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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Moffett Field, California 94035

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REPORT MDC A0013 • 2 OCTOBER 1970
VOLUME III • PART 2

FOREWORD

This report summarizes the results of Phase II of the Hypersonic Research Facilities Study performed from 19 September 1969 through 2 January 1970 under National Aeronautics and Space Administration Contract NAS 2-5458 by McDonnell Aircraft Company (MCAIR), St. Louis, Missouri, a division of McDonnell Douglas Corporation.

The study was sponsored by the Office of Advanced Research and Technology with Mr. Richard H. Petersen as Study Monitor and Mr. Hubert Drake as alternate Study Monitor.

Mr. Charles J. Pirrello was Manager of the HYFAC project and Mr. Paul A. Czysz was Deputy Manager. The study was conducted within MCAIR Advanced Engineering, which is directed by Mr. R. H. Belt, Vice President, Aircraft Engineering. The HYFAC study team was an element of the Advanced Systems Concepts project managed by Mr. Harold D. Altis.

The basic task of Phase II was to subdivide into research tasks the desirable research objectives for hypersonic flight determined in Phase I, and to refine and evaluate through parametric studies those attractive facilities retained from Phase I. The Phase II study has been conducted in accordance with the requirements and instructions of NASA RFP A-15109 (HK-81), McDonnell Technical Proposal Report G970, and OART Correspondence received during the Phase II period.

This is Volume III, Part 2 of the overall HYFAC Report, which is organized as follows:

		<u>NASA CONTRACTOR REPORT NUMBER</u>
Volume I	Summary	CR 114322
Volume II	Phase I Preliminary Studies	
	Part 1 - Research Requirements and Ground Facility Synthesis	CR 114323
	Part 2 - Flight Vehicle Synthesis	CR 114324
Volume III	Phase II Parametric Studies	
	Part 1 - Research Requirements and Ground Facility Synthesis	CR 114325
	Part 2 - Flight Vehicle Synthesis	CR 114326
Volume IV	Phase III Final Studies	
	Part 1 - Flight Research Facilities	CR 114327
	Part 2 - Ground Research Facilities	CR 114328
	Part 3 - Research Requirements Analysis and Facility Potential	CR 114329
Volume V	Limited Rights Data	CR 114330
Volume VI	Operational System Characteristics	CR 114331

MCDONNELL AIRCRAFT

ACKNOWLEDGEMENTS

This work was performed by an Aircraft Advanced Engineering study team with Charles J. Pirrello as Study Manager.

The following contributed to the contents of this volume:

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SUMMARY

Airbreathing hypersonic aircraft employing liquid hydrogen fuel have the potential of satisfying a number of mission requirements in the 1980-2000 time period. However, major advances in the technological state of the art are necessary before such aircraft can be considered either feasible or practical. The objective of Contract NAS 2-5458 was to assess the research and development requirements for hypersonic aircraft and based on these requirements, to provide the NASA with characteristics of a number of desirable hypersonic research facilities. The study is organized in three phases. Phase I was a preliminary analysis of a broad group of concepts which were reduced to seven flight research facilities and eleven ground research facilities for Phase II study. The purpose of Phase II was to perform parametric studies to refine the facility designs and obtain sensitivity information in the neighborhood of "near optimum" designs, and to select those facilities that appear most attractive in the sense of research potential vs cost for further refinement in Phase III. This part of Volume III presents the results of the synthesis of the flight research facilities. The significant results obtained are:

1. Flight research aircraft used in conjunction with existing ground facilities are capable of accomplishing about 80 to 95% of the necessary research for $M = 4.5$ to $M = 7$ operational systems.
2. Flight research aircraft used in conjunction with existing ground facilities are capable of accomplishing about 65 to 85% of the necessary research for $M = 10$ to $M = 12$ operational systems.
3. Development of ramjet, scramjet, and convertible scramjet engines are significant cost elements. Development of turbomachinery, if required, is a major cost element.
4. Off-the-shelf rocket and turbojet engines are available to satisfy the requirements for accelerator engines, thus reducing program costs and program initiation risks.
5. Propellant costs are a minor cost element. The use of LH_2 for rockets and hypersonic engines is both feasible and economical.
6. The diversity of the defined research clearly indicates that the flight research vehicles should be flexible and adaptable to new and varying research goals. When evaluated for a broad research program capability, unmanned vehicles neither provide this capability nor do they reduce overall program costs.
7. Active thermal protection systems and integral propellant tanks reduce vehicle weight and cost.
8. A conservative design approach has a small cost effect. All vehicles are therefore designed for 3.5 g at maximum thermal protection system temperatures and 5.0 g structurally at reduced temperatures and provide a 1500 pound payload capability.

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9. A design cruise speed capability of $M = 6$ and 10 minutes of cruise was best for the $M = 6$ to $M = 8$ class of vehicles.
10. A design cruise speed capability of $M = 12$ and 5 minutes of cruise was best for the $M = 8$ to $M = 12$ class of vehicles.
11. Airlaunched vehicles are substantially lower in program cost and provide the best test operation capability.
12. The most attractive vehicle is a $M = 12$ airlaunched, rocket accelerated and rocket cruise manned vehicle and is available at an acquisition cost of under 300 million dollars. Adaptability of this vehicle to accommodate modular installations of advanced airbreathing propulsion systems and to accommodate various types of thermal protection systems is feasible.

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LIST OF SYMBOLS

<u>Symbol</u>	<u>Definition</u>
a	acceleration
A	area
AR	aspect ratio
α	angle of attack, ratio of wing span to vehicle length
s	ratio of mean aerodynamic chord to vehicle length, side slip
b	wing span
C_D	drag coefficient
C_{D_0}	zero lift drag coefficient
C	cross sectional area of wind tunnel test section
$\bar{c}$	mean aerodynamic chord
C_R	wing root chord
C_T	wing tip chord
C_l	balance normal force load capacity divided by balance diameter squared
C_L	lift coefficient
C_{L_α}	lift curve slope
$C_{L_{\alpha_0}}$	lift curve slope at zero lift
C_m	pitching moment
γ	ratio of specific heats, flight path angle
d	diameter, balance diameter
D	drag
δ	deflection
Δ	increment between two values

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LIST OF SYMBOLS (Cont)

