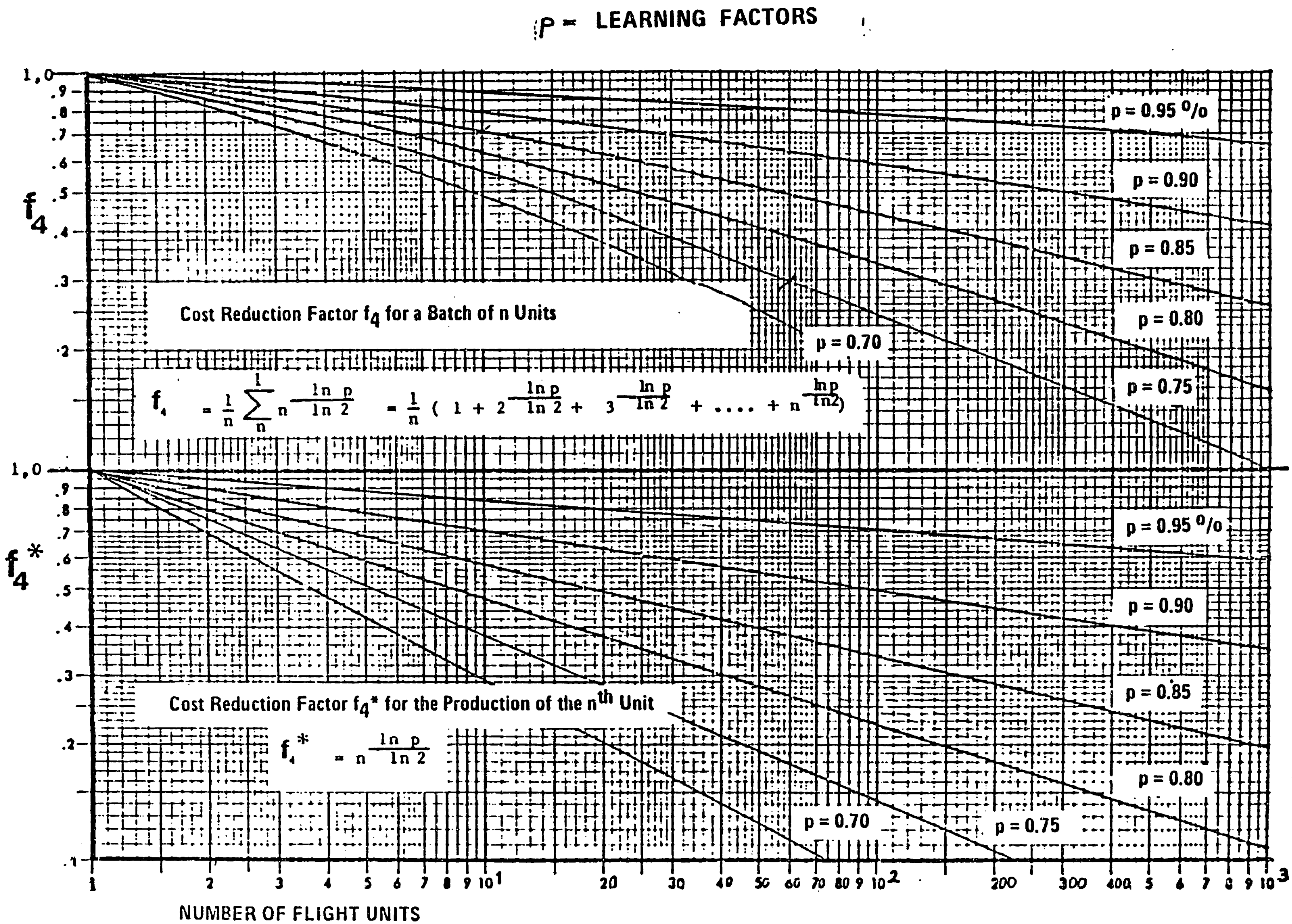
2Nth unit as compared to the Nth unit. Historically in the vicinity of 80 percent for aircraft production (Miller and Akin, 1980, p. 203), this factor of .80 means that the second unit costs only 80 percent of the first, the fourth unit only 80 percent of the second, and the eighth unit only 80 percent of the fourth and so on. I will assume a learning factor value of 80 percent for the ASP. Referring to figure 3-8, we can determine the values of f4 for the airframe and the engines. Since 10 airframes will be produced, the airframe f4 equals 0.62 (using the top chart, the cost reduction factor for a batch of units). Since 50 of each type of engine will be produced, the engine f4 equals 0.40.

ASP Airframe Production Costs. For winged space vehicles only a few reference points exist. The Transcost Model used both historical cost data from the space shuttle and Boeing's projected costs for its Heavy-Lift Solar Power Satellite Launchers (AIAA paper 78-316). Figure 3-9 presents the Transcost "First Unit Production" cost curves. These curves indicates that the fabrication cost of a manned, winged vehicle is approximately ten times higher than that of an unmanned, ballistic vehicle of the same mass. Thus, the preliminary CER for a reusable, winged orbiter is (Koelle, 1983, p. 28):

$$(3.15) \qquad Faf,n = n \cdot 54 \cdot Maf^{0.46} \cdot f4 \quad [MY]$$

where the notation Faf,n means the total fabrication cost for n airframes.

Using Maf = 109,848 kg as in the development cost calculation, gives:



$$f_4 = \frac{1}{n}\sum_{1}^{n} n^{\frac{\ln p}{\ln 2}} = \frac{1}{n}\left(1 + 2^{\frac{\ln p}{\ln 2}} + 3^{\frac{\ln p}{\ln 2}} + \dots + n^{\frac{\ln p}{\ln 2}}\right)$$

$$f_4^* = n^{\frac{\ln p}{\ln 2}}$$

Figure 3-8: Transcost Curves for Determining Value of f4, the Cost Reduction Factor

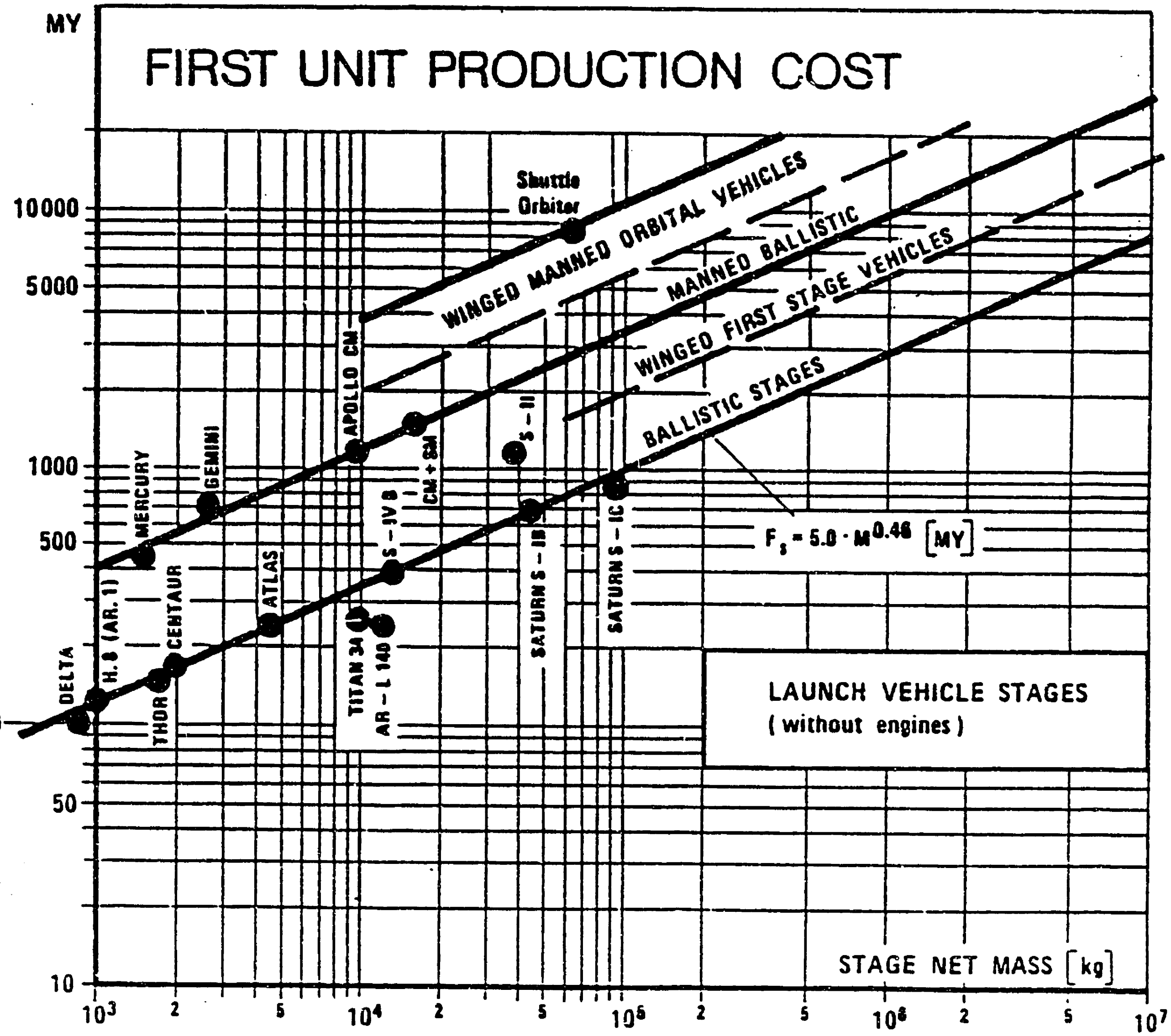


Figure 3-9: Transcost First Unit Production Cost Equations for Various Space Transportation Systems

(3.16) $F_{af,10}$ = \$8.9 billion, or per ASP:

F_{af} = 6963 MY = \$891 million

The exponent, 0.46, in the fabrication cost equation shows that a vehicle of 10 times higher mass requires only 3 times higher costs, or doubling the vehicle mass increases cost by only about 40 percent.

Booz-Allen Engine Production Cost Model. The Transcost Model does not provide information on the fabrication costs of air breathing engines, so for the ATRs and the scramjets, I have used a different cost model. The "Supersonic Transport Development and Production Cost Analysis Program" was developed by Booz-Allen & Hamilton, Inc. for the Federal Aviation Administration (FAA) in 1967 when the FAA was estimating the costs of supersonic transport (SST) development (Booz-Allen & Hamilton, Inc., 1967). This cost model served as the basis for the FAA's projections of US SST costs.

While correlating cost with mass has been shown to provide satisfactory estimates in the case of airframe production, the Booz-Allen study showed that the best characteristics to regress engine cost against were: (1) maximum thrust, (2) highest operational altitude, and (3) engine thrust to weight ratio. This study combined data from engine manufacturers, Federal agencies, and past studies to arrive at an advanced air-breathing engine CER. The following engines were included in the sample, which totalled 57 observations: J-52, J-57, J-58, J-60, J-69, J-75, J-79, J-85, TF-30, and TF-33. The majority of the information was supplied by Pratt & Whitney and General Electric, two of the leading contenders for ASP engine contracts.

The overall statistical results obtained in developing this engine CER are good--the correlation coefficient was 99 percent, standard errors were low, and all F- and t- ratios were significant at the 99 percent level. On the other hand, a great amount of extrapolation is required from the sample to estimate ASP engine costs. From a statistical standpoint, such extrapolations are sound as long as there are no significant discontinuities between the limits of the sample and the point of extrapolation.

Figure 3-10 shows the Booz-Allen engine CER as it was derived. I have changed the units of measurement to be consistent with the rest of this section and have altered the coefficient appropriately. The revised equation for the production cost of a single engine is:

$$\text{Fab} = 5.5\ (\text{TMAX})^{0.56} \cdot (\text{ALT})^{1.93} \cdot (\text{TN/W})^{0.48} \tag{3.17}$$

where:

- Fab = total production cost of air-breathing engine in 1963 dollars
- TMAX = maximum rated thrust (kg)
- ALT = absolute engine operational altitude (km)
- TN/W = normal rated thrust divided by the dry engine weight

<u>ATR and Scramjet Production Costs</u>. TMAX for the ATRs is 1/2 the initial take-off mass, since the thrust to weight ratio is 1/2, but this must be divided by 5 since we are assuming a modular engine design as shown on page 2. TMAX for the scramjets is 1/2 the ASP mass at scramjet ignition, again divided by 5. The absolute altitude of the ATRs will be taken as 16 km as in chapter 2, while that for the scramjets

Engine

The equations used to calculate engine costs are:

Production \$ = 4.80×10^{-4} (TMAX)$^{[illegible]}$ · (ALT)$^{[illegible]}$ · (TN/W)$^{[illegible]}$ · Q$^{[illegible]}$

Development \$ = .01 (HTEST)$^{[illegible]}$ · (ALT)$^{[illegible]}$ · Q$^{[illegible](TMAX) + [illegible](TA/A)}$

ETC \$ = −616.85 + .16 (TMAX)$^{[illegible]}$ + 388.68 LOG_{10}(ALT)

Where:

Production \$ = Total production cost, including profit, in millions.
Development \$ = Total development cost in millions.
ETC \$ = Development cost to Engine Test Certification in millions.
TMAX = Maximum-rated thrust in thousands of pounds.
Q = Quantity of engines being costed.
ALT = Absolute altitude in thousands of feet.
TN/W = Normal-rated thrust (k lbs.) divided by dry engine weight (lbs.).
HTEST = Heated test discontinuity variable.
TA/A = Maximum-rated thrust divided by the airflow.

The sum of these three equations is the total program cost of the engine.

[1] Developed by Booz-Allen Applied Research, Inc. and Resource Management Consultants, Inc. under contract to FAA.

Figure 3-10: Booz-Allen CER for Advanced Air-Breathing Engine Production Costs

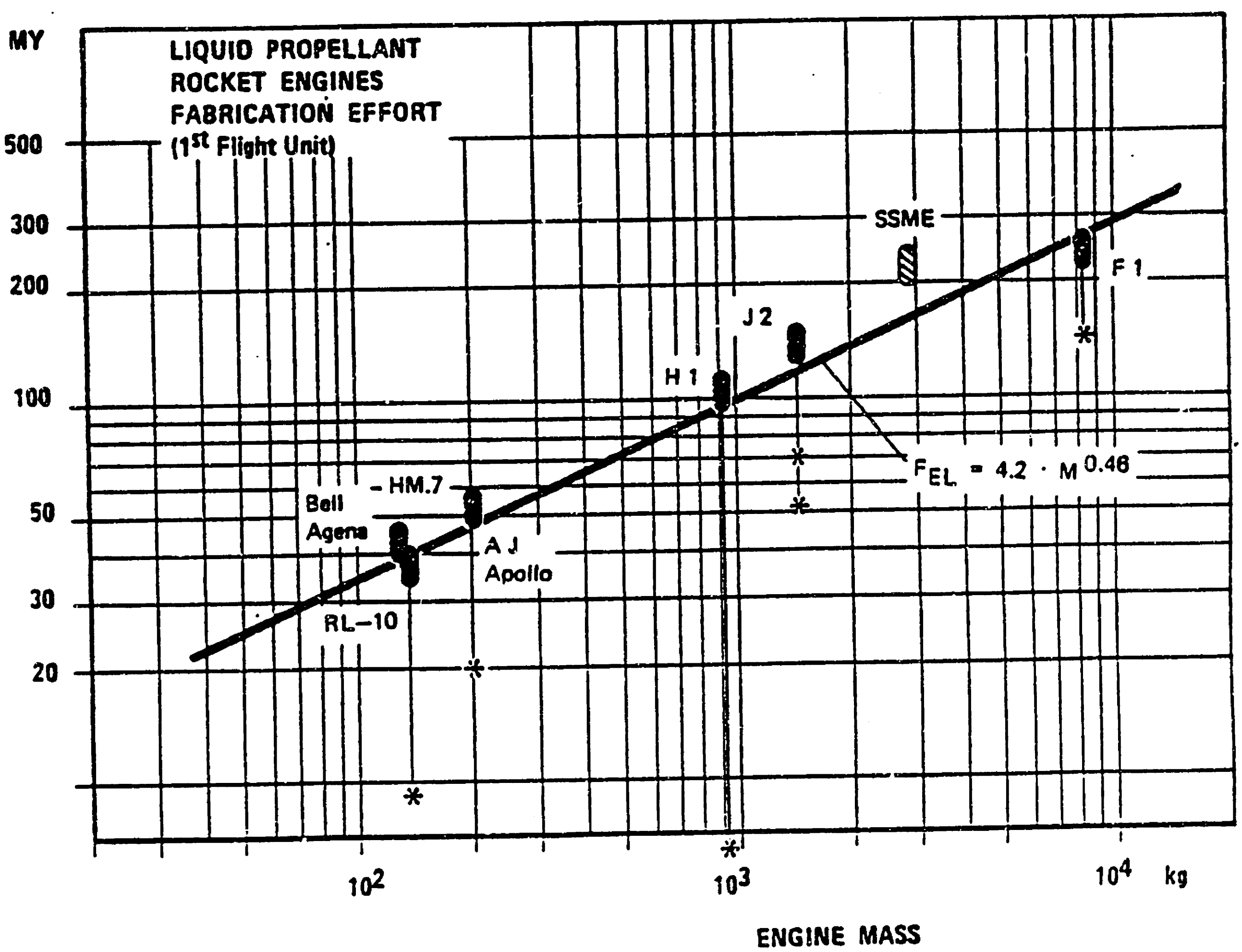


Figure 3-11: Transcost First Unit Production Cost Equation for Rocket Engines

will be taken to be a constant 45 km. Assuming a constant scramjet ceiling of 45 km permits the full range of scramjet operating regimes without distorting the cost calculations for other Mach number limits by varying this altitude limit. For the ATR, the value of TN/W was shown in section 2.6 to be 25; for the scramjet TN/W is about 10. Using GNP deflators again, in order to convert \$ 1963 to \$ 1985, multiply by 3.28. To get the total production costs for 50 engines of each type assuming 80 percent learning ($f4 = 0.4$), multiply again by $n \cdot f4$, or 20. Note that, unlike the other calculations which were in man-years, these cost estimates are directly in units of 1985 dollars. Thus,