<u>Symbol</u>	<u>Definition</u>
L/D	lift to drag ratio
m	mass
M	Mach number, bending moment
$\dot{m}$	mass flow
n_z	flight path normal load factor
η_{KE}	inlet kinetic energy efficiency
N.F.	normal force
n	inlet height-to-width ratio
N ₂ O ₄	nitrogen tetroxide
O ₂	molecular oxygen
o/f	oxidizer to fuel weight flow ratio
p	pressure
ϕ	fuel equivalence ratio, ratio of actual fuel flow to stoichiometric fuel flow
θ	angle between shock attachment point and cowl lip
q	dynamic pressure
R	specific gas constant
R _E	mean radius of the earth 6,371,100 m
R*	universal gas constant (8.31432 joules/°K mol)
Re	Reynolds number
ρ	density
σ, F_s	stress
S	area
S/R	dimensionless entropy

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LIST OF SYMBOLS

<u>Symbol</u>	<u>Definition</u>
t	time
T	temperature
T_r	recovery temperature
T_w	wall temperature
V	velocity
Vol	volume
$\dot{w}$	weight flow
w	weight
ψ	heading angle, yaw angle
Z	geometric altitude

LIST OF SYMBOLS (Cont)

SUBSCRIPTS

Propulsion Station Designations

O	free stream
c	capture, a fixed reference area on vehicle
cowl	cowl lip
2	engine face
3	engine exit
e	nozzle exit
t	nozzle throat

General

aero	attributable to aerodynamic forces
c	chamber conditions, cruise
cent	attributable to centrifugal forces
D	drag
E	empty
e	engine exit
eff	effective
f	final
F	frontal
i	initial
∞	free stream
G	associated with gravity forces, gross
I	ideal
M	maneuvering

LIST OF SYMBOLS (Cont)

max	maximum
min	minimum
N	net
o	isentropic reservoir conditions, evaluated at zero lift
prop	attributable to propulsion system
p	associated with pressure forces, planform
R	wing root
S	structural
s	vehicle, model stagnation
t	total conditions corresponding to isentropic case
TO	takeoff
TJ	attributable to turbojet propulsion system
SJ	attributable to scramjet propulsion system
t	wing tip
test	associated with test time
wet	wetted
vac	associated with vacuum conditions
x	longitudinal direction
y	lateral direction
z	vertical direction

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LIST OF ABBREVIATIONS

<u>Abbreviation</u>	<u>Definition</u>
ARC	Ames Research Center
A	ampere
A-h	Ampere-hour
AB	all body
A/D	analog to digital conversion
Alt	altitude
AM	amplitude modulation
Aero 50	Aerozine 50, a 50/50 mixutre of UDMH and Hydrazine
bp	boiling point
Btu	British thermal unit
°C	degrees Celscius (centigrade)
c.g.	center of gravity
c.p.	center of pressure
cm	centimeters
CSJ	convertible scramjet
db	decibel
D/A	digital to analog conversion
diam	diameter
eng	engine
°F	degrees Fahrenheit
FRC	Flight Research Center
ft	feet
fps	feet per second
GE	General Electric Co.

LIST OF ABBREVIATIONS (Cont)

<u>Abbreviation</u>	<u>Definition</u>
hr	hour
Hz	hertz
HF	high frequency
HTO	horizontal takeoff
HYFAC	Hypersonic Research Facilities
ILS	instrument landing system
in.	inch
inst	installed
IRFNA	inhibited red fuming nitric acid
J	Joule
JP	jet propulsion fuel
°K	degrees Kelvin (absolute)
kg	kilogram
L	liquid
lb	pounds, force
LO ₂	liquid oxygen
LH ₂	liquid hydrogen
lbm	pounds, mass
mi	mile
m	meter
max	maximum
min	minimum
MCAIR	McDonnell Aircraft Company
MDAC (EAST)	McDonnell Douglas Astronautics Company (EAST)

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LIST OF ABBREVIATIONS (Cont)

<u>Abbreviation</u>	<u>Definition</u>
nmi	nautical mile
N	newtons
No.	number
OWE	operational weight empty
psi	pounds per square inch
PFRT	Preliminary Flight Rating Test
P&WA	Pratt & Whitney Aircraft
°R	degrees Rankine (absolute)
R&D	research and development
RDT&E	research, development, test, and evaluation
RF	radio frequency
RJ	ramjet
RKT	rocket
RP	rocket propellant
s, sec	seconds
SJ	scramjet
smi	statute mile
TF	turbofan
TIT	turbine inlet temperature
TJ	turbojet
TMC	The Marquard Corporation
TRJ	turboramjet
TOGW	takeoff gross weight
UARL	United Aircraft Research Laboratory

MCDONNELL AIRCRAFT

LIST OF ABBREVIATIONS (Cont)

<u>Abbreviation</u>	<u>Definition</u>
UDMH	unsymmetrical dimethyl hydrazine
UHF	ultra high frequency
uninst	uninstalled
VTO	vertical takeoff
V	volt
WB	winged body
W/O	without
wt	weight
W	watt

4. FLIGHT RESEARCH VEHICLE SYNTHESIS

(U) The seven Phase I flight research vehicle concepts indicated on Figure 4-1 were refined in Phase II through optimization and tradeoff studies. Design features and operational characteristics were varied so as to identify the best combination of variables for each of the seven concepts.

(U) Figure 4-2 lists the studies performed on the seven vehicles. Three configurations were selected on which to conduct the majority of the studies. Results from the study of these configurations were then applied, as applicable, to the remaining concepts, to insure that the best design features were incorporated in all vehicles.

(U) The principal elements compared in Phase II and the concepts on which they were analyzed are as follows:

- (a) Control Mode - Manned vs Unmanned - Compared on the B233 and B284 Mach 12, rocket, air-launched vehicles.
- (b) Cruise Engine-Rocket vs Scramjet - Compared on the B232 and B233 Mach 12, manned, air-launched vehicles.
- (c) Propulsion Mode - Airbreather vs Rocket - Compared on the B252 and B260 Mach 12, manned, horizontal takeoff vehicles.
- (d) Launch Mode - Ground Takeoff vs Air Launch - Compared on the B233 and B260 Mach 12, manned, rocket vehicles.
- (e) Design Speed - Mach 6 vs Mach 12 - Compared on the Mach 6 B212 and the Mach 12 B257 airbreathing, manned, ground takeoff vehicles and the Mach 6 B207 and the Mach 12 B232 rocket boosted, manned, air-launched vehicles.

(U) Sections 4.1 and 4.2 describe the design requirements, the technical approach, and methodology used in performing the Phase II tradeoff studies. A complete description of the vehicles used as the basis for the parametric and tradeoff studies is given in Section 4.3. The results of the tradeoff studies and their impact on the weight, cost, and research value of the flight research aircraft are discussed in Section 4.4. Overall summaries of the design characteristics, performance, and costs of the many flight research aircraft studied during Phase II are presented in Section 4.5.

4.1 DESIGN REQUIREMENTS

(U) Design criteria and ground rules selected for the Phase II flight research aircraft refinement and tradeoff studies were based primarily on information obtained from the Phase I studies. For the most part, the design requirements are unchanged from Phase I. However, some changes have been made to insure consistency, to increase the validity of the study results, and to take advantage of specific improvements identified during Phase I. In the following paragraphs, flight research aircraft design criteria and ground rules selected for Phase II are presented. Significant differences between Phase I and II design criteria and ground rules are also discussed.

(U) FIGURE 4-1 PHASE I FLIGHT RESEARCH VEHICLES
RETAINED FOR PHASE II REFINEMENT

CONFIGURATION NO.	207	212	232	233	257	260	284
Mach Number	6	6	12	12	12	12	12
Engine	RKT/RJ	TJ/RJ	RKT/SJ	RKT	TJ/CSJ	RKT	RKT
Launch Mode	Air	HTO	Air	Air	HTO	HTO/VTO	Air
Body Shape	AB	WB	AB	AB	AB & BWB	AB	AB
Control Mode	Manned	Manned	Manned	Manned	Manned	Manned	Unmanned
Baseline Engines	RKT- LR-129	TJ F100-GE-100 (H ₂)	RKT LR-129	RKT LR-129	TJ F100-GE-100 (H ₂)	RKT LR-129	RKT LR-129
	RJ MCAIR 2D	RJ MCAIR 2D	SJ MCAIR		CSJ MCAIR		
Engines Used In Tradeoff Study	RKT RL10-A-3-8	TJ F100-GE-100 (JP) STJ-233		RKT RL10-A-3-8 J2S		RKT J2S	RKT RL10-A-3-8

WB - Winged Body
AB - All Body
BWB - Blended Wing Body
HTO - Horizontal Takeoff
VTO - Vertical Takeoff

(U) FIGURE 4-2 TRADEOFF STUDIES
PHASE II FLIGHT RESEARCH VEHICLES

CONFIGURATION	B207	B212	B232	B233	B257	B260	B284
Engine	Rubberized, Off Shelf		Rubberized, Off Shelf	Rubberized, Off Shelf		Rubberized, Off Shelf	Rubber- ized Off Shelf
Mach No.	6,7,8			8,10,12	8,10,12	8,10,12	
T/W Rubberized	1.25,1.5,2.0			1.25,1.5,2.0			
-Off Shelf	-			1.117,1.375,1.55	0.96,1.17		
- JP (Boost)	-			-	0.97,1.03		
LE Sweep/ Fineness Ratio				75,77,80	75,77,80		
				-	3/1,4/1		
Test Time		5,10,15		5,10,15	5,7.5,10		
Basing	Single,Dual	Single,Dual		Single,Dual	Single, Dual	Single,Dual	
Landing				Power(1), Unpower			
Payload - Wt				600,1000,1400			
Density				20,25,30			
Fuel		JP,LH ₂		NBP(2), Subcooled	JP,LH ₂ , NBP, Subcooled		
TPS	Active,Passive (Inlet)	Passive,Hot		Passive, Active			
Tankage		Int.,Non-Int.					
Load Factor(3)		3.5,5.0,7.33		3.5,5.0,7.33	3.5,5.0, 7.33		

- (1) Pulsed Rocket
(2) Normal Boiling Point
(3) Heat Shield - 3.5 and 5.0

4.1.1 (U) RESEARCH AIRCRAFT DESIGN CRITERIA AND GROUND RULES - A summary of the design criteria and ground rules used as the basis for Phase II flight research aircraft studies is given in Figure 4-3. Discussion concerning the selection of these criteria is presented in Volume II, Part 2, which documents the results of the Phase I study. The most significant differences between Phase I and Phase II criteria and ground rules are:

(a) Number of Flights - Initial studies in Phase I were based on 180 flights for air-launched aircraft and 220 for horizontal takeoff aircraft. This was based on a projected reduced utility rate for the air-launched vehicles. Further evaluation showed that this difference could not be completely substantiated. Therefore, for consistency, Phase II comparisons were based on 200 flights for all aircraft.

(b) Fuel Temperature - Phase I aircraft were compared on the basis of normal boiling point hydrogen. The results of the Phase I and early Phase II studies showed, however, that the use of subcooled fuel results in a weight reduction primarily due to the lower design tank pressure (10 psig/6.9N/cm²) resulting from the reduced vapor pressure. This yields a 1% to 2.7% reduction in program cost. Therefore, the Phase II baseline aircraft employ subcooled hydrogen.

(c) Thermal Protection System - Preliminary studies at the beginning of Phase I indicated that an active cooling system on the primary structure of the Mach 12 aircraft would result in minimum weight. Active cooling was therefore selected as the baseline for the Phase I, Mach 12 aircraft. However, at the direction of the study monitor, passive insulation systems were used as the baseline design for Phase II aircraft analysis.

(d) Rocket I_{sp} - In the Phase I mission performance studies, the I_{sp} for both rocket engines and the airbreather engines was reduced by 5% to account for differences usually found between originally estimated vehicle performance and subsequent actual flight operational performance. For Phase II, it was decided that a 1% reduction was more representative for rocket engine performance. Therefore, for the Phase II performance studies, the I_{sp} was reduced 1% for rocket engines and the 5% reduction for airbreather engines was retained.

4.1.2 (U) PERFORMANCE AND SIZING GROUND RULES - The following paragraphs summarize pertinent assumptions and ground rules which were used for vehicle sizing in the Phase II technical evaluations.

4.1.2.1 (U) Rocket Accelerator - Two Base Operation - The acceleration and climb phase of flight follows the flight path that yields minimum propellant usage to the constant test Mach number at $(L/D)_{\max}$ equilibrium altitude. Cruise is at $(L/D)_{\max}$ at equilibrium altitude and the descent phase of flight is unpowered (zero fuel usage). For rubberized rocket engines, the nominal thrust-to-weight ratio (vacuum thrust to takeoff gross weight) is 1.5. For the rocket/ramjet configuration, where the ramjet is used only in the cruise phase of flight, the installed ramjet thrust when operating at minimum SFC is set equal to the drag at the start of cruise. For the rocket/scramjet configuration, where the scramjet is used only in the cruise phase of flight, the scramjet capture area is sized as 4.5% of vehicle planform area.