(3.18) $$F_{atr,50} = \$361.80\ (Mo/10)^{0.56}\ (16)^{1.93}\ (25)^{0.48}$$
$$F_{atr,50} = \$358{,}830\ (Mo/10)^{0.56}$$

For Mo = 425,187 kg:

(3.19) $F_{atr,50}$ = 1100 MY = \$140.8 million, or per engine module:
F_{atr} = 22 MY = \$2.8 million

and for scramjets:

(3.20) $$F_{sj,50} = \$361.80\ (Mo\text{-}Mprop1/10)^{0.56}\ (45)^{1.93}\ (10)^{0.48}$$
$$F_{sj,50} = \$1{,}695{,}959\ ((Mo\text{-}Mprop1)/10)^{0.56}$$

For Mo-Mprop1 = 387,485 kg:

(3.21) $F_{sj,50}$ = 4944 MY = \$633 million, or per engine module:

$Fsj = 99\ MY = \$12.7$ million

Rocket Engine Production Costs. The fabrication costs for the rocket engines may be estimated by the Transcost CER shown in figure 3-11:

$$(3.22) \qquad Fr = n \cdot 4.2 \cdot Mroc^{0.46} \cdot f4 \quad [MY]$$

or substituting 5675/5 for Mroc as in equation (3.6), 50 for n, and 0.4 for f4:

$$(3.23) \qquad Fr,50 = 2131\ MY = \$273 \text{ million, or per engine module:}$$

$$Fr = 43\ MY = \$5.5 \text{ million}$$

Total Combined Engine Cost. As described in the development cost section, I have assumed that the development and fabrication costs of the combined or hybrid engine is the simple sum of its component parts analyzed separately. Thus,

$$(3.24) \qquad Fe = Fatr + Fsj + Fr = (22 + 99 + 43)\ MY = 164\ MY$$

$$Fe = (\$2.8 + \$12.7 + \$5.5) \text{ million} = \$21 \text{ million}$$

where Fe = the production cost of a single integrated ASP engine module

Since current planning calls for 5 engine modules per ASP (see page 2), the total engine fabrication costs for each ASP would be \$21 million times 5, or \$105 million.

For comparison, the SSMEs cost approximately $27 million per engine to produce (Koelle, 1983, p. 26). Each shuttle carries 3 SSMEs; thus, the total engine fabrication cost per shuttle is about $81 million. In the previous section, ASP engine development costs were estimated to be about 22 percent higher than than SSME development costs, reflecting their greater technical complexity. Here, total propulsion system fabrication costs for the ASP are estimated to be about 30 percent higher than total SSME fabrication costs, even though the mass of the ASP engine system is about 5 times greater than the combined mass of the three SSMEs (33,553 kg v. 6,000 kg).

This shows the effect of "the economy of serial production," due to the large number of ASP engines produced. In the algorithm, this is the factor f4, which was shown in figure 3-9 to be 0.4 assuming a production run of 50 and a learning rate of 80 percent. If there were only one ASP produced, and thus, 5 engines produced, the learning factor, f4, would be 0.75. The total ASP propulsion system fabrication costs would then be $105 x 0.75/0.4 million, or $197 million.

Total Production Costs. The total production cost of an ASP is 5 times the sum of the fabrication costs of the individual engine modules plus the fabrication cost of an ASP airframe, all multiplied by the factor 1.02 , for management, integration, and check-out. Using the values calculated above, and taking N to be 6 since there are 5 engine modules and 1 airframe, gives:

(3.25) $$C_{prod} = 1.13\ [5\ (164) + 6963]\ MY = 8765\ MY, \text{ or}$$

Cprod = \$1.12 billion per ASP (assuming a production run of 10)

The cost of procuring a fleet of 10 ASPs is, then, approximately \$11.2 billion, which when added to the development costs (\$17.4 billion), gives a total of about \$28.6 billion in non-recurring expenses.

OPERATIONS COSTS

Operations Cost Model. In the areas of development and production costs, both the costs and the product are well defined by the industrial contract. In the launch operations area, nothing comparable exists yet. The Transcost model acknowledges the lack of a sizable data base on which to establish operations costs models. However, a preliminary version has been published, bearing in mind that launch operations costs are very difficult to analyze--there being so many intangibles: weather, schedules, mission requirements, aborts, and housekeeping cost allocation. Flight operations costs may be broken down into three categories: (1) direct operations costs, (2) refurbishment costs, and (3) indirect operations costs (see figure 3-12).

Direct operations costs include costs associated with: (1) technical systems management, (2) prelaunch operations, (3) launch and mission control, (4) propellant usage, and (5) recovery and transport costs.

Refurbishment costs are the costs associated with maintaining, servicing, repairing, and replacing various components between flights. Clearly, the number of re-uses per ASP will have an influence on the total refurbishment costs. The most economic number of flights for a reusable vehicle is still an open issue: the present range of assumptions is between 20 and 300; and for rocket engines between 10 and 100 (Koelle, 1984, p. 42). Recent NASA estimates (Eldred, 1984, p. 5) suggest that the optimum number of reuses is closer to the high end of these ranges. Thus, I will assume the ASP life to be 300 flights and the

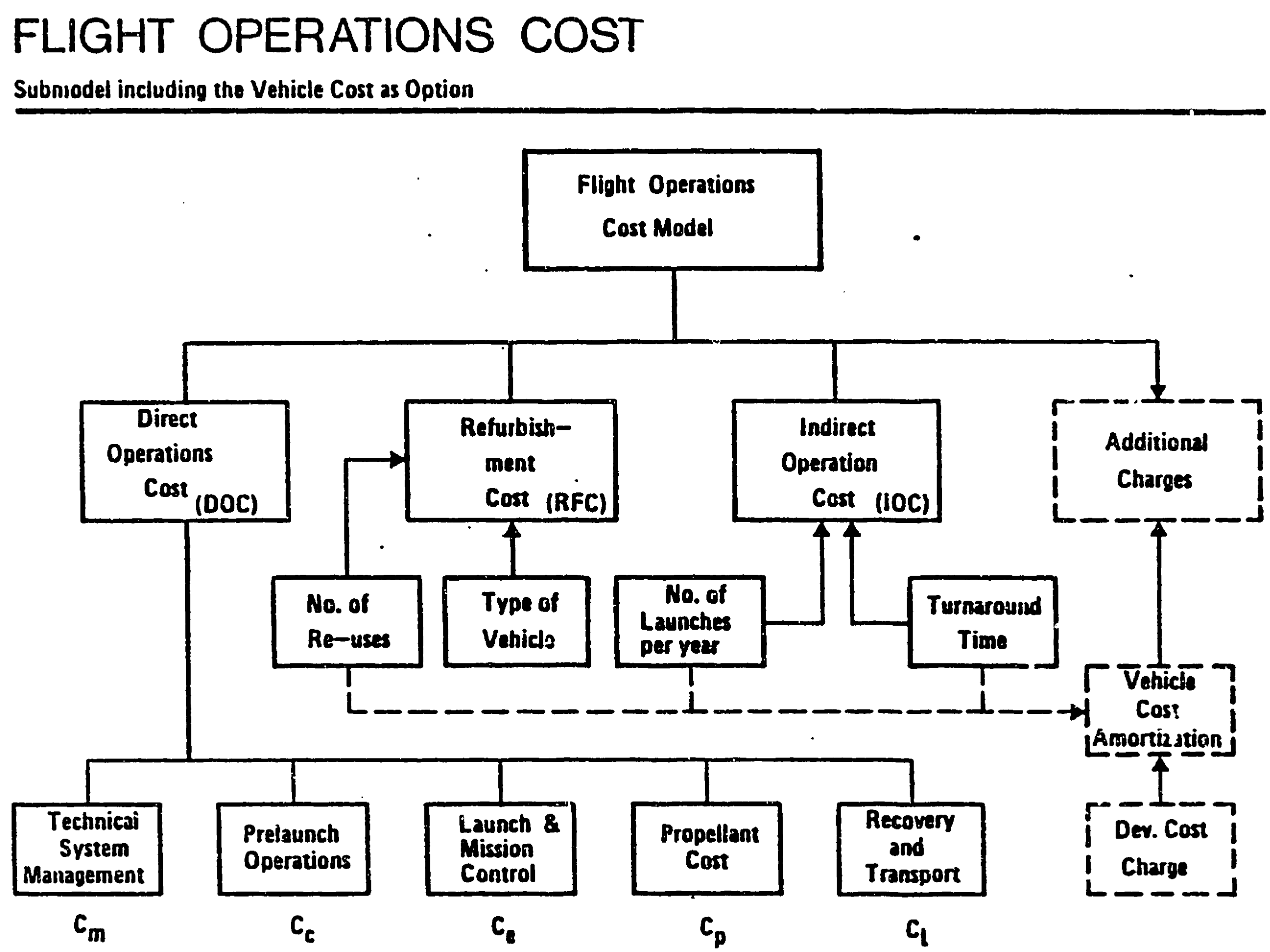


Figure 3-12: Transcost Flight Operations Cost Model Organization

engine life to be 100 flights.

Indirect operations costs are those related to the launch base and flight range. These costs consist of fixed costs for: (1) site management and administration, (2) engineering support, (3) facilities maintenance and amortization, and (4) spares storage and supply service.

Annual ASP Launch Rate. In contrast to the development and production cost equations, the operations cost equations are based on the assumption that operations costs are a fixed amount per launch, essentially independent of vehicle size, however, influenced by the type of vehicle and number of stages. Thus the annual launch rate comes into these equations as a primary influence since it affects the facilities required.

In the process of estimating the number of ASP launches per year, I will assume that the ASPs would not be utilized at their maximum rate (GTAT is projected to be about a week) of 52 launches per year per ASP. Instead, it is probable that they would be flown at a slower initial rate, allowing room for traffic growth and for decreasing the GTAT to the desired goal as operational experience is gained. This initial utilization factor I will take to be about 40 percent, giving about 21 launches per year per ASP. Furthermore, I will assume for these calculations that the ASP fleet is evenly distributed between the east coast (KSC) and the west coast (VAFB). Twenty-one launches per year times 5 ASPs per launch site gives about 105 launches per year from each site, or 210 launches per year in total (roughly 4 launches per week).

Assuming an average payload bay load factor of 80 percent per launch, 210 launches per year at 8,000 kg per launch is about 1,680 metric tons per year placed in LEO. This is consistent with current NASA/DOD joint estimates of the combined DOD, civilian, and NASA to-orbit traffic for this time period (Lucas, 1985, p. 4). These estimates forecast an annual mass requirement to LEO at the turn of the century of approximately 1,600 tons; 3,200 tons if SDI deployment occurs. Currently, some 300 metric tons are delivered to LEO annually.

Technical System Management Costs. As mentioned above, the Transcost model assumes that the costs for technical system management are independent of vehicle size, however they are influenced by the vehicle type and the launch rate. The vehicle type is given consideration by the proper selection of a coefficient for the cost relationship. Not surprisingly, reusable vehicles are more costly to manage than expendable ones. The launch rate is raised to the -0.35 power, indicating that the per launch costs for technical system management decrease as launch rate increases. Again, it is understandable that a higher launch rate results in lower costs per launch since the investment in fixed equipment can be amortized over more launches. Specifically, doubling the launch rate results in a 22 percent decrease in costs per launch for this element. Note that in the following calculations, the costs are expressed in man-years per launch and dollars per launch. The CER was found to be (Koelle, 1983, p. 34):

(3.26) $$Cm = (7 + a)\, L^{-0.35} \quad [\text{MY/launch}]$$

where L = number of ASP launches per year per site = 105 a = 3 for an expendable vehicle

a = 5 for a reusable vehicle

Thus, for the ASP, a = 5, and

$$C_m = \$300{,}000 \text{ per launch} \tag{3.27}$$

When applied to the Ariane 1, for example, this specific CER is accurate to within 20 percent (Koelle, 1983, p. 34). At 4 launches per year, the model predicts $1,280,000 per launch. The actual value is about $1,540,000 per launch.

<u>Pre-launch Operations, Assembly, and Check-out Costs</u>. While the costs associated with technical systems management may not decrease much in the near future (all technical systems require managers, engineers, technicians, accountants, etc), the chances are very good that the costs of pre-launch operations, assembly, and check-out will decrease significantly.

The Transcost model is predicated on extrapolations from existing ground support methods, and as described in Chapter 1, future space transportation systems will have radically different manpower requirements. A shuttle launch requires about 1500 people on-site, 6000 if software programmers are included (Bekey, 1986). NASA Advanced Systems planners are working towards the goal of reducing the required manpower for the launch of an orbit-on-demand vehicle to 15. Therefore, any advanced launch vehicle, be it horizontal take-off or vertical take-off, will likely be designed for vastly streamlined launch operations.

Thus, the Transcost CER for this aspect of operations costs is likely to over-estimate the actual future costs, even though it is accurate to within a few percent of current vehicles (3 percent for Ariane 1). In the model, the following CER has been defined for pre-launch operations, assembly, and check-out (Koelle, 1983, p. 34):

(3.28) $$Cc = (16 + b1 + b2 + \ldots + bn)\, L^{-0.35} \text{ [MY/launch]}$$

where b1,2,3... = dedicated values for each stage, depending on its technical complexity:

b = 5 for a solid motor kick stage

b = 12 for an expendable stage

b = 15 for expendable LH2 stages

b = 20 for reusable flyback systems that land at or near the launch site

b = 25 for reusable ballistic first stages to be returned to the launch site

Thus, for the ASP, b = 20, and

(3.29) Cc = $900,000 per launch, based on current practices

Currently, extensive research is being conducted on ways to decrease this component of operations cost. It does not seem overly optimistic to postulate that if and when the technological challenges posed by the ASP are overcome, the efficiency of the pre-launch assembly and check-out phase will be doubled. Thus, while it may seem arbitrary to halve the above cost, this is almost certain to provide a more accurate estimate than the current figure. Thus,

(3.30) $\quad Cc = \$450{,}000$ per launch, (based on a doubling the efficiency of the pre-launch, assembly, and check-out phase)

<u>Launch and Mission Control Costs</u>. The costs associated with launch and mission control operations and software depends on the vehicle complexity, the mission profile, and to a smaller extent on the annual launch rate. In the case of the ASP, it would be necessary to have much of the mission control software on-board since the vehicle is intended to fly on short notice and in rapidly changing environments far from the launch site. However, the ultimate location of the software is largely irrelevant to this analysis; it will still need to be written and programmed. Thus, we can assume that the Transcost CER is still valid for this aspect of operations costs (Koelle, 1983, p. 38):

(3.31) $\quad Ce = (4 + c)\, L^{-0.15}$ [MY/launch]

where $c = 1$ for expendable stages

$c = 2$ for stages to be recovered

$c = 4$ for unmanned reusable orbital stages

$c = 6$ for a manned orbital system

Thus, for the ASP, $c = 6$, and

(3.32) $Ce = \$640,000$ per launch

Propellant Costs. The propellant cost depends on the amount of propellant needed per launch, the number of launches per year, any specific cost reduction due to economies of scale, and the fraction of propellant that boils-off between propellant delivery and actual launch. The amount of propellant needed per launch was determined in chapter 2. Figure 3-13 from the Transcost model shows propellant costs for LH2 and LOX, in man-years, as a function of the daily production rate. The boil-off rate I will assume to be half the shuttle's since the ASPs will be launched faster. Koelle (1983, p. 39) cites values of 35 percent LH2 boil-off and 20 percent LOX boil-off for the shuttle. Thus, I will assume 17 percent LH2 boil-off and 10 percent LOX boil-off for the ASP.