(U) FIGURE 4-3 PHASE II CRITERIA AND GROUND RULES

All Baseline Research Vehicles	
Test Time	5 min
Operational Life	200 flights
Payload	
Weight	1000 lb (.454 kg)
Density	20 lb/ft ³ (320 kg/m ³)
Design Limit Load Factor	
Taxi	2.0 g
Structure	5.0 g
Thermal Protection	3.5 g
Dynamic Pressure	2500 psf (119,300 N/cm ²)
Inlet Pressure	
Ramjet/Convertible Scramjet (Subsonic Combustion)	150 psi (103.2 N/cm ²)
Scramjet (Supersonic Combustion)	100 psi (68.9 N/cm ²)
Turbojet	60 psi (41.3 N/cm ²)
Landing Speed	200 kts (103 m/sec)
Sink Speed	20 fps (6.08 m/sec)
Tank Pressure	
LH ₂	10 psig (6.9 N/cm ²)
LO ₂	25 psig (17.2 N/cm ²)
JP	15 psig (10.3 N/cm ²)
Propellant Temperature	
LH ₂	30°R (16.65°K) Subcooled
LO ₂	163°R (90°K) Normal Boiling Point
JP	530°R (294°K) Room Temperature
Tankage	Integral, Internal Insulation
Structure	250°F max (395°K) Insulated, 300°F max (421°K)
Thermal Protection	
Surface	Passive insulation
Inlet for Mach 6	Passive insulation
Inlet for Mach 7 and up	Regeneratively cooled
Allowable Heat into Fuel	100 Btu/ft ² -hr (315 watt/m ²)
Fuel Reserves, Isp Reduction	
Rocket	1%
Airbreather	5%
Program	
Time	5 yrs
Vehicles	3
Ullage	
Wing Body	1%
All Body	2.5%
Crew	1
Fuel	Hydrogen, JP-4
Max Angle of Attack, Low Speed	15°

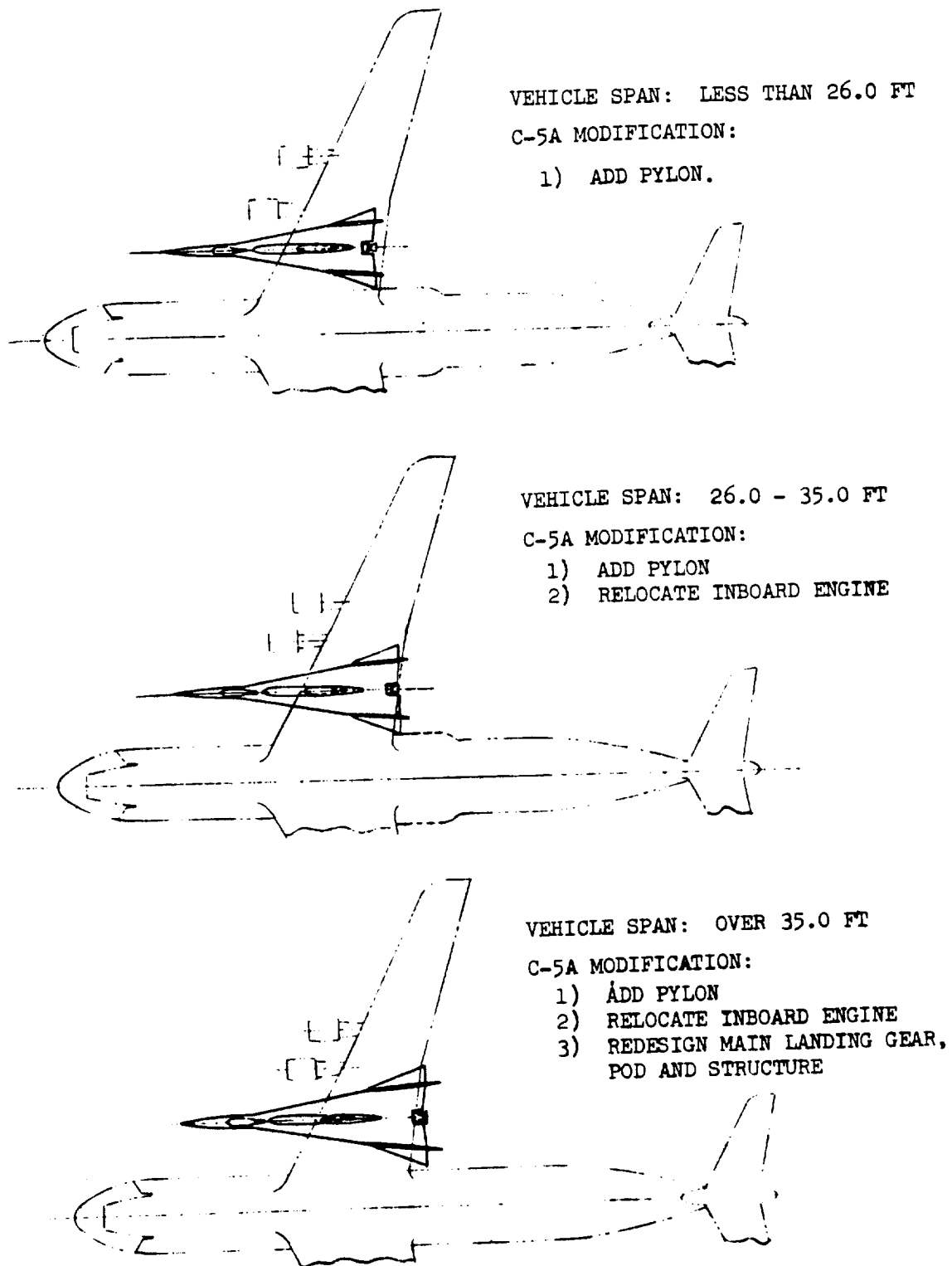
4.1.2.2 (U) Rocket Accelerator - Single Base Operation - Turnaround is accomplished by performing an unpowered 3.5g windup turn initiated at end of 5 minute test Mach number cruise. (Initially in the turn the g load is somewhat less than 3.5g because angle-of-attack is restricted to 20°.) The return phase of flight was at the Mach number attained at end of turn: 2.5 for Mach 6 design vehicles and 5.0 for Mach 12 design vehicles. No glide range was allotted at end of Mach 2.5 cruise back; a 200 nmi (370 km) glide range was allotted at end of Mach 5.0 cruise back.

4.1.2.3 (U) Airbreather Accelerator - Two Base Operation - Climb profiles used for the Phase II studies are as presented in a later section (See Figure 4-30) and cruise is at $(L/D)_{max}$ at equilibrium altitude. Turbojets are fixed size engines (F100-GE-100) and are shut down at Mach 3. Ramjet engines are sized to provide thrust equal to drag at Mach 6 cruise when operating at minimum SFC. Ramjet sizes are then checked to assure adequate full power thrust to accelerate the aircraft from the Mach 3 shutdown of the turbojet engines to Mach 6 cruise. Scramjet engines are sized to provide the maximum capture area possible without ingesting the vehicle bowshock or any inlet ramp shocks. This results in capture areas which average about 4.5% of the vehicle planform area. Ramjet and convertible scramjet engines are ignited at Mach 1 and remain on for duration of powered flight.