A stoichiometric mixture of hydrogen and oxygen will react as:

(3.33) $$2\ H_2 + O_2 \rightarrow 2\ H_2O$$

Thus, the proportion of the rocket engine fuel that is hydrogen is 2(2)/2(2)+32, or 1/9. Likewise, the proportion of the combined fuel that is oxygen is 8/9. Using the same example (Mach number 18) as before, we have the total hydrogen mass needed per launch equal to the sum of the ATR propellant mass, the scramjet propellant mass, and one-ninth of the rocket engine propellant mass, or 152,059 kg. Multiplying this by 105 launches per site per year, multiplying by 1.17 to account for LH2 boil-off, and dividing by 365 days per year, gives a LH2 production level of 51,000 kg per day. Referring to figure 3-13, we see that a production rate of 51,000 kg per day results in production costs of about 12 MY per

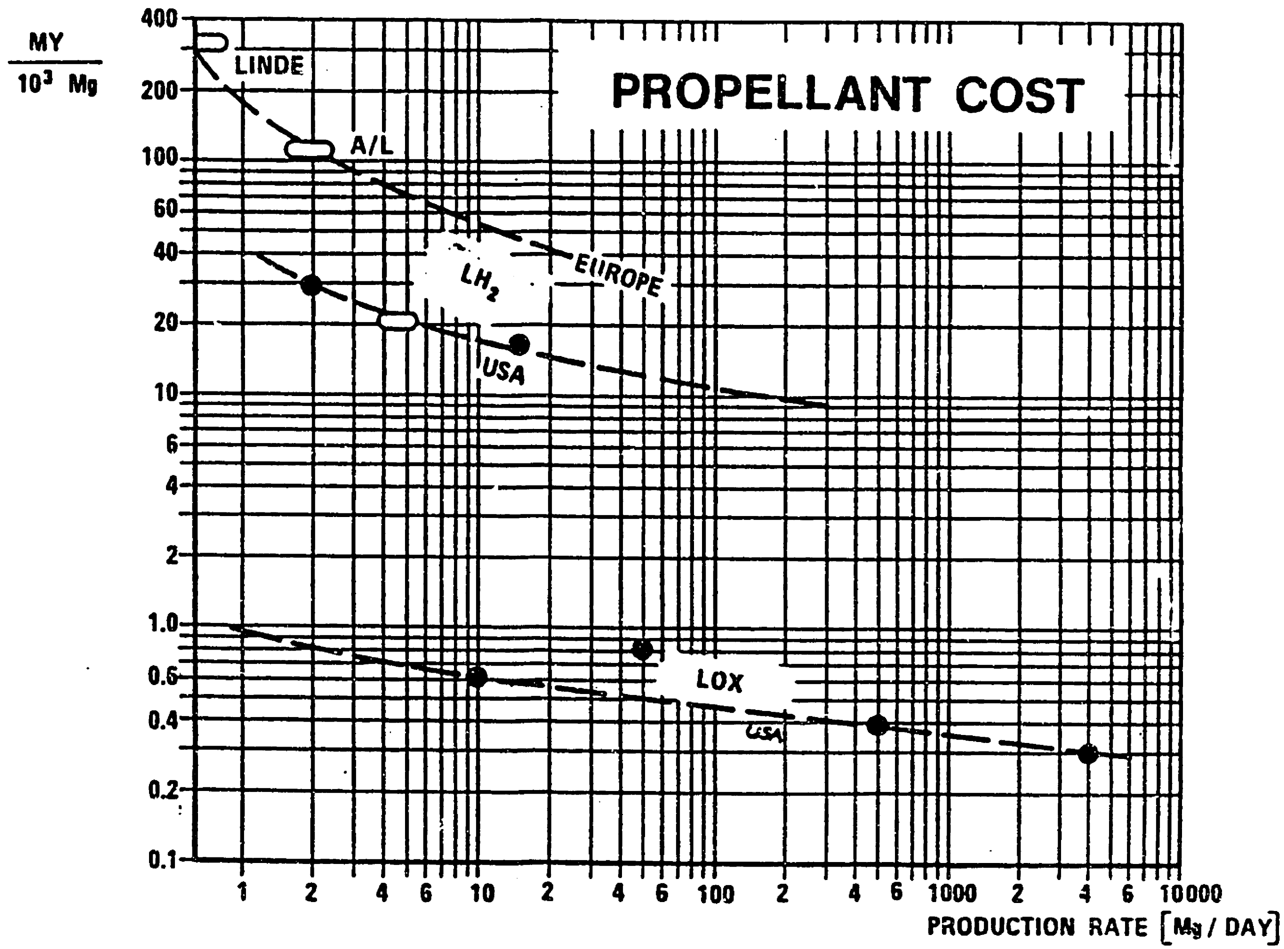


Figure 3-13: Propellant Cost (LH_2/LOX) vs. Production Quantity

million kg. This works out to be \$234,000 (\$ 1985) per launch for liquid hydrogen.

The remainder of the fuel is LOX. This consists of the other eight-ninths of the rocket engine proellant, or 119,723 kg. This is about 10 times less LOX than the space shuttle carries. Performing the same calculations as for LH2, yields a LOX production level of 38,000 kg per day. Referring to figure 3-13 gives a cost associated with this production level of 0.5 MY per million kg. The liquid oxygen costs are then, \$8,000 (\$ 1985) per launch. Total propellant costs are then the sum of the liquid hydrogen costs (\$234,000) and the liquid oxygen costs (\$8,000):

$$\text{Cprop} = \$242{,}000 \text{ per launch} \tag{3.34}$$

Referring back to figure 1-3, the break-down of space shuttle operations costs, we see that liquid propellant costs represent about \$2 million of the space shuttle's operations costs. This analysis indicates that the ASP's propellant costs are likely to be in the vicinity of \$242,000, or about 12 percent of the shuttle's.

<u>Recovery Costs</u>. Recovery costs apply to first stages or boosters which fall into the ocean and have to be recovered and towed back to the launch site for refurbishment. Since the ASP will be a single-stage, fully reusable vehicle, the recovery costs do not apply.

Refurbishment Costs. Refurbishment costs are a special category of costs related to both fabrication costs (of spare units) and manpower costs (to perform the work). Refurbishment costs, as noted above, are affected by the number of reuses per ASP. Certainly, different ASP subsystems will have different lifetimes, so the amount of refurbishment will vary from flight to flight, with a general trend to increase. The general philosophy of refurbishment is to replace the vehicle once the total accumulated refurbishment costs have reached the cost of fabricating a new vehicle.

Since very little reference data exist yet about this type of cost, it is appropriate to modify the Transcost model to conform better to the specific nature of the ASP. The Transcost model suggests the following CER (Koelle, 1983, p. 43):

$$Crf = (Rs + Rg) + (n \cdot Re) + 10^{-5}(Laf \cdot Faf) + 5 \cdot 10^{-5}(Le \cdot Fe) \qquad (3.35)$$

where:

n = number of engines, in this case: 5 combined-cycle engines

Rs = 3 MY for inspection, partial disassembly/assembly and test operations (from Transcost Model)

Rg = 2 MY for inspection, check-out and partial replacement of guidance, control, and telemetry equipment (from Transcost Model)

Re = 0.2 MY for inspection, control, and refurbishment of each engine (from Transcost Model)

Laf = design ASP airframe lifetime, taken to be 300 (Eldred, 1984, p. 5)

Faf = ASP airframe fabrication cost = 6963 MY from equation (3.16)

Le = design ASP engine lifetime, taken to be 100 (Eldred, 1984, p. 5)

Fe = fabrication cost for a single ASP engines module = 164 MY from equation (3.24)

Does this cost model make sense? The first observation we make is that the first two terms in the equation are fixed manpower costs; the second two terms are variable parts costs. It comes as no surprise that refurbishment costs would have variable and fixed components--parts and labor.

Second, we see that the variable parts cost scales linearly with both the number of reuses per component, and the fabrication cost of that component. This, too, makes sense in that more money is needed to keep a vehicle running longer. The annual maintenance costs for an automobile intended to be kept only five years are less than if a fifteen year life is desired. Similarly, assuming equal reliabilities, the maintenance costs for a Rolls-Royce are higher than those for a Chevy.

Thus, there is logic behind the formulation of this refurbishment cost equation. However, one modification is in order. After the Transcost Model was published, we have had the opportunity to gather some actual data on the reliability of complex engine systems--namely, the SSME maintenance record in figure 1-5. Like the SSMEs, the ASP engines represent an extremely complex, state-of-the-art engine system, and thus may experience similar "growing pains." Figure 1-5 illustrates that the average manpower required to repair the SSMEs after each of the first 12 flights was 632 man-days, or 1.7 MY. Replacing the second term in (3.35) with 1.7 MY gives a total refurbishment cost of:

(3.36) $$Crf = 5 + 1.7 + 20.9 + 0.8 = 28.4 \text{ MY} = \$3.6 \text{ million}$$

It should be noted that the first two terms are the costs of labor and the second two terms are the costs of replacement parts. In the cost accounting below, the labor costs will be included in the manpower costs.

Total Operations Costs. Summing the elements of operations costs gives:

(3.37) $$\begin{aligned} Cops &= Cm + Cc + Ce + Cprop + Crf \\ &= \$300{,}000 + \$450{,}000 + \$640{,}000 + \$242{,}000 + \$3{,}600{,}000 \\ &= \$5.2 \text{ million per launch} \end{aligned}$$

This operating cost, \$520/kg or \$240/lb, is about 10 times lower than that of the shuttle (see figure 1-2); not quite the 2 orders of magnitude ASP supporters advertise, but a substantial improvement nevertheless. Propellant costs are about 5 percent of the direct operating costs. Refurbishment (parts only) accounts for 53 percent. The other 42 percent is manpower costs (Cm, Cc, Ce, and the labor component of Crf).

It should be emphasized that the operations cost calculations were based on a launch rate of 105 launches per year per site. This is over 10 times the pre-Challenger launch rate, so it is not surprising that drastically different operations costs result.

Manpower Required. A rough estimate of the manpower required to perform the launch operations described above can now be made. Fortytwo percent of \$5.2 million is the costs due to manpower of various sorts. This is equal to \$2.2 million, or 17 MY. But one MY equals 52 man-weeks, so the manpower can be expressed as 887 man-weeks. If ASP GTAT was one week, then about 887 people would be required to perform the necessary operations. However, in the beginning of the operations cost section I assumed an initial ASP utilization rate of 40 percent of its maximum turn-around rate. Thus, instead of one launch every 7 days, initially there would be one launch every 18 days. Assuming that extending the GTAT reduces the required manpower proportionately gives an initial manpower requirement of about 355 persons. As mentioned earlier, the shuttle requires a ground crew of about 6000 people in total.

3.6 Costs for Other Scramjet Performance Levels

Figure 3-14 on the following page presents the cost break-down for other values of the maximum scramjet operating speed. Appendix C contains a more detailed listing of the cost components.

Cost estimates of this type can be used by budget analysts in a variety of ways. For example, at some point in the future, the question may arise as to whether it is worth additional R&D expenditures to attempt to increase the operating range of scramjet engines, say from Mach 19 to Mach 20. Referring to Appendix C, we see that the difference in the non-recurring costs (R&D plus fleet fabrication) resulting from an increase in scramjet maximum operating speed from Mach 19 to Mach 20 is about \$700 million. Thus, \$700 million is the maximum one should pay for such an increase (neglecting the small benefits in recurring costs). This is a rough example meant to show how analyses like this can be used; it is not meant to be taken literally.

Mach number at Scramjet cut-off	Development Costs (billions)	Fabrication Costs (billions)	Operations Costs (millions)
12	27.5	2.6	9.8
13	23.3	2.0	7.8
14	21.1	1.7	6.8
15	19.7	1.5	6.1
16	18.6	1.3	5.8
17	17.9	1.2	5.5
18	17.4	1.1	5.2
19	17.0	1.1	5.1
20	16.7	1.0	5.0
21	16.4	1.0	4.9
22	16.3	1.0	4.8
23	16.2	1.0	4.8

Figure 3-14: ASP Costs v. Scramjet Cut-off Mach Number

3.7 ASP Financing Considerations

In the foreseeable future, it will be extremely difficult to obtain government commitment to long-term projects, because of more immediate budget demands. And by themselves, private companies are not well-suited to fund research and development programs that involve high risks and require a long period of investment. Corporate investment horizons are measured in years, not decades. Stockholders expect a return on investment, at the very least, within their lifetime. Sometimes, subject to anti-trust restrictions, large projects can be undertaken as joint ventures with corporations pooling their resources to form an R&D

consortium (e.g., Bobby Ray Inman's Microelectronics and Computer Technology Corporation).

Other times, government involvement is required when the potential pay-off is significantly "in the national interest", but the risk is still quite large. Yet, political considerations place limits on the level of government investment in projects with commercial potential, such as the ASP. Many are opposed to having the government fund technologies over many years, at great cost to the taxpayer, and then hand them over to private interests for private gain (e.g., the COMSAT experience). The opinion of the current Administration is that investment should be driven by market forces whenever possible. If the ASP shapes up as primarily a commercial vehicle, financing its development will require a mixture of public and private sector investment, recognizing the needs of both industry and government.

If the ASP still looks promising after the government's technology demonstrations are completed, it will be time to negotiate a joint government-industry financing package. A cost-sharing scheme (perhaps 75 percent government, and 25 percent industry as was used for the SST program) would protect the government against loose spending by contractors, make Congressional appropriations go further, and demonstrate industry interest in the program. Other options such as loan guarantees, special tax provisions, and market guarantees provide only marginal incentives, since they are subject to policy changes over time.

Most of all, the financing package depends on whether the ASP concept evolves as a military vehicle, a commercial launch vehicle, or a hypersonic transport. A strictly military vehicle would not require any private sector investment; a commercial launch vehicle would require some; a hypersonic transport would require a great deal of industry participation. The question of which of these applications does the ASP concept make sense for is addressed in the next chapter.

CHAPTER 4: ASP APPLICATIONS AND INSTITUTIONAL PERSPECTIVES

"We are talking about the speed of response of an ICBM and the flexibility and recallability of a bomber, packaged together in a plane that can scramble, get into orbit, and change orbit so the Soviets can't get a reading accurate enough to shoot at it."

- Gen. Lawrence A. Skantze, commander USAF Systems Command

"We are going forward with research on an aircraft that could, by the end of the next decade, take-off from Dulles Airport and accelerate up to 25 times the speed of sound, attaining a low-earth orbit and fly to Tokyo within two hours."

- President Ronald Reagan

"I was not among those who shouted "Oh boy!" when President Reagan endorsed a flying machine to get people from the East Coast to Japan in two hours. There is something dull about places that are two hours away, and I can't believe Japan will be an exception."

- Russell Baker, humorist

4.1 The Aerospace Plane--"What is it?"

The ASP is undergoing an identity crisis. Eliot Marshall, in a commentary in Science magazine, has described how the ASP is seen as all things to all people: "What is it [the ASP]? Is it the shuttle's next of kin, a self-propelled vehicle to carry freight into orbit, or is it to be--as news stories report--an incredibly fast passenger plane, able to whisk executives from Los Angeles to Tokyo in two hours? The enthusiasts say it will be both; the catious ones say, a research plane; the realists say, a military project (Marshall, 1986)."

Implicit in Marshall's observation is the question as to why the ASP has been touted for so many applications. Is it indicative that hypersonic, air-breathing engine technology truly has potential for a wide range of aerospace applications, or is it "hypersonic hyperbole" designed to generate a broad-based constituency in favor of the project?

Undeniably, the surge in hypersonic propulsion research could foster the development of air-breathing launch vehicles, civilian hypersonic transports, or transatmospheric military aircraft. However, this chapter suggests that while the ASP concept does have possibilities as a launch vehicle, its potential as a hypersonic transport or as a useful military aircraft is overstated.

To clarify, no one proposes that any one vehicle be designed to perform all these functions. Rather, that much of the technology being developed in the hypersonic propulsion and thermal protection areas could be assembled to yield a variety of configurations, including a launch vehicle, a hypersonic transport, or a military aircraft. The discussions in the first three chapters have only been relevant to the ASP as a launch vehicle; this chapter considers other potential applications of the above technologies. It is necessary to evaluate the other applications before reaching any programmatic conclusions because the ASP program is justified on the grounds that it may benefit space transport, civilian aviation, and the military. For consistency, I have used the rubric ASP for all three applications, regardless of the differences in their mission requirements. Both the launch vehicle and military aircraft configurations would require a top speed of orbital velocity to perform their missions; the hypersonic transport might only need a Mach 5 capability.