(U) Propulsion System Selection - In selecting propulsion systems for the Phase II vehicles, "off-the-shelf" engines were chosen wherever such engines could perform the desired mission. Minor modifications of these engines were accepted. The criterion for defining "off-the-shelf" engines was that they reach PFRT status by 1975, without funding beyond that currently committed or planned. This PFRT status for turbojets, turbofans, and rockets is taken to be PFRT as currently practiced (typified by specifications such as MIL-E-5009D). For ram-compression engines (RJ, SJ, and CSJ) no MIL standards exist; thus a PFRT criterion judged applicable to an experimental, rather than operational, vehicle was employed. This criterion required approximately five cycles of maximum power at conditions corresponding to the nominal aircraft acceleration flight path, plus five cycles of cruise time (of five minutes each) at the cruise Mach number and altitude. In all of the Phase II propulsion systems except turbojets, hydrogen was the only fuel considered because of Phase I study results. For turbojets JP fuel was used, based on a tradeoff study made between hydrogen and JP fuel.

4.1.3 (U) C-5A (MOTHER SHIP) LIMITATIONS AND MODIFICATIONS - C-5A limitations for this study are based on the C-5 Airframe Structural Description (LG1S37-1-1) and C-5 Operational Planning Manual (MER 400A) reports. Wing mounted air launched vehicles and suspension pylon are restricted to a total height of 18 feet (5.5 m). Research vehicles would be mounted between the fuselage and the inboard engine nacelle of the C-5A, Figure 4-4. For vehicles with a span of less than 26 feet (7.9 m), no C-5A modifications appear to be required. For spans over 26 feet (7.9 m), the C-5A right hand inboard engine would have to be relocated outboard of its present position. For wing spans over 35 feet (10.6 m), a major modification of the C-5A would be required in addition to the engine relocation. This includes modifying the landing gear and fuselage structure to eliminate the landing gear pod protrusion. The wing is capable of carrying shear loads in excess of 200,000 pounds (889,600 N) during taxi conditions. Since the research vehicles weight less than 100,000 pounds (444,800 N), no major revisions are anticipated for wing strength. Cruise altitude of the C-5A at launch is assumed to be 35,000 feet (10,670 m).

(U) FIGURE 4-4 C5A MODIFICATION REQUIREMENTS

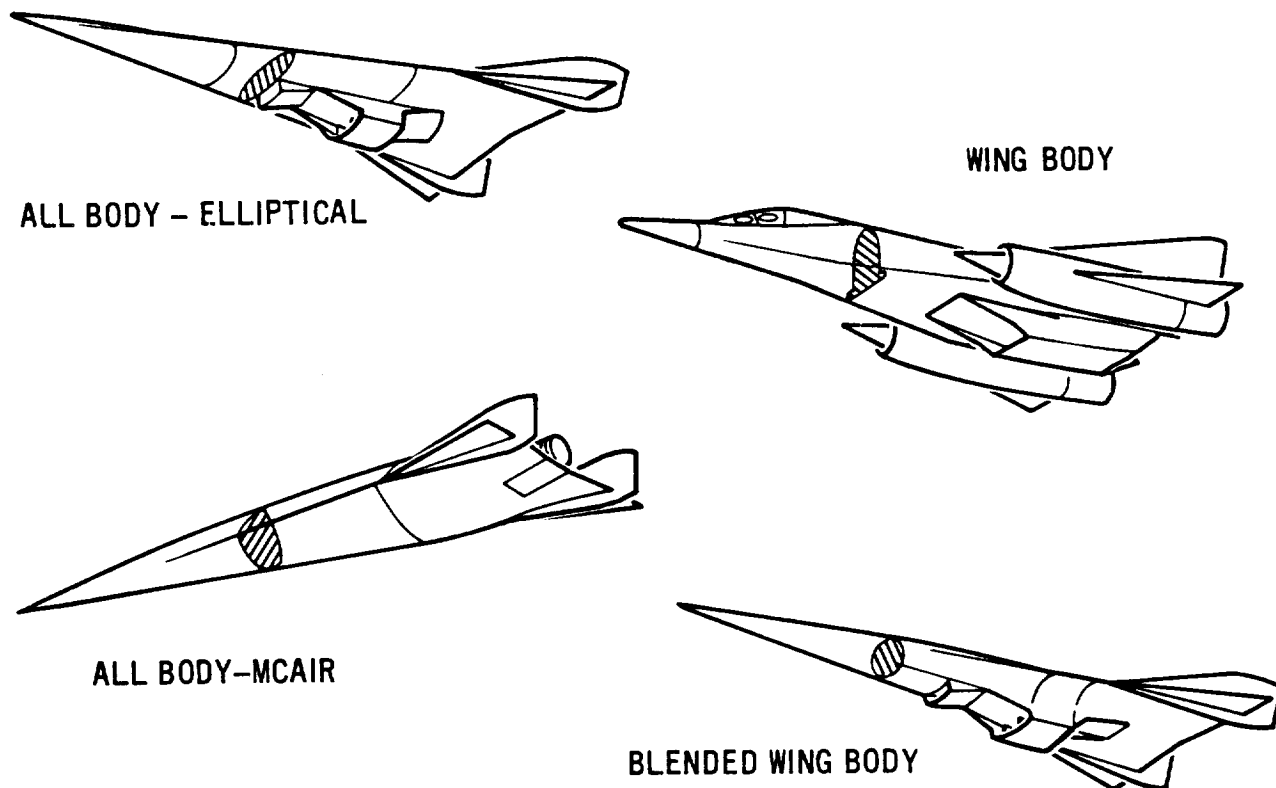


4.2 TECHNICAL APPROACH AND METHODOLOGY

(U) The baseline Phase II aircraft are refinements of the seven aircraft configurations (Figure 4-1) retained from the Phase I screening. The following section describes the technical approach and methodology used in this refinement, and in the parametric tradeoff studies (Figure 4-2) performed during this phase. Also presented are the results of additional study efforts which were necessary to support the parametric tradeoffs.

4.2.1 (U) CONFIGURATION DESIGN - The three basic Phase I configuration shapes were retained for Phase II. These configurations were refined to provide more efficient utilization of the internal volume of each vehicle. A fourth configuration shape (Blended Wing-Body) was developed during Phase II for the airbreather accelerated, Mach 12 concept which provided improved performance over the all-body shape. These four configurations are illustrated in Figure 4-5.

(U) FIGURE 4-5 VEHICLE CONFIGURATIONS



HRF-183

4.2.1.1 (U) All-Body Elliptical - The elliptical all-body shape was established by using a 4 to 1 ratio ellipse at the maximum cross section point as the control for the basic body size. The vehicle shape was then achieved by fairing from the maximum cross section to the other controlling design parameters, listed

(a) (U) Nose - A standardized nose shape was used on all hypersonic vehicles in this study, with the maximum cross section controlled by minimum dimensions for fixed wing aircraft cockpits as established by MIL-STDs MS33573, MS33574, MS33575, and MS33576. The nose was also shaped to minimize adverse effects on aircraft stability and on the first ramp for scramjet.

(b) (U) Forebody Ramp Angle - The Hypersonic Scramjet Vehicle Study, Reference (1), determined that the conical underbody extending from the aircraft nose to the second inlet ramp of the scramjet should form an angle of 3° with the aircraft horizontal reference plane. This 3° forebody ramp angle was found to produce a "near optimum" fineness ratio for turbojet accelerated aircraft where some volumetric efficiency must be sacrificed to achieve satisfactory transonic acceleration. While that study was conducted for operational vehicles, it appeared reasonable to adopt these results for the research vehicle.

(c) (U) Aft Fuselage Underbody - The asymmetric nozzle geometry necessary to provide the scramjet exhaust flowfield characteristics that maximize the engine thrust and propulsive lift for operational size vehicles was developed in Reference (1). This geometry appears reasonable for research size vehicles and was therefore adopted for HYFAC.

4.2.1.2 (U) All-Body - MCAIR - The MCAIR all-body shape is the result of shape optimization studies in Reference (1). It has a delta planform and a modified rhombic cross section, with a width-to-height ratio of 1.7 to 1, optimized to achieve a high volume-to-wetted-area and planform fraction. Other design controlling parameters for this shape are:

(a) (U) Nose - Identical to that used on the elliptical all-body shape.

(b) (U) Forebody Ramp Angle - Established at 4° in Reference (1) as "near optimum for rocket accelerated hypersonic vehicles. The resulting lower fineness ratio is acceptable for rocket vehicles which characteristically do not present transonic acceleration problems.

(c) (U) Aft Fuselage Underbody - Shaped to scramjet requirements, as on the elliptical all-body shape.

4.2.1.3 (U) Wing Body - The wing body shape was initially developed in Reference (1) as a high fineness ratio vehicle optimized for turbojet acceleration, and has been the basis for the HYFAC airbreather configuration work in the Mach 6 to 8 speed regime during Phase II.

4.2.1.4 (U) Blended Wing Body - During the early portion of Phase II considerable difficulty was experienced in achieving converged design solutions to the Mach 12, turbojet/convertible scramjet concept (B257). Although the all-body shape was capable of the required performance, the blended wing body shape was introduced into the study to provide added visibility into the effect of cross sectional shape on mission capability and vehicle size.

(U) This shape is essentially the same as the wing body shape studied in Phase I with the addition of larger fillets at the wing/fuselage juncture. These fillets permit an increase in fuel tankage, creating a better fuel-volume-to-planform ratio. The converged design achieved with this approach resulted in its choice as the baseline shape for parametric studies of Configuration B257.

4.2.1.5 (U) Additional Study Effort - Two integrated turbojet/convertible scramjet arrangements, illustrated in Figures 4-6 and 4-7, were evaluated during Phase II.

(U) Figure 4-6 is a design layout of the first candidate propulsion system installation which satisfies the requirements of pressure recovery in the turbojet system and variable capture area for the accelerating convertible scramjet.

(U) In this design, the convertible scramjet can be retracted to eliminate cooling requirements during descent. It is capable of wide variations in capture area to maintain a "shock-on-lip" condition from Mach 8 to 12. A boundary layer diverter, incorporated forward of the turbojet inlets, retracts flush with the forebody during the scramjet mode. The turbojet capture area requirements are met by employing a scoop-type inlet, with side plates to reduce spill and a variable geometry throat.

(U) Convertible scramjet capture area requirements and forebody ramp angle requirements establish the forebody length, while the nozzle requirements dictate the afterbody length. The resulting aft location of the scramjet propulsion package on the delta planform creates a nominally tail-heavy design. Variable capture area for the CSJ is accomplished by moving the engine duct further out from the fuselage (See Figure 4-58). Inlet and nozzle ramps are hinged to maintain surface continuity as the CSJ is translated throughout the flight. In this manner, variable geometry is accomplished with no movement of regeneratively cooled surfaces relative to each other.

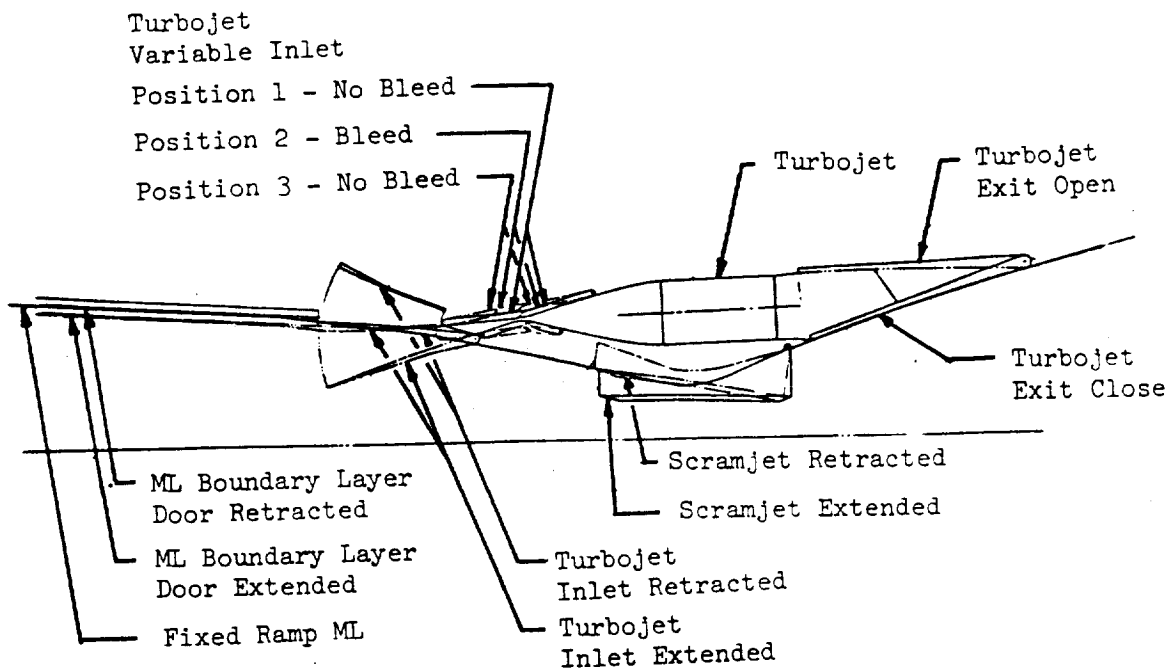
(U) The turbojet and convertible scramjet inlets operate with relative independence although the turbojet scoop will produce some wake effects on the scramjet below Mach 3. (At Mach 3 the turbojet is shut-off and the inlet and boundary layer control are closed.) Capture area and normal shock location are therefore controlled for each engine independently.