4.2 The Aerospace Plane as a Launch Vehicle

Concept. Development of ASP technologies may be best suited for the purpose of delivering payloads to LEO at low cost. In chapter 3, operations costs in the vicinity of $520/kg were estimated, about one-tenth the cost with the shuttle.

Thus, we have come full circle; a space shuttle retrospective in chapter 1 identified the factors crucial to reducing launch costs and improving launch vehicle performance, while the technology analysis of chapter 2 provided a preliminary ASP mass baseline from which the development, fabrication, and operations costs of chapter 3 were derived. Now it is time to analyze the ASP baseline of chapters 2 and 3 in light of the criteria of chapter 1 in order to see whether any unique characteristics of the ASP make it cost-effective, or whether its low-cost is the result of external, "generic" factors. If it is the result of factors which are not intrinsic to the ASP, then we must consider whether the same cost savings would apply to alternative launch vehicle configurations.

To reiterate, the desired launch vehicle characteristics of chapter 1 were: (1) vastly reduced manpower requirements, (2) a high launch rate, (3) technical reliability, (4) total reusability, and (5) specialization as a low cost launch vehicle (as opposed to a "jack-of-all-trades" launch vehicle). Of course, these factors interact: more reliable engines help increase launch frequency, full reusability reduces manpower requirements and so on.

1. Manpower. In light of the first criterion, the cost analysis of section 3.5 indicated that approximately 350 persons would be necessary to support the launch of an ASP as opposed to the 6000 needed for the space shuttle. Since, according to the Transcost data, the required manpower per launch is approximately inversely proportional to the annual number of launches, the ASP's decreased manpower requirements are largely a result of its high launch rate and not from any specific characteristic of the ASP.

A second cause of the ASP's low manpower requirements is my assumption that the efficiency of pre-launch operations, assembly, and check-out will double by the turn of the century. This is based on the fact that current launch vehicles are processed vertically, while the ASP would be processed horizontally. Common sense and our experience with airplanes says horizontal processing would be more efficient; imagine if airplanes had to be serviced while standing vertically on their tails. It is generally accepted that a single-stage-to-orbit, horizontal take-off vehicle powered by pure rocket propulsion is impractical because the vehicle would be huge and its performance marginal. However, horizontal processing is not an area where the ASP gains unilateral advantage: a vertically-launched rocket could also be processed horizontally, towed unfueled to the launch pad, and then be erected, fueled, and launched.

A unique feature of the ASP that would reduce needed manpower is that the air-breathing engines would serve the ASP in vehicle self-ferry operations, thus simplifying overall logistics and eliminating the requirement for special carrier aircraft and mating/demating ground facilities.

2. High Launch Rate. While the launch rate of 105 launches per site per year (two sites) postulated in this analysis may seem high by today's standards, it is, nonetheless, consistent with the official NASA/DOD forecast of demand for launch services at the turn of the century (1,600 tons to LEO annually). The ASP requires such a high launch rate because its payload bay would be three times smaller than the shuttle's.

The ASP's smaller payload bay size offers other advantages, unrelated to the ASP's nature as single-stage-to-orbit, air-breathing launch vehicle. First, the ASP's small payload bay would mean that most ASP launches would be dedicated to serving a single customer. This permits a greater degree of flexibility with respect to launch windows and scheduling.

Second, payload launch insurance would be more readily available since the maximum possible loss in case of an accident would be smaller.

Third, on-orbit conflicts between payloads would be reduced. Several shuttle experiments have been disrupted when one payload's operations adversely affected another's.

Fourth, the ASP's smaller payload bay size serves to reduce the risk associated with inaccurate projections of future launch demand. As demand estimates become more concrete, the ASP fleet size could be tailored more responsively than, for example, the shuttle fleet could since the shuttle payload quantization is more "lumpy."

Lastly, the smaller payload bay size offers economy. The dimensions of the reference ASP payload bay were estimated in chapter 2 to be about 3.08 m in diameter and 12.33 m in length. This could easily accomodate most communications satellites headed for LEO or geostationary orbit (GEO) along with their perigee kick motors. In fact, British Aerospace conducted a market survey of payload masses to different orbits and arrived at the following conclusion: a vehicle designed with a payload mass capability of only 8,000 kg would be able to launch 70 percent of all satellites destined for GEO, 70 percent of all satellites heading for a polar orbit, and 90 percent of all satellites bound for LEO (Parkinson, 1985). This is the design goal for their HOTOL.

Thus, the smaller payload bay is more tailored to the needs of the commercial customers. It would be three times smaller than the shuttle's payload bay, but could still capture up to 90 percent of the commercial launch service market. This could very well price shuttle-sized vehicles out of the market, leaving them the remaining occasional large payloads such as the Space Telescope and photo-reconnaissance satellites.

3. Technical Reliability. It is likely that the complex combined-cycle, air-breathing/rocket engine needed for the ASP would be less

reliable than an incrementally-improved, conventional rocket engine used on an advanced vertically-launched rocket. Indeed, my assumption in the chapter 3 cost analysis that the ASP engines would require the same amount of maintenance and repair as the SSMEs probably understates the complexity of the ASP engine system. In general, air-breathing launch vehicles would involve new technologies; rockets utilize proven technologies, and even they are far from fully understood, as the recent ELV failures attest.

On the other hand, there are indications that the ASP might offer improvements in vehicle safety. First, the ASP would be capable of a powered landing, as opposed to the shuttle's "dead-stick," power-off landing. Second, as an aerodynamic vehicle, the ASP would have more flexible abort options than a ballistic launch vehicle. Third, by its nature as a single-stage-to-orbit vehicle, the ASP would avoid the risks associated with stage integration and separation.

4. Reusability. The ASP would be fully reusable, but again this is a characteristic it shares with many other competing designs for the next-generation launch vehicle.

5. Specialization. The ASP as it is now defined--with civilian and military, air and space applications all intertwined--runs the risk of becoming, like the space shuttle, capable of many things, but not optimal for low cost delivery of payloads to LEO. The factors which drive technology for military applications are very different from those which drive technology for commercial applications. Military

specifications stress performance; on the other hand, commercial developments tend to emphasize lowered production costs, vehicle operating efficiency, and high availability with low maintenance. Thus, an ASP developed first for the military, and then "handed over" for commercial applications may not produce the most cost-effective launch vehicle possible.

Opposition

Space Scientists. No one really opposes the goal of reducing launch costs, but as the ASP program gets into full swing, the nation's scientific elite is likely to question our priorities in space. In a recent issue of Scientific American, James Van Allen reflected the views of many space scientists when he called the proposed space station largely unnecessary, more expensive than described, and a diversion of resources from more meriting projects (Van Allen, 1986). The ASP could easily be a target for similar criticisms from scientists angered by the cancellation or postponement of 17 scientifically important missions over the past four years.

The argument runs as follows. Figure 4-1 shows the annual NASA budget expressed in constant 1982 dollars. For almost a decade, NASA's budget has stabilized at about $6 billion ($ 1982). Thus, Van Allen argues that, "it appears that the US has achieved an approximate equilibrium between advocates and skeptics as to the proper overall level of our national civil space effort." With no reason to expect a sizable real increase or decrease in the NASA budget in the next decade, funding civilian space projects is a zero-sum game: an increase in one area must inevitably cause a cut-back somewhere else. Former Presidential Science Advisor George A. Keyworth II, a staunch advocate of the ASP, concurs, saying that he does not envision increased funding for NASA, despite the ASP program (Covault, 1985). Thus, it appears that Van Allen's hypothesis is correct: funding civilian space activities is a zero-sum game in which the ASP program will channel money away from other aeronautical projects.

Space scientists may view with alarm a NASA budget becoming dominated by the space shuttle, the space station, and now the space plane. In the current fiscal year, the shuttle absorbs approximately half the agency's budget; the station, about 6 percent, but that is expected to increase rapidly. Estimates of the development costs for the station range from $8 to $30 billion in constant 1984 dollars. And while NASA's share of the ASP effort is presently only 20 percent, 20 percent of a $3 billion development effort would be enough to fund 3 major planetary missions.

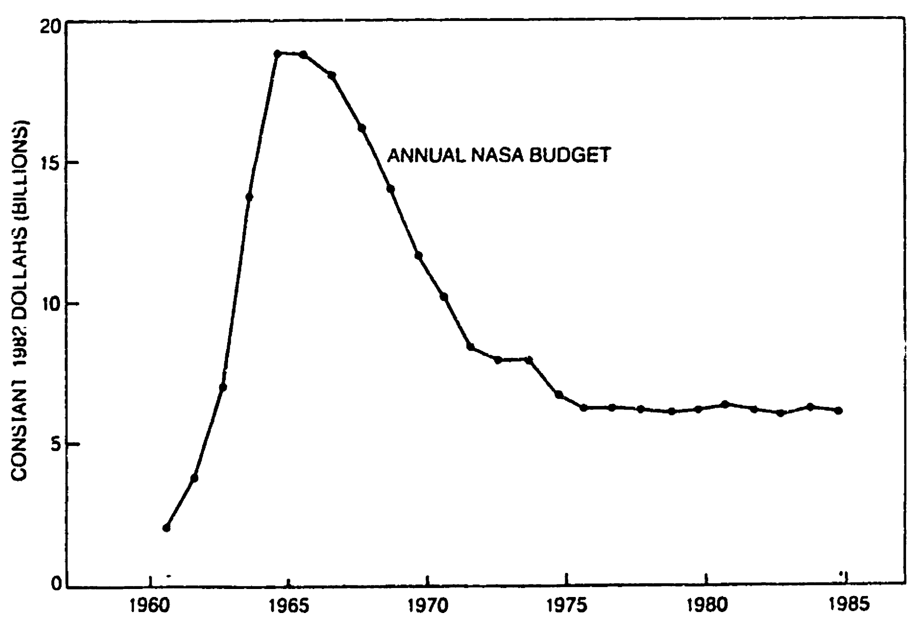


Figure 4-1: Stabilization of the NASA Budget

Support

The Executive Branch. ASP support in the Executive Branch comes from the top. The President's backing of the ASP may be the result of several factors. First, President Reagan is keenly interested in the space program and has twice before endorsed sizable space plans in his annual State of the Union Addresses. In 1983 it was the Strategic Defense Initiative; in 1984, the Space Station. Unfortunately for the President, both these proposals have met with substantial criticism. Some speculate that the President wishes to be remembered for a landmark contribution to the space program, much as President Kennedy is remembered for challenging NASA to put a man on the moon before the 1960s were over. The ASP may have appeared to be a third (and final) opportunity for President Reagan to make a lasting impact on America's future in space.

Second, with the importance President Reagan attaches to SDI, it is likely that he feels it necessary to spend some seed money with the hopes of reducing the cost of access to orbit for SDI components.

A related point is that while the ASP may greatly benefit the SDI program, it can be sold as a civilian aircraft because of the overlapping technologies. In the 1986 State of the Union Address, the President described the ASP as an aircraft that could take off from Dulles Airport and fly to Tokyo in two hours, giving the impression that the vehicle was being developed for civilian purposes. Thus, the promotion of the ASP may also be seen as an attempt to appease a public generally supportive of the space program, but increasingly concerned about the "militarization of space."

The ASP also has some support in the Office of Science and Technology Policy (OSTP), though perhaps this support is being misconstrued. In March 1985, the OSTP Aeronautical Policy Review Committee (APRC) completed its study of current U.S. aeronautical policies and planning. The team recommended that the Administration foster the development of advanced technology for: (1) a new generation of fuel-efficient, affordable subsonic aircraft, (2) a sustained supersonic cruise capability for civilian and military aircraft, and (3) a transatmospheric vehicle capable of routine flight in air and space with take-off and landing from conventional runways. However, the OSTP report placed the greatest emphasis on subsonic technology, saying:

> The goals are time-phased with the first goal (subsonics) being crucial as an enabling goal. It must serve to reenergize American R&D momentum, because the product technologies it fosters must generate private sector resources necessary for exploitation of ensuing opportunities in supersonic and transatmospheric flight ... Emphasis in the near-term years must necessarily be focused on the first goal.

Thus, the OSTP report can be interpreted in different ways. ASP supporters can read the OSTP report as saying that the ASP is one of our "top three aeronautical goals." Another way of looking at OSTP's recommendations is that hypersonics is "our last priority," because after subsonic and supersonic research is revitalized, what remains but hypersonics? In fact, the chairman of the APRC and former vice-president of Boeing, John Steiner, has said that one of the "disappointments of my life" is that officials have "jumped on" goal number three, at the expense of the more close at hand technology of goal number two, a supersonic transport (Marshall, 1986).

The National Commission on Space. In July 1984, Congress approved the Space Act, Public Law 98-361, which created a National Commission on Space (NCOS) to assist Congress in articulating long-range goals and policy options for the US civilian space program to the year 2035.

The April 1986, NCOS report, Pioneering the Space Frontier, Our Next 50 Years in Space, strongly supported lowering launch costs to LEO, and recommended that the ASP be considered as a means to accomplish this. The report characterized the shuttle system as "...a complex, somewhat fragile vehicle, both expensive and technically demanding to operate (Mann, 1986a)." Accordingly, the commission felt that the two most important contributions to space development the government could make were (1) ensuring continuity of launch services, and (2) drastically reducing space transportation costs.

> Cheaper, more reliable means for transporting both people and cargo to and from orbit must be achieved in the next 20 years. While all space programs would benefit from lower cost orbital transportation, it is especially important that the cost be dramatically reduced for free enterprise to flourish with commercialization of space operations. The Commission is confident that the cost of transportation can and should be reduced below $200 per pound (in 1986 dollars) by the year 2000 (Mann, 1986a).

The Commission felt that the follow-on to the shuttle should either be an air-breathing ASP, or an advanced, fully-reusable rocket vehicle. The menu of technologies singled out for intensive research over the next five years would support both vehicles. Among these key technologies are computational fluid dynamics, dual-fuel rocket propulsion, scramjets, high-performance materials, aerodynamics, thermal shielding, and launch automation. The NCOS proposed that intensive technology demonstration efforts be undertaken leading to a vehicle selection by 1992. The Commission's report also urged that such vehicles be truned over to the commercial sector for operation as soon as possible: "The sooner the private sector can assume responsibility...the sooner the US can begin to pattern Earth-orbit transportation after commercial airline operations."

Department of Defense. The DOD has two main incentives to support developing the ASP as a launch vehicle. The first is that in a normal, non-crisis situation, the ASP would offer the military the same basic advantage it offers commercial customers, namely reduced costs to LEO. Furthermore, the need for reduced launch costs will become more acute since spacecraft survivability is a growing concern for the DOD, and designing for survivability means heavier spacecraft.

Second, the DOD has reason to support the ASP because of its horizontal take-off and flexible basing. The air-breathing engines on the ASP would produce a much lower temperature, more diffuse exhaust stream than an equivalent rocket system would. This is due primarily to the lower pressure combustion conditions and the presence of nitrogen as a dilutant in the air (20 percent oxygen, versus 100 percent oxygen usage in rockets). Consequently, air-breathing engine exhausts have a lower noise-generation potential than those of rockets. This less intense, comparatively low-noise exhaust may permit the ASP to operate on conventional runways at conventional airports, providing facilities for cryogenic fuel production or storage were in place.

The military seeks the ability to take off and land from conventional airports because the most minimal attack on the US (even sabotage) could be sufficient to wipe out our entire space launch facilities, since they are fixed targets. The ASP offers the potential to increase the survivability of our spacecraft launch capacity through dispersed inland basing. Survivability is enhanced the further inland the ASP is based because it allows the ASP more time to get aloft once an incoming threat is detected. Coastally-based ASPs could be moved inland in times of crisis.

Strategic Defense Initiative Organization. Officials in the SDIO readily admit that their organization is a leading figure in the push for the ASP. The projected development and production costs of SDI infrastructure alone are alarmingly high; with launch costs at today's rates added in, the figure becomes colossal. The current baseline SDI architecture calls for lifting 26 million kilograms into orbit at a cost today of about $130 billion for transportation alone (Mann, 1986a). SDIO officials have openly stated the need to reduce launch costs by "an order of magnitude" for SDI deployment to become plausible (Ulsamer, 1985).

According to John Pike of the Federation of American Scientists, this requirement became even more acute this fall after the completion of an SDI architecture study which showed that 100 large space platforms would be far less effective than 1000 small ones (Marshall, 1986).

Although much of the push for the ASP within the DOD comes from the SDIO, they are not a major funder because they see the ASP as being too long-term for their priorities, which are on near-term technology demonstrations (Pike, 1986).