(U) Weight and balance requirements of the aircraft are easily met due to the selected location of the turbojet above the convertible scramjet. In addition, turbojet performance is maximized by the relatively straight flow path through the aircraft.

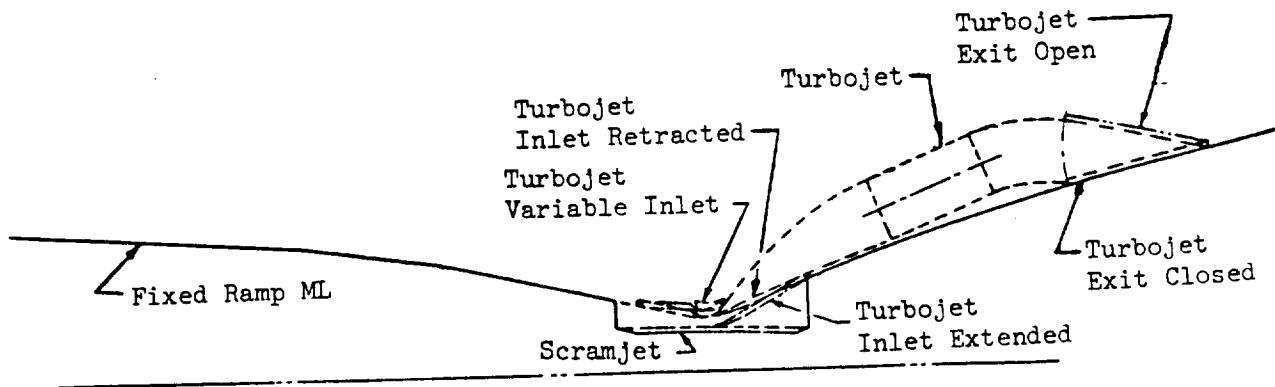
(U) Previous study efforts, including those of other investigators, to integrate the turbojet inlet with the scramjet inlet, have resulted in a second candidate design like the one shown by Figure 4-7. These aircraft designs are difficult to balance, exhibit performance penalties and system integration problems.

(U) Location of the turbojet engines within the Mach 12 vehicle moldline requires added inlet length between the inlet and the engine face both for recovery and uniformity of airflow. In addition, the turbojet exhaust must be properly vectored away from the vehicle. These characteristics tend to drive the location

(U) FIGURE 4-6 INTEGRATED INLET TURBOJET/CONVERTIBLE SCRAMJET



(U) FIGURE 4-7 COMMON INLET TURBOJET/CONVERTIBLE SCRAMJET



of the turbojet aft and tend to increase the individual component weights, all of which tend to shift the c.g. further aft, making acceptable aircraft balance more difficult to achieve.

(U) Major propulsion considerations in such a concept are the high turning angles required in both the inlet and nozzle ducts for the turbojets, leading to lower recovery efficiencies. Additionally, the complexities of maintaining the proper airflow to both the turbojet and the convertible scramjet between Mach 1 and Mach 3 are formidable. The inlet system operates in a started mixed compression mode over most of this speed range. The normal shock in the TJ diffuser is controlled by varying wall geometry and by a variable bypass system, while in the CSJ diffuser the normal shock is controlled by regulating combustor fuel injection to control heat addition. Interaction between the two diffusers significantly compounds the magnitude of this always difficult control problem.

(U) Of major importance in system integration of the engines shown in Figure 4-7 is the necessity of sliding regeneratively cooled panels relative to each other. Although not shown in Figure 4-7, a boundary layer diverter such as shown in Figure 4-6 would probably be required which would add further to the complexity of this design concept.

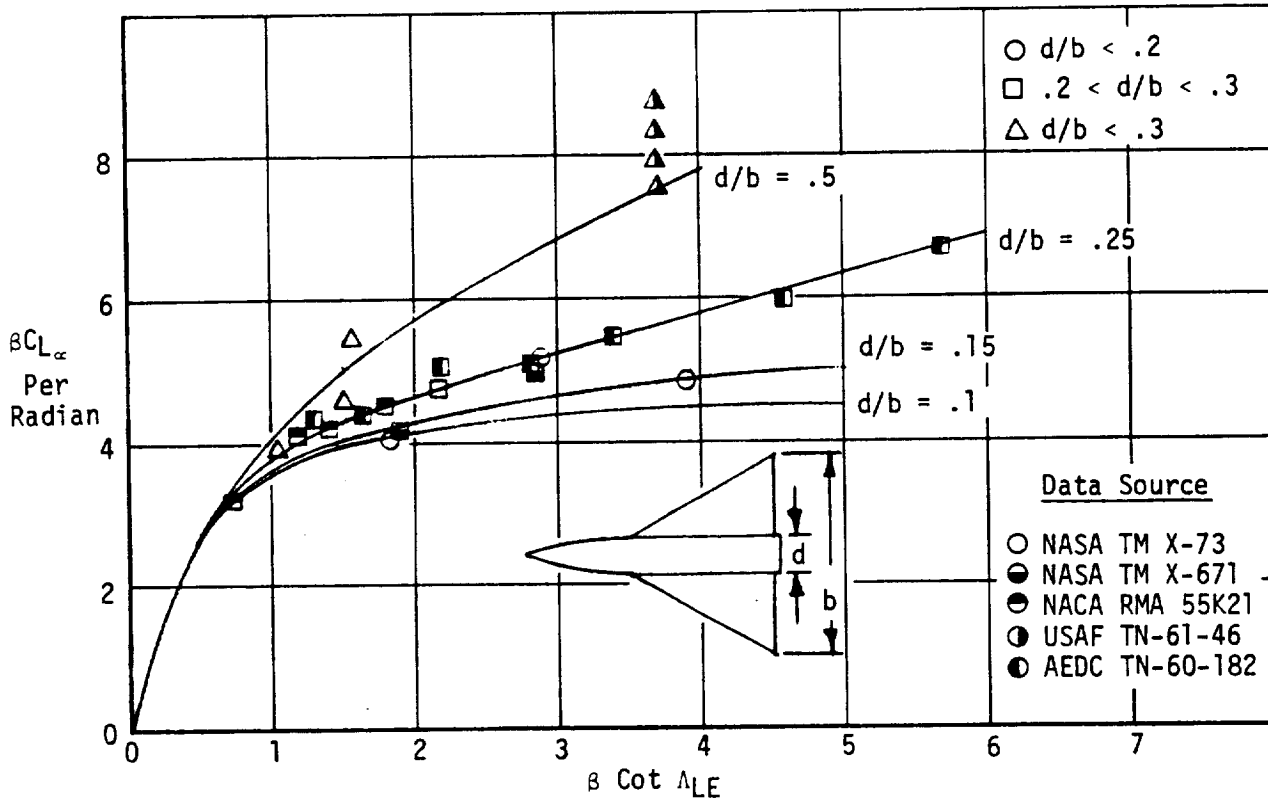
(U) Selection of the inlet system was based on the factors just discussed and one additional consideration. An over-turning pitching moment is generated due to the vertical offset between the inlet and exhaust vectors of the turbojet engines. In this regard the through flow design shown in Figure 4-6 is easier to balance and therefore is more suitable for further investigation. The design shown in Figure 4-6 was selected for the research vehicles.