NASA. NASA is in a delicate situation with respect to the ASP. Although NASA engineers are excited about the challenge of developing such a plane, NASA officials fear that too much talk about the ASP would give the false impression that they were giving up on the shuttle (Wilford, 1986). When the shuttle program gets back on track, NASA will become less reticent about discussing plans for the future, and will be free to promote the ASP. Since it is Reagan Administration policy to step up the ASP effort, NASA will support the idea publicly. However, NASA is not monolithic. The internal debate as to what is the best option for a next-generation launch vehicle is not yet over. An analysis of alternative launch vehicles was not possible within the scope of this thesis, though the following two citations illustrate that within NASA, the ASP is not unanimously viewed as the most attractive launch vehicle choice.

Ivan Bekey, NASA's Director of Advanced Programs, has said that his analyses indicate that an equally promising, but less promoted, launch vehicle configuration appears to be a vertically-launched rocket employing dual-fuel engines that burn kerosine in the first part of the atmosphere and then switch to hydrogen fuel (Bekey, 1986). Like the ASP, the dual-fuel rocket would need advanced, lightweight structural materials as well as a vastly simplified launch environment.

Charles Eldred is the Assistant Head of the Vehicle Analysis Branch of the Space Systems Division at NASA-Langley. Eldred has evaluated both "conventional" (vertical take-off, all rocket propulsion) and "unconventional" (horizontal take-off, air-breathing propulsion) launch vehicle concepts and concluded as recently as June 1984 that:

> Although many of these unconventional concepts [the ASP] could be feasible in the desired time period, the conventional concepts [advanced rockets] will be smaller in size, lower in dry mass, less complex in design and operations, less costly, and lower in technological and programmatic risks (Eldred, 1984).

In view of the upper-level positions held by Messrs. Bekey and Eldred, this dissention does not appear to be merely a turf battle between rocket engine specialists and those who specialize in air-breathing engines.

Of course, Bekey and Eldred's vertically-launched rocket would not have the flexible basing and autonomy the USAF desires, and, more to the point, lacks the "political sexiness" the ASP has: "Seen one rocket, seen 'em all. But an airplane that flys into orbit? Now there's something interesting." Thus, there may be conflicting interests within NASA regarding the ASP. Everyone favors reducing launch costs to LEO, and while the ASP appears to be a step in that direction, the dual-fuel rocket may be superior. On the other hand, the ASP arouses interest in almost everyone who hears about it and may be an easier concept to sell. In the conceptual development stage, NASA-the-space-agency can be objective about the relative merits of various concepts. However, if the ASP program becomes a national priority, NASA-the-bureaucracy can be expected to be less critical of the ASP and more dependent on continuing ASP funding.

In a certain defensive respect, NASA can ill-afford the risk of not going ahead with the ASP. Keeping in mind the British Aerospace analysis that said that a HOTOL-sized vehicle could accomodate 90 percent of the world's to-LEO traffic, it is possible to imagine a scenario where the British forge ahead with HOTOL and are successful with it, while the United States does nothing. Though it is not clear that the British can develop such a vehicle, the possibility exists.

In the short-term, NASA has several more immediate concerns that impair its ability to champion the ASP. In addition to re-evaluating the space shuttle system and fighting to keep funding levels up for the space station, NASA officials may deliberately wish to keep a low profile until NASA's leadership is restored and any organizational shake-ups brought on by the Challenger investigation are completed.

Commercial Users. Commercial satellite owners obviously have a pecuniary interest in the potential for launch cost reduction that the ASP offers. Also, there is a commercial advantage in flexible basing. Dispersing ASPs near the areas of the country where spacecraft are constructed (Los Angeles, Denver, Philadelphia, Washington D.C., etc) could reduce the travel time, expenses, and risks of damage associated with transporting fragile spacecraft to distant launch sites.

4.3 The Aerospace Plane as a Hypersonic Transport

Concept. A strawman hypersonic transport (HST) configuration has been developed by government and industry to explore the feasibility of applying ATRs and scramjets to civilian aircraft. The expressed intent is to improve the efficiency of air transportation to nations in the Pacific Rim, especially; hence the nick-name, "Orient Express." The "Orient Express," as currently envisioned, would be a 300 to 500 passenger, Mach 5 transport aircraft able to fly from New York to Beijing in about two hours (United States Congress, Hearings on High Speed Aeronautics, 1985, p. 168). The vehicle would be powered by an ATR engine fueled by cryogenic methane and would benefit from much of the technology development planned for the ASP.

The logical place to begin an appraisal of the merits of an HST is with an examination of our experience with its stillborn cousin, the Supersonic Transport (SST). In 1971, Congress killed the SST program, citing the following reasons: (1) the dubious prospects of the SST being a commercial success, (2) airport noise and sonic boom problems, (3) possible negative environmental effects on ozone and climate, and (4) the impropriety of devoting large sums of public money to a project that would benefit only a small percentage of the people (Bugos, 1980, p. 74).

These same questions will inevitably come up for the HST. It is beyond the scope of this thesis to determine what impact the last three concerns would have on an HST program. Clearly, however, Congress' fears about the profitability of an SST have been borne out by the

Anglo-French Concorde, which (though a technological achievement) has been a financial disaster. Lenders are even more skeptical and will insist on a demonstrably economic aircraft before putting up any private money for development or aquisition. A necessary prerequisite, then, for a viable HST program would be convincing, favorable economic projections supported by both government and the aerospace industry.

Opposition

Industry. Many industry officials, however, see the HST as strictly a military program and feel it is a mistake to pin any commercial hopes on it. John Steiner, who spent the last 22 years as the vice-president of Boeing and recently served as chairman of the White House Aeronautical Policy Review Committee, asserts, "This whole idea of a hypersonic airplane is good from a military standpoint, but is being way overplayed as an 'Orient Express' (Marshall, 1986)." Steiner says he believes there is a market for rapid travel to the Pacific Rim region, but favors the more close-at-hand supersonic transport technology because airlines would be willing to invest in it. At a time when many airlines are strapped for cash, struggling just to stay out of the red, talk about developing an entirely new fuel system based on either hydrogen or methane is "baloney" in Steiner's opinion.

Evidently there are those on the other side of the Atlantic who agree with Steiner. France's Aerospatiale has been busy designing a second-generation supersonic transport as a follow-on to Europe's

Concorde, and the company is offering to work with U.S. industry in a cooperative development of such an aircraft. One Aerospatiale official remarked:

> We see a resurgence of U.S. interest in high-speed transports, but we can't understand why there's so much talk now about such far out ideas as Mach 3-4 or even Mach 8 aircraft. Does this mean airlines are going to qualify passengers for space flight in order to carry them in these technological marvels? And have they forgotten that the flight time difference between Concorde's Mach 2 and a Mach 3 aircraft over the Paris-New York segment is less than 15 minutes--and that a Mach 3 transport is going to be much more complex to develop than the Mach 2 transport? We French often are accused of going off with our head in the clouds on new ideas, but I think in this case it is we that have our feet on the ground while our American friends are doing the daydreaming (Lenorovitz, 1986).

This raises an important point. While it is possible that the HST could become an economically viable successor to the Concorde, the HST would almost certainly be too expensive and too risky to develop in the face of a competing second-generation SST from France. The project, therefore, lends itself to international cooperation, a stated goal of U.S. civilian aerospace policy. Yet, because much of the technologies are similar for the HST and the highly classified ASP, international cooperation on the HST appears a long way off.

Support

Industry. Regardless of what aerospace industry officials think of the merits of the HST in private, publicly they can not afford to criticize the plan, just in case the government decides to go ahead with the idea. This conflict has produced some amusing arguments in favor of the HST. For example, the only justification for the HST that a major airframe contractor offered to the House Committee on Science and Technology was the following:

> Why high speed? Commercially the need for high speed transport increases as distances extend. Studies have shown that 3 hours is the nominal comfort limit of passengers. To extend beyond the 3 hour limit, services such as meals and entertainment must be provided to distract passengers. These services cost fixed weight, add crew members, and are expensive in themselves. Over long distances such as the Pacific Basin, the advantage of speed becomes very obvious (United States Congress, Hearing on <u>High Speed Aeronautics</u>, 1985, p. 164).

It strains the imagination to think that this firm honestly believes that the cost and weight of sandwiches, soda, and video-taped movies justifies a multi-billion dollar development project. The truth is that currently there are just too many uncertainties (technical uncertainties as well as financial, political, and sociological ones) for anyone to make a compelling case for the HST, and until the case is made, private industry will not undertake the expensive R&D initiatives on its own. It seems that with vague statements about the "obvious advantages of increased speed," the major airframe contractors are "testing the waters" to see if the government is seriously interested in investing in the HST.

4.4 The Aerospace Plane as a Military Vehicle

<u>Concept</u>. It is not clear which potential military ASP applications are being given serious consideration within the DOD. There has been speculation on military missions for hypersonic aircraft for over a decade (Hearth and Preyss, 1976), (Jones and Huber, 1978). Yet as recently as 1983, Dr. Robert S. Cooper, former DARPA Director, testified

before Congress that there were no military requirements for hypersonic flight per se: "...currently Defense has no mission for a hypersonic aircraft...But, in my view, that shouldn't deter us from exploring that flight regime (United States Congress, Hearing on The Future of Aeronautics, 1983, p. 31)."

Even retired Brigadier General Charles E. Yeager, when pressed, admitted that while the military is interested in researching hypersonic aircraft, it is not quite sure what it would do with such an aircraft if it had one.

Mr. Carney: You mentioned planes that fly...from Mach 7 to Mach 20. Are they aircraft that you think would be wanted by the military? Is there a mission to have such a high-speed aircraft?

General Yeager: In my opinion...there has to be a reason to develop a system...I think that would be an area that we could look at. It's an expensive area, but it is an area that research can be done.

Mr. Carney: But again, you feel there would be a mission for those hypersonic--

General Yeager: It's a research and development mission, yes, sir.

Mr. Carney: ...you look at it as a purely R&D mission and not one where you could take a plane, let's say, that had the capability of Mach 10, and...put it into the Air Force inventory and assign it a mission?

General Yeager: Basically (United States Congress, Hearing on The Future of Aeronautics, 1983, p. 8).

NASA was also unable to substantiate any operational need for hypersonic aircraft. Dr. Hans Mark, Deputy Administrator of NASA, stated that an ASP might be useful in connection with the President's SDI proposal, but qualified his remark with (United States Congress, Hearing on The Future of Aeronautics, 1983, p. 65):

Okay, that's a personal speculation...but I want to hasten to

> add...that we are not at the stage in the hypersonic area where there are any missions that one can hang a program on. We are...in a research phase. I thoroughly agree with General Yeager there when you asked him the question whether there were missions for hypersonic vehicles, the answer right now is no. I am speculating that there will be [missions] and that they will be related to the problem of strategic defense because they will fly in precisely the region of the atmosphere where that conflict will go on.

The DOD's reluctance (or inability) to specify what missions they believe are worthwhile for an ASP makes the task of analyzing military applications difficult. Figure 4-2, a DARPA illustration, suggests that the DOD's evaluation ranges from tactical to strategic to space missions. A survey of past arguments for development of hypersonic aircraft for military purposes revealed the following five applications to be most often suggested: tactical air defense, strategic reconnaissance, strategic offense, strategic defense, and space combat.

Mission 1: Tactical Air Defense. Air defense of civilian and military installations requires aircraft that can take off quickly, and locate and destroy enemy cruise missiles and aircraft with long range missiles or cannon fire. Dr. Raymond S. Colladay, NASA Associate Administrator for Aeronautics and Space Technology, has suggested that the ASP could be configured as a long-range air defense interceptor based on its rapid response and short flight times (Defense Daily, 1985). S. A. Tremaine, Deputy for Development Planning at the Aeronautical Systems Division (ASD) of the Air Force Systems Command, has said that the ASP represents "a major shift" in the history of fighter aircraft (Marsh, 1983).

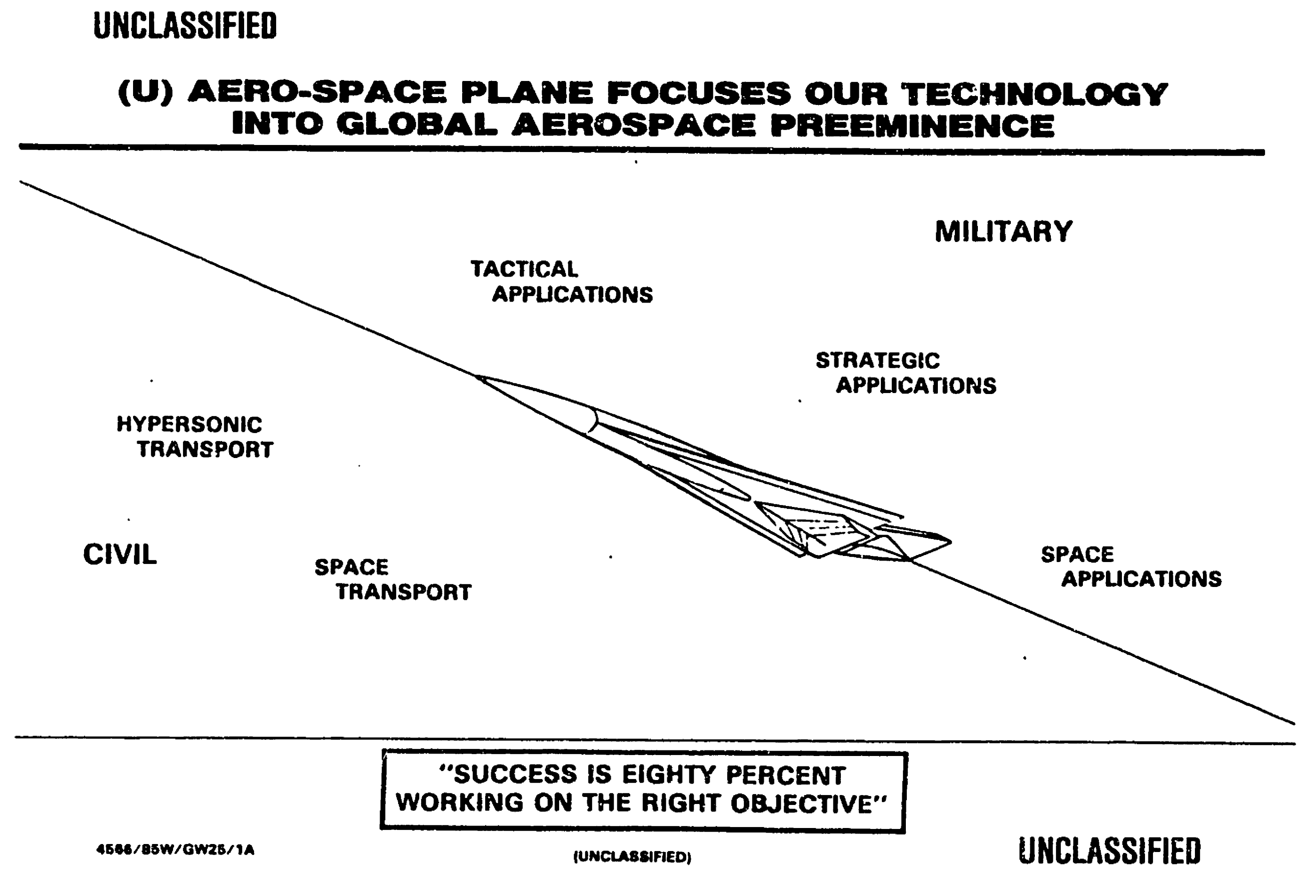


Figure 4-2: The Wide Range of Applications Being Considered for the Aerospace Plane

Figure 4-3, another DARPA concept drawing, shows an ASP intercepting a Soviet bomber within thirty minutes of detection, while it is still over the North Sea (United States Congress, Hearing on High Speed Aeronautics, 1985, p. 60). This would imply an average ASP speed of about Mach 12 over the distance, assuming a 5 minute reaction time. If an ASP can be built at all, there does not appear to be any fundamental technical reason why it could not be equipped with long-range missiles to perform such tactical air defense missions, though speeds in excess of Mach 12 may be overly optimistic because of the effects of engine inefficiencies described in chapter 2.

On the other hand, the ASP does not appear to be the most cost-effective air defense system. In the previous chapter, the ASP as a launch vehicle was estimated to cost in the vicinity of $1 billion per vehicle. The ASP as a tactically-equipped, hypersonic fighter plane would certainly cost less than that, but compared to F-15s at approximately $27 million per aircraft (1985 dollars), it seems unlikely that the ASP could perform this role as cost-effectively. Tremaine himself added that affordability of the ASP is a key issue. For example, if the advanced tactical fighter, also under study by ASD, ended up costing $100 million a copy, the USAF would not build it (Marsh, 1983).

Furthermore, the ASP's high speed does not seem to be as much of an advantage considering that: (1) F-15s can be forward-based at air bases in Great Britain, West Germany, Greenland, Spain, and the Azores, and (2) as Dr. Robert S. Cooper, former DARPA Director, has said:

> As you recall from General Yeager's testimony, aircraft for fighter aircraft purposes in a regime beyond about Mach 2 really have

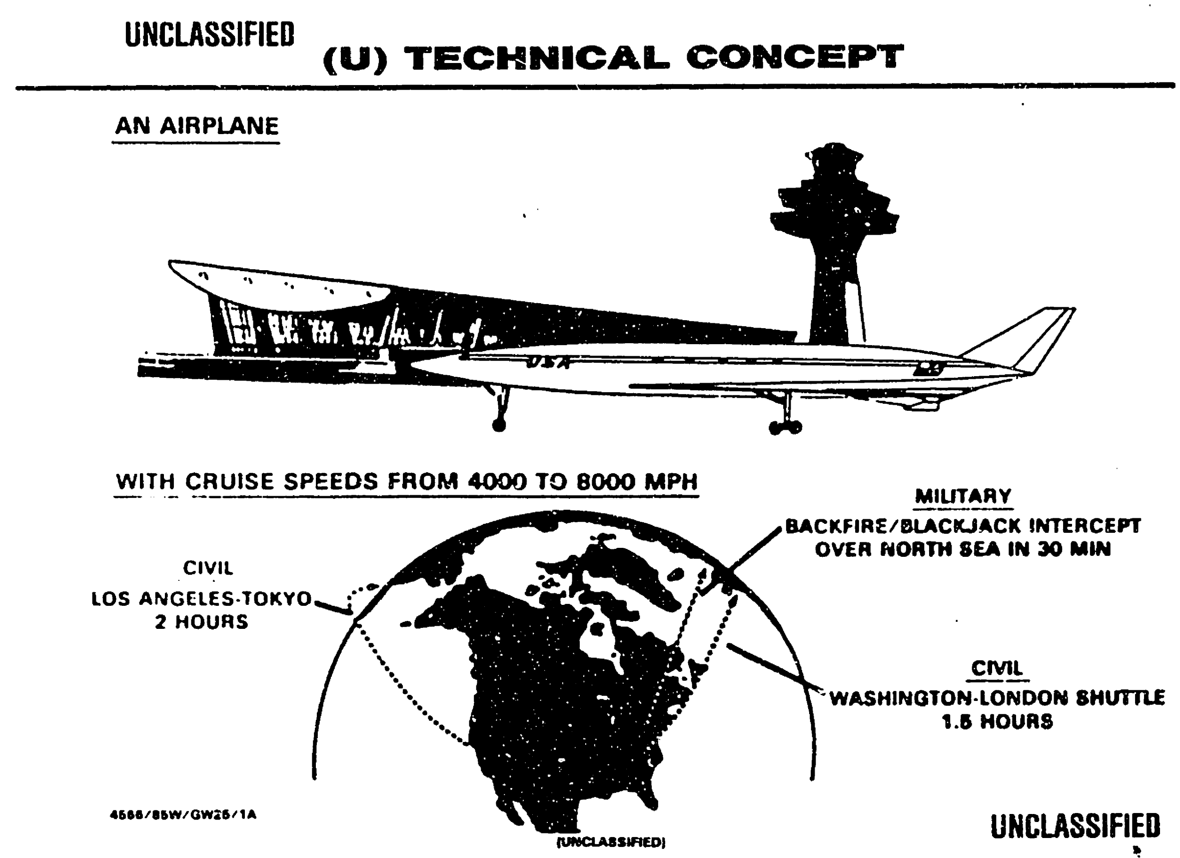


Figure 4-3: The Aerospace Plane as a Tactical Aircraft?

little or no utility. Aircraft that engage one another at whatever speed ultimately end up in their final throes of engagement with one another in subsonic regimes. And so, supersonic aircraft beyond about Mach 2 are probably of little or no utility for the military. That conclusion is further enhanced by the rapid rate at which missiles are geting smarter; that is, missiles can be used in the regimes beyond Mach 2 where that is required for terminal engagements (United States Congress, Hearing on The Future of Aeronautics, 1983, p. 81)."

Mission 2: Strategic Reconnaissance. Tremaine has said that the most promising use of the ASP at present appears to be reconnaissance (Marsh, 1983). Military reconnaissance employs both sophisticated satellite systems (KH-11 and Big Bird) and high-altitude, high-speed aircraft (primarily the SR-71). Photo-reconnaissance satellites are said to provide the best ground resolution, but follow highly predictable, low perigee orbits which makes them potentially vulnerable to attack by future anti-satellite weapons (ASATs).

The SR-71 has less ground resolution, but is able to maneuver and can map with its cameras over 100,000 square miles in less than an hour (Gervasi, 1977, p. 119). Able to cruise at a speed of about Mach 3.3 at 86,000 feet, no SR-71 has ever been shot down, though eventually it may be vulnerable to a future Soviet high velocity interceptor missile. With a flight envelope far beyond that of the SR-71, an ASP configured as a reconnaissance aircraft could improve the survivability of our aerial reconnaissance system, though it must be emphasized that currently no threat exists. Designing this vehicle would be a challenge since integrating a camera system into the vehicle's fuselage would further complicate already formidable thermal protection and aerodynamic problems.

Mission 3: Strategic Offense. Another often-suggested military use of an ASP is as a means to deliver nuclear weapons. It appears to offer the prospect of coupling characteristics of bombers with the short flight times of ICBMs. Like bombers, an ASP would be recallable, providing a "fail-safe" mechanism for limiting the escalation of a nuclear war. Also, as with bombers, an ASP could place its re-entry vehicles (RVs) in a re-entry corridor much closer to, if not over, the enemy's territory, resulting in increased warhead accuracy as compared with ICBMs.

But like ICBMs, a bomber ASP would follow an exoatmospheric trajectory. In the absence of a Soviet ASP-style hypersonic interceptor aircraft, an American ASP could operate in this flight regime with virtual impunity, concentrating on weapons delivery without the fear of being intercepted. The ASP would still have to be EMP-hardened so that high-altitude nuclear explosions would not upset the ASP's avionics. Thus, as with the strategic reconnaissance mission, the advantage offered by the ASP's potentially high speed and wide operating envelope is that they could increase the probability that the vehicle would penetrate Soviet air defenses during war.

A technical challenge facing an ASP used in this capacity would be to achieve a 15 minute response time from non-alert status to ASP take-off in order to insure that a Soviet first strike could not destroy ASPs on the ground. Strategic bombers achieve this level of readiness, but do not face the fuel storage problems that cryogenic hydrogen creates.

Furthermore, the question of cost-effectiveness arises again. An MX missile, carrying ten warheads, costs in the vicinity of $50 million to $100 million--5 to 10 percent of the cost I projected for the ASP as a launch vehicle. Assuming an RV weight of about 300 kg, the ASP reference payload of 10,000 kg could accommodate approximately thirty RVs--the equivalent throw-weight of only three MX missiles.

Mission 4: Strategic Defense. As mentioned above, Dr. Hans Mark, NASA Deputy Administrator, has suggested that the ASP would be useful for missions involving the interception of ballistic missiles in flight (United States Congress, Hearing on The Future of Aeronautics, 1983, p. 51. See also, Hearth and Preyss, 1976, p. 24). Mark testified,

> I believe we will do hypersonic flight ... I believe that a requirement will emerge shortly. Last March the President made a remarkable speech about ... creating a defense against nuclear ballistic missiles ... built on the boundary line between space and the atmosphere. And it is my considered opinion that at some point or other we will require either manned or unmanned vehicles that fly in a sustained way in the upper atmosphere with ... propulsion ... in the hypersonic ... range. I believe that will happen.

There are two main problems with this idea. First, with ICBM flight times of 30 minutes from time of launch detection to warhead explosion, the ASP would either (1) not be able to get to orbit in time to intercept the RVs before they entered the atmosphere, or (2) would need an almost instantaneous reaction time and acceleration levels in excess of that which a human could tolerate. Second, even if an ASP with suitable weapons were in orbit at the time of attack, the RVs would be dispersed over such an expanse of space that very few could be targeted and intercepted during their 20 minute coast phase.

Mission 5: Space Combat. Finally, it has been suggested that the ASP be used in a space combat role, with space-to-space fighting capabilities. Space-to-space missions may be either defensive, protecting our space systems from attack by Soviet ASATs, or offensive, destroying Soviet space assets.

The defensive role does not appear to be one the ASP could perform for the following reason: the ASP would be manned while an ASAT weapon is unmanned. Thus, an ASAT can travel faster than the ASP, which is limited by the acceleration levels its pilots can withstand. The Soviet's current generation of ASAT, the Satellite Interceptor System (SIS), takes three hours to rendez-vous with its target. If the US proceeds with a military ASP in order to intercept Soviet ASATs before they reach US satellites, the Soviets can respond by developing a faster ASAT (note that the US ASAT takes only a few minutes to destroy its target).

In an offensive context, the ASP would be essentially just another ASAT system. But Paul Czysz, McDonnell Douglas' ASP program manager, thinks that the ASP as a military spaceplane could, "...avoid annihilating things--[it could] simply cause them not to function (Canan, 1984)." It is claimed that this capability might offer the military the option of neutralizing selected enemy space assets without being detected. Czysz offers two examples: an ASP might fire needle-like projectiles into an enemy tracking and fire control radar in order to overwhelm the antennae, or low-intensity lasers might be used to "blind" satellite sensors.

However, in a military context, the capability to make a satellite "cease to function" does not appear to offer any advantages over the capability to simply destroy it. Satellite "sabotage," as described by Czysz, would not be any more covert than the F-15 launch of an MHV ASAT, since Soviet space-tracking radar would easily be able to detect the presence of an ASP in the vicinity of a Soviet satellite before it was disabled.

Furthermore, it must be noted that such sabotage would not only be readily detected, but it would be of marginal military value. The Soviets typically operate over 60 satellites in LEO; the US operates less than 20 in this regime. The 10 Transit type navigation satellites and the almost 30 tactical C3 satellites maintained by the Soviets present a formidable target list. The loss of several of either type of spacecraft would be met by only a graceful degradation of capability, not a catastrophic loss of service (Johnson, 1985).

And finally, merely possessing the capability to perform such missions could be destabilizing, especially in a crisis, since a normal satellite anomaly might then be misconstrued as a pre-emptive "black-out" attack. The enemy might not be sure whether the satellite malfunctioned on its own, or whether it was attacked.

Opposition

Congress. Because military officials stress the research nature of the ASP project, criticism has not materialized yet. When the research phase ends in 1988, Congress will face the question of whether to fund construction of a test vehicle. At that point, the military will have to be more open about what they intend to do with an ASP. By that time, Congress may ask the Office of Technology Assessment (OTA) for an examination of the usefulness of hypersonic aircraft for civilian and military purposes. OTA would likely recommend that Congress consider questions such as those raised in this analysis.

The first question raised is, "would the ASP be cost-effective in role X?" This relates to the "quantity-versus-quality" debate. With a fixed budget, one may buy a large number of comparatively inferior aircraft, or buy fewer, but more capable (and more expensive) aircraft. In the face of Soviet willingness to produce large numbers of aircraft, the USAF has traditionally opted for quality, seeking to compensate for probable numerical inferiority with planes that are more capable, one on one. A military ASP may appear to be too costly compared to existing tactical and strategic aircraft.

In addition to cost-effectiveness, Congress often considers the implications of weapon systems for arms control and foreign policy objectives. Examining these issues raises questions like, "would the ASP be stabilizing or de-stabilizing?" I am not arguing this point one way or the other; the effects on stability depend on how the ASP is used.

Though it may be argued that the increased survivability of an ASP bomber would add to deterrence, there are also rather ominous implications of expanding our capacity to destroy space-based targets that may weigh in Congressional deliberations.

NASA. If development of the ASP proceeds with primarily military applications in mind, the tenuous partnership between NASA and the DOD may become unglued. The NASA/DOD coalition was strongest in the early 1970s when both felt they had something to gain from shuttle development, and each knew the other's support was essential to obtain the funds. Recently, the relationship has been strained by issues such as the pricing of military shuttle launches, the military push for complementary ELV development, and the military's refusal to back the space station. Furthermore, as the military space budget grows, NASA increasingly finds itself in competition with the DOD for funding. Figure 4-4 shows that the DOD space budget was on par with the civilian budget in 1981, but has since grown to double the civilian budget. In the context of this rivalry, it is undoubtedly the cause of some frustration at NASA that the hypersonic technology developed at NASA-Langley with discretionary funds is now being managed by the Air Force.

Support

NASA. On the other hand, NASA may decide it is in its best interests to go along with a military-ASP push. With the military space budget outpacing the civilian space budget, NASA may become co-opted by the DOD, and the ASP may represent a turning point in NASA's history as an independent civilian space agency.

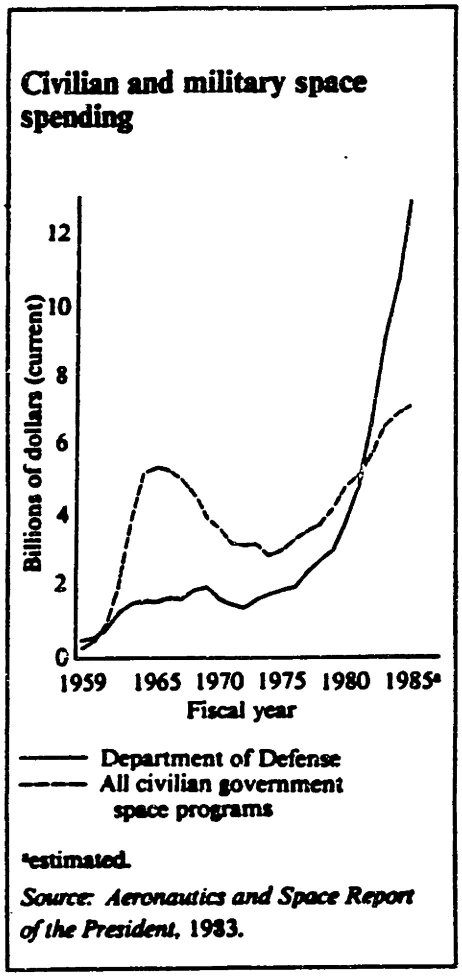


Figure 4-4: Shifting Priorities in Space Spending

<u>The Air Force</u>. As shown earlier, the ASP may have a valid military application as a strategic reconnaissance aircraft. But to the USAF, it is more than just a matter of military requirements; it is also a matter of prestige. The ASP represents the final step towards an independent Air Force manned capacity in space. The USAF has long sought to loosen NASA's monopoly on manned space flight. Now the Challenger disaster and the recent ELV failures strengthen the USAF's hand in arguing for an independent space flight capacity. For years the USAF maintained that total reliance on the shuttle for military launchings could prove to be

extremely risky in case of a shuttle accident.

The USAF almost had an independent space research laboratory twenty years ago. In the late 1960s, the USAF had to scrap plans for a Manned Orbiting Laboratory (MOL) intended to explore man's ability to perform military functions in space, because they appeared to duplicate NASA's Manned Orbital Research Laboratory (MORL) efforts. Some money was spent to develop the MOL launch facility at Vandenberg Air Force Base (VAFB) in California. Recently, the USAF has paid $3 billion to modify these facilities to accommodate military space shuttle missions. Still, the USAF is forced to rely heavily on NASA for management, equipment, and support of its space operations. Until the new Consolidated Space Operations Center (CSOC) outside Colorado Springs becomes operational, the USAF has to run its shuttle missions out of NASA's Manned Space Flight Center in Houston.

The importance the USAF attaches to the ASP is indicated by the fact that it was chosen as one of the highest priorities among 70 emerging technologies in the Project Forecast 2 last August, following a similar study of high-leverage technologies initiated in 1963. The goal of the Project Forecast is "...to avoid the evolutionary approach to R&D and avoid being driven by requirements or constrained by budget considerations in identifying technologies that would have a payoff in the year 2000 (North, 1986)."

Navy. The Navy has been active in hypersonic research since its beginnings in 1958. Their interest has been primarily in scramjet-powered, high velocity missiles for fleet defense. Now with the aggressive ASP effort, the Navy may also be exploring the feasibility of carrier-based, hypersonic interceptor aircraft. The Navy's involvement does not mean as much in terms of funding or management support as it does in terms of creating a majority with the USAF. With two of the three main armed services backing the ASP, the program can probably gain the approval the Joint Chiefs of Staff (Keller, 1986).

Reconnaissance Community. The reconnaissance community includes the National Reconnaissance Office (NRO) and the intelligence agencies it supports. Ray L. Chase, a consultant with Analytic Services Inc. on military applications for the ASP, has said that the reconnaissance community favors a reconnaissance version of the ASP because their needs would be better served by Mach 20 flights at 45 miles high than by Mach 3 flights at 18 miles (Marsh, 1983).

Congress. Just as some members of Congress will probably oppose the ASP on grounds of high cost or lack of a clearly defined mission, there are also those who will champion the ASP. Among the chief space advocates is Representative Ken Kramer of Colorado Springs, who introduced a bill in 1981 to rename the Air Force the "United States Aerospace Force." The bill would have required the Aerospace Force "...to be trained and equipped for prompt and sustained offensive and defensive operations in air and space...for the preservation of free access to space for US spacecraft." The Aerospace Plane will certainly have great appeal for Kramer and those who share his enthusiasm about space.

CHAPTER 5: SUMMARY, CONCLUSIONS, AND RECOMMENDATIONS

5.1 Summary

Chapter 1: ASP Concept and Motivation. Chapter 1 discussed the history and technologies of two space transportation systems: the space shuttle and the proposed aerospace plane (ASP). The evolution of launch costs to LEO was traced and showed that we have reached a plateau in our efforts to reduce launch costs despite the fact that space transportation costs are high, both in absolute terms, and relative to air transport.

An analysis of the space shuttle's development costs showed that the actual shuttle development costs were about 25 percent higher than the official initial estimates, highlighting the difficulty of making such projections. An examination of shuttle operations costs revealed that even though 3/4 of the shuttle's mass at lift-off is propellant (5/6 of that is liquid oxygen), propellant costs are negligible compared with the highest cost of all, manpower. Launch and flight manpower together account for nearly half of the shuttle's $108 million launch cost. The high manpower requirements were found to result from (1) technical deficiencies in the shuttle's thermal protection system (TPS) and main engines, and (2) labor-intensive flight planning, payload check-out and loading, and vehicle maintenance and repair operations. A list of "lessons learned" from the shuttle experience identified the following factors as vital to reducing launch costs: (1) reduced manpower requirements through advanced automation and intelligent avionics systems, (2) a higher launch rate, (3) excellent technical performance, (4) full reusability, and (5) placement of the highest priority on reducing launch costs.

Next, the proposed ASP's technology and history were described. Its most salient characteristic would be its use of air-breathing engines to reduce (but not eliminate) the vehicle's on-board liquid oxygen requirements, resulting in: a horizontal take-off and landing capability, and most importantly, reduced operational costs. Officials differ as to the extent of the cost reduction, but factors of ten to one hundred are commonly cited. Development of advanced propulsion systems such as air turbo-ramjets (ATRs) and supersonic combustion ramjets (scramjets) is seen as crucial to the ASP's technical feasibility. Finally, while the ASP concept has aroused interest in the aerospace community for at least twenty years, only recently has it been considered both technically and politically possible to develop such a vehicle.

Chapter 2: ASP Systems Analysis. The first two sections of chapter 2 analyzed the performance of the ASP propulsion system and the thermal protection system (TPS). Both analyses indicated that the margin for error is slim when designing for hypersonic speed. At high Mach number flight, small unexpected inefficiencies in the inlet, engine, or nozzle could reduce the exhaust velocity to the point where zero net thrust is produced. In the case of the TPS, the feasibility of cooling the ASP through radiative and regenerative (using the cryogenic fuel as a heat sink) cooling was demonstrated at intermediate Mach numbers, around 6.

As the Mach number increases, it was shown that the thermal protection problem becomes very acute, with a fine line between the available heat load and the heat sink.

The second half of the chapter concerned the overall vehicle performance and mass ratios. Equations for the ASP's propellant mass ratios were determined using space systems engineering techniques. The masses of the various ASP components were expressed as linear functions of each other based on a survey of the technical literature on mass estimation. The solution algorithm required an iterative solution of a ten-by-ten matrix for which a computer program was written. Additionally, a sensitivity analysis was performed on the mass estimation parameters.

The most important result of this chapter concerned the effect that variations in the maximum scramjet operating velocity has on the total ASP mass. If scramjets function up Mach 21, the ASP gross lift-off weight (GLOW) is about 1/6 that of the shuttle; at Mach 12, the ASP GLOW is 50 percent greater than the shuttle's. The sharp increase in ASP GLOW at lower transition Mach numbers is due to the extension of the less efficient rocket propulsion phase over a wider range, meaning that more total fuel is required. This indicates the "unforgiving" nature of scramjet technology. At low maximum scramjet operating Mach numbers, scramjet performance falls off precipitously, in contrast with rocket engine performance, which degrades more gradually.

Chapter 3: ASP Costs. ASP development, fabrication, and operations costs were estimated using the Transcost space transportation system cost-estimating model, modified to account for the ASP's air-breathing engines. The Transcost cost-estimating relationships (CERs) were statistically derived from component costs of past US and European launch vehicles, as well as from advanced launch vehicle paper studies. The fabrication costs for the air-breathing engines were estimated using a CER developed for the FAA in its SST analyses. The costs differed for different values of the maximum scramjet operating velocity, but were in the vicinity of the values given below.

The most important conclusion of this chapter was that the official government estimate of $3 billion for ASP development is probably an underestimate. My cost model produced an estimate of $17 billion for ASP development, and a comparison with other aerospace vehicle development costs supported the position that ASP development costs would almost certainly exceed $10 billion.

An ASP fleet size of 10 was found necessary to meet the official government projected demand for launch services in the time period 2000-2010. The fabrication cost of an ASP was estimated to be approximately $1 billion, taking into account the effects of learning during production.

The operations cost sub-model is admittedly in a preliminary stage of development because of the lack of an operations cost data base. The model attempts to account for costs due to technical systems management, prelaunch operations, launch and mission control, propellant, and refurbishment. Costs per launch tend to scale inversely with the annual launch rate and vary with the type of launch vehicle. The calculated operating cost was $520/kg, or about 10 times lower than the space shuttle.

Finally, it was pointed out that neither the private sector nor the government could develop the ASP without the participation of the other. The extent of private sector involvement needed was seen as a function of which applications the ASP is developed for.

Chapter 4: ASP Applications and Institutional Perspectives. Three ASP applications were examined with the following conclusions: (1) launch vehicle: most promising civilian application, although potential pay-offs appear commensurate with technological complexity and development risks; (2) hypersonic transport: technological, political, and financial uncertainties cloud picture for industry participation, a necessary ingredient for HST development; and (3) military vehicle: most promising military application found to be strategic reconnaissance. Other military missions pose questions of cost-effectiveness, logistical feasibility, and effect on stability.

Only two and a half years ago, civilian and military experts testified before a Congressional subcommittee that they had no missions for hypersonic aircraft. Now hypersonics is the rage in NASA and the DOD and the ASP has been endorsed by the President. While technological progress and the economic incentives of reduced launch costs are ostensibly the reasons behind the decision to proceed with the ASP program, political factors may be overshadowing the technical and economic arguments about the merits of an ASP.

Among the less overt factors which may be coming to bear on the ASP development effort are: (1) an Executive Branch priority on reducing launch costs for SDI infrastructure, and (2) a long-standing desire within the USAF for an independent, non-NASA, manned spaceflight capacity.

5.2 Conclusion

In a nutshell, the ASP is being oversold. Chapter 2 showed that the futuristic vehicle's feasibility is critically dependent on scramjet engines performing over the most optimistic operating range. Chapter 3 showed that ASP development, fabrication, and operating costs are all likely to be higher than advertised. Furthermore, Chapter 4 showed that the vehicle does not appear to have myriad applications as is often suggested.

To be sure, large-scale space projects have always been difficult to get authorized, and so a certain amount of exaggeration can be expected. Nevertheless, problems arise when technological difficulties are downplayed, costs are underestimated, and utility is inflated, as the shuttle example of chapter 1 so readily attests.

5.3 Policy Recommendations

I recommend that policy-makers consider the following:

(1) Continue funding ASP research at current levels leading to a decision around 1990 on whether to proceed with phase 3, the construction of an experimental test vehicle. Phase 2 "technology demonstration" contracts awarded recently amount to $450 million over a three and a half year period. Put in perspective, $450 million is about the cost of launching the space shuttle four times. This does not seem like much to invest in technologies that may greatly reduce the cost of accessing low earth orbit. And, as shown in this thesis, an investment in reducing launch costs to LEO may pay for itself as more and more commercial activities become profitable. Furthermore, ASP research must be seen as a long-term funding commitment since aerospace design and development is a process that can stretch over decades.

(2) Question ASP operations cost predictions. The hard-sell of the ASP has resulted in overblown expectations. Some ASP enthusiasts are repeating the shuttle saga by suggesting that hundred-fold cost reductions are just over the horizon. They are not. First, even under the most optimistic conditions, the ASP would not be operational before the turn of the century. The nation's immediate, near, and intermediate term need for space transportation should be totally decoupled from the ASP program.

Second, it is important to remember that the ASP's reduced propellant requirements would not have as large an effect on operations costs as would any increase in the efficiency of pre-launch operations, assembly, check-out, inspection, or post-launch refurbishment. Any reduction in operations costs will require progress in collateral technologies such as automated check-out and maintenance systems, and intelligent avionics capable of real-time flight planning.

Third, any projection of future operations costs must have its predicted launch rate examined since this factor strongly affects the calculated launch cost to LEO.

The analysis in chapter 3 suggests that it is more realistic to plan for a ten-fold reduction in launch costs than a hundred-fold reduction. Furthermore, similar cost reductions may be possible with alternative launch vehicles.

(3) Question ASP development cost forecasts. Official estimates of $3 billion for ASP development appeared low compared to both my own estimate based on the Transcost model, and development costs for other, less complicated aerospace projects. Underestimating ASP development costs is not only inefficient in the long-run, but may precipitate some poor decisions in the short-run. An important near-term space policy issue is whether the government should replace the Challenger. The House space science subcommittee is reportedly "considering the wisdom of shelving a replacement [shuttle orbiter] in favor of putting the money--about $2 billion--into next-generation technology (Mann, 1986b)."

That approach seems attractive because some believe that "building a next-generation 'space plane' wouldn't take that much longer than gearing back up to assemble a fifth shuttle (Mann, 1986b)." Furthermore, with official estimates of ASP development costs around $2 billion, lawmakers may mistakenly believe that the nation could have an ASP for only a slightly higher cost than a new shuttle, and therefore decide not to authorize a replacement for the Challenger. Not building a replacement shuttle may or may not be a good decision, but that decision should not be influenced by unrealistic ASP cost projections.

(4) Question the utility of an ASP. While scramjet technology holds promise, it may be too much to expect that it will give us a vehicle able to combine the vantage point of a reconnaissance satellite with the maneuverability of an SR-71, deliver the ordnance of a bomber with the speed of an ICBM, whisk passengers across oceans with flight times like that of the Eastern Shuttle, and launch payloads into orbit with the ease of a DC-9.

(5) Balance military and civilian funding and management. The ASP is, at present, most definitely a military program with 80 percent of the funding coming from the DOD and 20 percent from NASA. There are two problems associated with this. First, while much of today's civilian aviation technology can trace its roots back to military research (for instance, the first U.S. commercial jet, the Boeing 707, arose from the technologies developed for the B-47 and B-52 bombers), the research orientation of NASA and the procurement orientation of the DOD are in conflict. The danger exists that military requirements may push

prototype development at the expense of basic research which would have made the vehicle cheaper or better.

Second, while there may be some defense applications, the most promising use of ASP technology appears to be as a civilian launch vehicle. Thus, the ASP may be better suited for civilian purposes, but the organization that contributes the most money to a project (in this case, the DOD) usually has the biggest say in how the technology is used. The factors which drive technology for military applications are very different from those which drive technology for commercial applications. Military specifications stress performance; on the other hand, commercial developments tend to emphasize lowered production costs, vehicle operating efficiency, and high availability with low maintenance. Thus, an ASP developed first for the military, and then "handed over" for commercial applications may not produce the most cost-effective launch vehicle possible.

(6) Do not push the ASP at the expense of alternative launch vehicle research. If a dual-fuel, vertically-launched rocket appears to offer economies similar to that of the ASP, as suggested in section 4.2, then we may be making the same mistake we made with the shuttle, namely, "putting all our eggs in one launch vehicle." Patently, an all rocket-powered vehicle would involve far less development risk than the ASP, and would not exhibit the extreme sensitivities to non-optimum performance levels that the scramjets are subject to. As suggested by the National Commission on Space, both concepts merit funding until more is known about the performance of scramjet engines.

Of course, the ASP appears superior to the shuttle. But any competition between 1973 technology (the shuttle) and projected 1990s technology (the ASP) is inherently uninformative. If a new space shuttle were built today incorporating the latest composite materials in the fuselage, a weight savings of over 7,000 kg has been projected (Morita, 1979, pp. 441-444). The correct comparison to make is between the ASP and the "next best alternative," whatever that may turn out to be. Future decision-makers should look at all available options before proceeding with production of the ASP.

(7) Do not sacrifice near-term aeronautical objectives for hypersonics. Chapter 4 described the White House Aeronautical Policy Review Committee's recommendations for research priorities in aeronautics. Specifically, they singled out subsonic R&D as the number one priority because the technological foundations it is based on will determine our success or failure in the other two areas: supersonics and hypersonics. However, the NASA aircraft research budget for FY86 shows that hypersonic research is receiving three times the funding of supersonic research (Coleman, 1986).

American preeminence in aeronautics depends on technical excellence and clear-cut cost advantages over foreign competitors. A low-cost, low-noise, fuel-efficient vertical take-off and landing (VTOL) civilian aircraft for use in urban areas might be an even greater boon to air travel than a long-range hypersonic aircraft. Likewise, the military's desire for a hypersonic interceptor must be balanced against other needs such as maneuverability, maintainability, fuel-efficiency, and reduced radar signatures, to name a few. We must avoid the trap of automatically thinking that flying higher and faster is necessarily better.

(8) Explore options for international cooperation. The possibilities for cooperation on the ASP between the US and the UK, our chief ASP competitor, are non-existent. There may be, however, the potential for a mutuallly beneficial US-French cooperative effort on a next-generation superscnic transport. The French already have an extensive technology base in supersonic transport design and have expressed interest in such cooperation.

```
10   REM ****************************************************
20   REM Mass Program: calculates the component masses based on
30   REM estimates of engine performance, payload mass and
40   REM various mass parameters. All masses in kilograms.
50   REM ****************************************************
60   REM
70   REM   Define terms
80   REM
90   REM del(1) = fuselage mass/mass contained
100  REM del(2) = mass of lift components/mass carried
110  REM del(3) = thermal protection mass/mass protected
120  REM del(4) = fixed equipment mass/initial mass
130  REM del(5) = air turbo-ramjet (ATR) mass/mass propelled
140  REM del(6) = scramjet engine mass/mass propelled
150  REM del(7) = rocket engine mass/mass propelled
160  REM del(8) = R1(eff) = (1-r1(eff))/r1(eff) =
170  REM                ATR propellant mass/inert mass
180  REM del(9) = R2 = (1-r2(eff))/r2(eff) =
190  REM                scramjet propellant mass/inert mass
200  REM del(10)= R3 = (1-r3(eff))/r3(eff) =
210  REM                rocket engine propellant masss/inert mass
220  REM
230  DIM DEL(10), A(10,10), B(10), M(10)
240  REM
250  REM   Read component mass fractions DEL(1) through DEL(10),
260  REM   based on technology level estimates as described in
270  REM   chapter 2
280  REM
290  DATA 0.4, 0.10, 0.20, 0.04, 0.02, 0.05, 0.037, 0.0973
300  FOR I = 1 TO 8
310  READ DEL(I)
320  NEXT I
330  DATA 0.30, 1.001
340  FOR I = 1 TO 2
350  READ DEL(I+8)
360  NEXT I
370  REM *****  Set iteration counter initially to zero ****
380  REM          MT = Tank Mass; set initially to zero
390  REM **************** MPL = Payload Mass **************
400  REM
410  IFLAG = 0
420  MT = 0
430  MPL = 10000
440  REM
450  REM ********** Read in 10 by 10 matrix A ***************
460  REM
470  FOR I = 1 TO 10
480  FOR J = 1 TO 10
490  A(I,J) = -DEL(I)
500  A(J,J) = 1-DEL(J)
510  NEXT J
```

```
520  NEXT I
530  A(1,1)=1: A(1,2)=0: A(1,3)=0: A(1,8)=0: A(1,9)=0: A(1,10)=0
540  A(2,8)=0: A(2,9)=0: A(2,10)=0
550  A(3,3)=1: A(3,8)=0: A(3,9)=0: A(3,10)=0
560  A(6,8)=0
570  A(7,8)=0: A(7,9)=0: A(7,10)=0
580  A(8,8)=1
590  A(9,8)=0: A(9,9)=1
600  A(10,8)=0: A(10,9)=0: A(10,10)=1
610   REM
620   REM ********** Read in matrix B **********************
630   REM
640   FOR I = 1 TO 10
650   B(I) = DEL(I)*(MT+MPL)
660   NEXT I
670   B(4) = B(4) + 4240
680   REM
690   REM ***** Call matrix solving subroutine ************
700   REM
710   GOSUB 770
720   REM
730   REM ***** Check whether Tank Mass has converged ******
740   REM
750   GOTO 1190
760   REM
770   REM ******** Subroutine: Solves Ax=b by pivoting ******
780   REM
790   FOR I = 1 TO 9
800   DIV = A(I,I)
810   FOR J = 1 TO 10
820   A(I,J) = A(I,J)/DIV
830   NEXT J
840   B(I) = B(I)/DIV
850   I1 = I + 1
860   FOR K = I1 TO 10
870   DIV = A(K,I)
880   IF DIV = 0 GOTO 930
890   B(K) = B(K)/DIV-B(I)
900   FOR J = 1 TO 10
910   A(K,J) = A(K,J)/DIV-A(I,J)
920   NEXT J
930   NEXT K
940   NEXT I
950   B(10) = B(10)/A(10,10)
960   A(10,10) = 1
970   M(10) = B(10)
980   M(9) = B(9)-A(9,10)*M(10)
990   M(8) = B(8)-A(8,10)*M(10)-A(8,9)*M(9)
1000   M(7) = B(7)-A(7,10)*M(10)-A(7,9)*M(9)-A(7,8)*M(8)
1010   M(6) = B(6)-A(6,10)*M(10)-A(6,9)*M(9)-A(6,8)*M(8)-A(6,7)*M(7)
1020   M(5) = B(5)-A(5,10)*M(10)-A(5,9)*M(9)-A(5,8)*M(8)-A(5,7)*M(7)
1030   M(5) = M(5)-A(5,6)*M(6)
1040   M(4) = B(4)-A(4,10)*M(10)-A(4,9)*M(9)-A(4,8)*M(8)-A(4,7)*M(7)
1050   M(4) = M(4)-A(4,6)*M(6)-A(4,5)*M(5)
```

```
1060   M(3) = B(3)-A(3,10)*M(10)-A(3,9)*M(9)-A(3,8)*M(8)-A(3,7)*M(7)
1070   M(3) = M(3)-A(3,6)*M(6)-A(3,5)*M(5)-A(3,4)*M(4)
1080   M(2) = B(2)-A(2,10)*M(10)-A(2,9)*M(9)-A(2,8)*M(8)-A(2,7)*M(7)
1090   M(2) = M(2)-A(2,6)*M(6)-A(2,5)*M(5)-A(2,4)*M(4)-A(2,3)*M(3)
1100   M(1) = B(1)-A(1,10)*M(10)-A(1,9)*M(9)-A(1,8)*M(8)-A(1,7)*M(7)
1110   M(1) = M(1)-A(1,6)*M(6)-A(1,5)*M(5)-A(1,4)*M(4)-A(1,3)*M(3)
1120   M(1) = M(1)-A(1,2)*M(2)
1130   IFLAG = IFLAG + 1
1140   PRINT IFLAG
1150   RETURN
1160   REM
1170   REM ********* Calculate new tank mass ************
1180   REM
1190   REM
1200   MPROP = M(8) + M(9) + M(10)
1210   OLDTANK = MT
1220   MT = .2*(MPROP^.9)
1230   IF ABS(MT-OLDTANK) > 1 GOTO 450
1240   REM
1250   REM ********* Print mass values ******************
1260   REM
1270   PRINT "Fuselage mass ="; INT(M(1)) "kg"
1280   PRINT "Wing mass ="; INT(M(2)) "kg"
1290   PRINT "Thermal Protection System mass ="; INT(M(3)) "kg"
1300   PRINT "Fixed Equipment mass ="; INT(M(4)) "kg"
1310   PRINT "Tank mass ="; TNT(MT) "kg"
1320   PRINT "Total engine mass ="; INT(M(5)+M(6)+M(7)); "kg"
1330   PRINT "          Air Turbo ramjet mass ="; INT(M(5)) "kg"
1340   PRINT "          Scramjet mass ="; INT(M(6)) "kg"
1350   PRINT "          Rocket engine mass ="; INT(M(7)) "kg"
1360   PRINT "Total Propellant mass ="; INT(MPROP) "kg"
1370   PRINT "          ATR Propellant mass ="; INT(M(8)) "kg"
1380   PRINT "          Scramjet Propellant mass ="; INT(M(9)) "kg"
1390   PRINT "          Rocket Engine Propellant mass ="; INT(M(10)) "kg"
1400   PRINT "Payload mass ="; MPL "kg"
1410   MTOTAL = 0
1420   FOR I = 1 TO 10
1430   MTOTAL = MTOTAL + M(I)
1440   NEXT I
1450   MTOTAL = MTOTAL + MPL + MT
1460   PRINT "Total vehicle mass ="; INT(MTOTAL) "kg"
1470   PRINT "Payload Mass Fraction = "; MPL/MTOTAL
1480   PRINT "Number of iterations ="; IFLAG
1490   REM **********************************************
1500   END
```

Appendix B: Output of Mass-Estimating Program for Various Levels of Maximum Scramjet Operating Speed

Scramjet cut-off at M=12

Fuselage mass = 199088 kg
Wing mass = 95019 kg
Thermal Protection System mass = 158365 kg
Fixed Equipment mass = 133948 kg
Tank mass = 106001 kg
Total engine mass = 247770 kg
    Air Turbo ramjet mass = 64854 kg
    Scramjet mass = 147759 kg
    Rocket engine mass = 35157 kg
Total Propellant mass = 2292529 kg
    ATR Propellant mass = 287539 kg
    Scramjet Propellant mass = 346901 kg
    Rocket Engine Propellant mass = 1658088 kg
Payload mass = 10000 kg
Total vehicle mass = 3242723 kg
Payload Mass Fraction = 3.083827E-03
Number of iterations = 46

Scramjet cut-off at M=13

Fuselage mass = 107465 kg
Wing mass = 51291 kg
Thermal Protection System mass = 85485 kg
Fixed Equipment mass = 71537 kg
Tank mass = 57841 kg
Total engine mass = 129289 kg
    Air Turbo ramjet mass = 33648 kg
    Scramjet mass = 76662 kg
    Rocket engine mass = 18977 kg
Total Propellant mass = 1169527 kg
    ATR Propellant mass = 149185 kg
    Scramjet Propellant mass = 211483 kg
    Rocket Engine Propellant mass = 808859 kg
Payload mass = 10000 kg
Total vehicle mass = 1682438 kg
Payload Mass Fraction = 5.943753E-03
Number of iterations = 32

Scramjet cut-off at M=14

Fuselage mass = 72300 kg
Wing mass = 34507 kg
Thermal Protection System mass = 57511 kg
Fixed Equipment mass = 47875 kg
Tank mass = 33583 kg
Total engine mass = 84293 kg
    Air Turbo ramjet mass = 21817 kg
    Scramjet mass = 49708 kg
    Rocket engine mass = 12767 kg
Total Propellant mass = 745821 kg

ATR Propellant mass = 96731 kg
Scramjet Propellant mass = 159433 kg
Rocket Engine Propellant mass = 489656 kg
Payload mass = 10000 kg
Total vehicle mass = 1090894 kg
Payload Mass Fraction = 9.166789E-03
Number of iterations = 24

<u>Scramjet cut-off at M=15</u>
Fuselage mass = 53863 kg
Wing mass = 25707 kg
Thermal Protection System mass = 42846 kg
Fixed Equipment mass = 35579 kg
Tank mass = 28198 kg
Total engine mass = 60882 kg
Air Turbo ramjet mass = 15669 kg
Scramjet mass = 35700 kg
Rocket engine mass = 9511 kg
Total Propellant mass = 526409 kg
ATR Propellant mass = 69473 kg
Scramjet Propellant mass = 130191 kg
Rocket Engine Propellant mass = 326744 kg
Payload mass = 10000 kg
Total vehicle mass = 783486 kg
Payload Mass Fraction = 1.276346E-02
Number of iterations = 20

<u>Scramjet cut-off at M=16</u>
Fuselage mass = 43540 kg
Wing mass = 20780 kg
Thermal Protection System mass = 34634 kg
Fixed Equipment mass = 28742 kg
Tank mass = 22258 kg
Total engine mass = 47852 kg
Air Turbo ramjet mass = 12251 kg
Scramjet mass = 27912 kg
Rocket engine mass = 7688 kg
Total Propellant mass = 404749 kg
ATR Propellant mass = 54316 kg
Scramjet Propellant mass = 115192 kg
Rocket Engine Propellant mass = 235239 kg
Payload mass = 10000 kg
Total vehicle mass = 612559 kg
Payload Mass Fraction = 1.632495E-02
Number of iterations = 17

<u>Scramjet cut-off at M=17</u>
Fuselage mass = 36724 kg
Wing mass = 17527 kg
Thermal Protection System mass = 29212 kg
Fixed Equipment mass = 24252 kg

Tank mass = 18271 kg
Total engine mass = 39289 kg
Air Turbo ramjet mass = 10006 kg
Scramjet mass = 22797 kg
Rocket engine mass = 6485 kg
Total Propellant mass = 325036 kg
ATR Propellant mass = 44363 kg
Scramjet Propellant mass = 105219 kg
Rocket Engine Propellant mass = 175453 kg
Payload mass = 10000 kg
Total vehicle mass = 500315 kg
Payload Mass Fraction = .0199874
Number of iterations = 15

<u>Scramjet cut-off at M=18</u>
Fuselage mass = 32141 kg
Wing mass = 15340 kg
Thermal Protection System mass = 25567 kg
Fixed Equipment mass = 21247 kg
Tank mass = 15553 kg
Total engine mass = 33553 kg
Air Turbo ramjet mass = 8503 kg
Scramjet mass = 19374 kg
Rocket engine mass = 5675 kg
Total Propellant mass = 271783 kg
ATR Propellant mass = 37702 kg
Scramjet Propellant mass = 99392 kg
Rocket Engine Propellant mass = 134688 kg
Payload mass = 10000 kg
Total vehicle mass = 425187 kg
Payload Mass Fraction = 2.351903E-02
Number of iterations = 14

<u>Scramjet cut-off at M=19</u>
Fuselage mass = 28972 kg
Wing mass = 13827 kg
Thermal Protection System mass = 23046 kg
Fixed Equipment mass = 19177 kg
Tank mass = 13653 kg
Total engine mass = 29601 kg
Air Turbo ramjet mass = 7468 kg
Scramjet mass = 17016 kg
Rocket engine mass = 5116 kg
Total Propellant mass = 235158 kg
ATR Propellant mass = 33113 kg
Scramjet Propellant mass = 96538 kg
Rocket Engine Propellant mass = 105506 kg
Payload mass = 10000 kg
Total vehicle mass = 373438 kg
Payload Mass Fraction = .0267782
Number of iterations = 12

Scramjet cut-off at M=20

Fuselage mass = 26713 kg
Wing mass = 12749 kg
Thermal Protection System mass = 21249 kg
Fixed Equipment mass = 17706 kg
Tank mass = 12287 kg
Total engine mass = 26790 kg
    Air Turbo ramjet mass = 6733 kg
    Scramjet mass = 15340 kg
    Rocket engine mass = 4717 kg
Total Propellant mass = 209157 kg
    ATR Propellant mass = 29851 kg
    Scramjet Propellant mass = 95796 kg
    Rocket Engine Propellant mass = 83509 kg
Payload mass = 10000 kg
Total vehicle mass = 336653 kg
Payload Mass Fraction = .0297041
Number of iterations = 12

Scramjet cut-off at M=21

Fuselage mass = 25007 kg
Wing mass = 11935 kg
Thermal Protection System mass = 19892 kg
Fixed Equipment mass = 16597 kg
Tank mass = 11248 kg
Total engine mass = 24672 kg
    Air Turbo ramjet mass = 6178 kg
    Scramjet mass = 14077 kg
    Rocket engine mass = 4416 kg
Total Propellant mass = 189595 kg
    ATR Propellant mass = 27395 kg
    Scramjet Propellant mass = 96198 kg
    Rocket Engine Propellant mass = 66001 kg
Payload mass = 10000 kg
Total vehicle mass = 308948 kg
Payload Mass Fraction = 3.236785E-02
Number of iterations = 11

Scramjet cut-off at M=22

Fuselage mass = 23918 kg
Wing mass = 11415 kg
Thermal Protection System mass = 19025 kg
Fixed Equipment mass = 15891 kg
Tank mass = 10580 kg
Total engine mass = 23323 kg
    Air Turbo ramjet mass = 5825 kg
    Scramjet mass = 13273 kg
    Rocket engine mass = 4223 kg
Total Propellant mass = 177144 kg
    ATR Propellant mass = 25830 kg
    Scramjet Propellant mass = 99030 kg
    Rocket Engine Propellant mass = 52283 kg
Payload mass = 10000 kg

Total vehicle mass = 291299 kg
Payload Mass Fraction = 3.432888E-02
Number of iterations = 11

<u>Scramjet cut-off at M=23</u>
Fuselage mass = 23174 kg
Wing mass = 11060 kg
Thermal Protection System mass = 18433 kg
Fixed Equipment mass = 15410 kg
Tank mass = 10123 kg
Total engine mass = 22402 kg
    Air Turbo ramjet mass = 5585 kg
    Scramjet mass = 12724 kg
    Rocket engine mass = 4092 kg
Total Propellant mass = 168654 kg
    ATR Propellant mass = 24762 kg
    Scramjet Propellant mass = 103190 kg
    Rocket Engine Propellant mass = 40702 kg
Payload mass = 10000 kg
Total vehicle mass = 279259 kg
Payload Mass Fraction = 3.580905E-02
Number of iterations = 10

Appendix C: Detailed Cost Break-Down for Various Scramjet Cut-off Mach Numbers

scramjet cut-off	12	13	14	15	16	17
Haf	18.9 B	16.6 B	15.3 B	14.4 B	13.7 B	13.2 B
Hatr	2.02 B	1.52 B	1.30 B	1.16 B	1.08 B	1.02 B
Hsj	1.00 B	1.00 B	1.00 B	1.00 B	1.00 B	1.00 B
Hr	3.04 B	2.04 B	1.59 B	1.32 B	1.15 B	1.03 B
Cdev	27.5 B	23.3 B	21.1 B	19.7 B	18.6 B	17.9 B
Faf	2.08 B	1.56 B	1.30 B	1.14 B	1.03 B	951 M
Fatr	8.8 M	6.1 M	4.8 M	4.0 M	3.5 M	3.1 M
Fsj	39.4 M	27.3 M	21.4 M	17.9 M	15.6 M	13.9
Fr	12.7 M	9.5 M	7.9 M	6.9 M	6.3 M	5.8 M
Cprod	2.6 B	2.0 B	1.7 B	1.5 B	1.3 B	1.2 B
Cm+Cc+Ce	1.4 M	1.4 M	1.4 M	1.4 M	1.4 M	1.4 M
Cprop	1.0 M	0.62 M	0.46 M	0.35 M	0.29 M	0.27
Crf	7.4 M	5.8 M	4.9 M	4.3 M	4.1 M	3.8 M
Cops	9.8 M	7.8 M	6.8 M	6.1 M	5.8 M	5.5 M
total non-recurring	54.4 B	43.3 B	37.7 B	34.2 B	31.6 B	29.9 B

Appendix C: Detailed Cost Break-Down for Various Scramjet Cut-off Mach Numbers (cont.)

scramjet cut-off	18	19	20	21	22	23
Haf	12.9 B	12.6 B	12.4 B	12.2 B	12.1 B	12.0
Hatr	0.97 B	0.94 B	0.92 B	0.90 B	0.89 B	0.88
Hsj	1.0 B	1.0 B	1.0 B	1.0 B	1.0 B	1.0 B
Hr	0.95 B	0.89 B	0.84 B	0.81 B	0.78 B	0.77
Cdev	17.4 B	17.0 B	16.7 B	16.4 B	16.3 B	16.2 B
Faf	0.89 B	0.85 B	0.82 B	0.79 B	0.78 B	0.76 B
Fatr	2.8 M	2.6 M	2.5 M	2.4 M	2.3 M	2.2 M
Fsj	12.7 M	11.8 M	11.1 M	10.6 M	10.2 M	10.0
Fr	5.5 M	5.2 M	5.0 M	4.9 M	4.8 M	4.7 M
Cprod	1.1 B	1.1 B	1.0 B	1.0 B	1.0 B	1.0 B
Cm+Cc+Ce	1.4 M	1.4 M	1.4 M	1.4 M	1.4 M	1.4 M
Cprop	0.24 M	0.22 M	0.21 M	0.21 M	0.21 M	0.21
Crf	3.6 M	3.5 M	3.4 M	3.3 M	3.2 M	3.2 M
Cops	5.2 M	5.1 M	5.0 M	4.9 M	4.8 M	4.8 M
total non-recurring	28.6 B	27.7 B	27.0 B	26.3 B	26.0 B	25.8 B

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