4.2.2 (U) AERODYNAMICS - The theoretical lift and drag prediction techniques employed in this phase of the study are discussed herein. Phase II efforts in this regard are somewhat more refined than those employed in Phase I (Volume II), particularly insofar as the drag analysis is concerned. In Phase I, vehicle drag was determined as a function of maximum lift-to-drag ratio, $(L/D)_{max}$, and the induced drag factor, L' , with zero lift drag, C_{D0} , a fallout. Because of the large number of configurations being screened, $(L/D)_{max}$ was estimated on the basis of simplified data correlations with vehicle volume and planform area. With the narrowing of the number of configurations under consideration in Phase II, individual lift and drag analyses were performed for each vehicle design utilizing the methods and techniques described in the following paragraphs.

4.2.2.1 (U) Lift Estimation Method - The lift characteristics were calculated from data correlations and theory, as follows. The subsonic lift curve slope $C_{L\alpha}$, was determined using the method of Reference (2). The supersonic lift curve slopes were obtained from the correlation of data on delta wing configurations shown in Figure 4-8. These values of $C_{L\alpha}$, were adjusted to account for the effects of taper ratio and trailing edge wing sweep, where appropriate, using design charts. Linear lift curves were assumed at all Mach numbers.

(U) For all-body configurations, an equivalent d/b ratio was used where b is the overall span of the vehicle and d is the equivalent diameter of the vehicle, based on its maximum cross-sectional area. For double delta wings, an effective leading edge sweep, based on an area-weighted average, was used to determine the lift curve slope from Figure 4-8.

(U) FIGURE 4-8 LIFT CURVE SLOPE OF DELTA WING-BODY COMBINATIONS



4.2.2.2 (U) Drag Estimation Method - Drag characteristics were calculated using the component build-up method. This method has evolved from theoretical calculations, experimental data, and correlations with experimental data. The total drag coefficient is given by:

$$C_{D_{TOTAL}} = C_{D_0} + L' C_L^2$$

Where C_{D_0} is the drag coefficient at zero lift, and $L' C_L^2$ is the incremental drag coefficient due to lift.

4.2.2.2.1 (U) Zero Lift Drag - The zero lift drag coefficient is composed of skin friction drag, protuberance drag, wave or pressure drag, base drag, and propulsion system drag.

(U) The Von Karman-Schoenherr equation is used to calculate the skin friction drag along the flight trajectory. For incompressible flow, the smooth flat plate skin friction is given by,

$$\log_{10} (R_N C_f) = 0.242 / (C_f)^{1/2}$$

The corrections of Reference (3) were applied to account for Mach number and temperature effects. For Mach numbers equal to or less than 5.0, the temperature correction corresponds to adiabatic conditions (T_w/T_r). For Mach numbers greater

than 5.0, the correction is an equivalent temperature ratio for an entire aircraft. This correction was obtained by dividing the aircraft into equivalent wedges and computing local values of T_w/T_r , using the methods outlined in Reference (4). These local values of T_w/T_r were then area-weighted to give an equivalent T_w/T_r for the entire aircraft.

(U) The thickness factors derived by Hoerner, Reference (5), were used at subsonic speeds to correct the flat plate skin friction drag coefficient to account for changes in local velocity along the wing body:

$$\frac{C_{F_{Wing}}}{C_{F_{Flat Plate}}} = 1 + 2 (t/c) + 60 (t/c)^4$$

$$\frac{C_{F_{Body}}}{C_{F_{Flat Plate}}} = 1 + 1.5 (d/l)^{3/2} + 7 (d/l)^3$$

According to two-dimensional theory, this additional viscous drag does not exist at supersonic speeds.

(U) Estimating the total drag due to unavoidable protuberances can be accomplished only with a detailed knowledge of the aircraft; however, statistical information reveals that these items on recent fighter aircraft account for a $C_{DF} = 0.00065$, based on total wetted area. Additional information confirms the subsonic level of 0.00065 and suggests that a value of $C_{DF} = 0.00085$ be used at supersonic speeds which was used for Phase II drag calculations.

(U) The transonic and supersonic wave drag coefficients of the basic aircraft components (fuselage, nacelles, wing, vertical tails, and horizontal tails) were estimated using the component build-up method. This method has several advantages. The fuselage components can be defined directly from the area distribution, and the best available correlation method for each component and Mach region can be used. That is, at transonic speeds where theoretical treatments are inadequate, data correlations can be used. For simple shapes at supersonic speeds, tangent wedge and tangent cone prediction techniques can be employed, and for complex shapes, linear theory can be utilized. The disadvantage of this method is in the inability to account for mutual interference effects between components in other than a general way. The only method commonly available, which includes interference effects, is the "area rule" method. Experience has shown that while there are advantages to be gained by using the "area rule" method, the results are usually not significant enough to influence preliminary design configurations. For this reason, the "area rule" method was not employed in Phase II, but will be used in Phase III to confirm basic design drag levels.

(U) Experimental data correlations for varying nose shapes, cone frustrums, and boattail configurations were employed to determine wave drag contributions for the nose, mid/fuselage, and aft body at transonic and supersonic speeds. Similar data correlations developed for boundary layer diverters were utilized to define the pressure drag contribution produced by these devices. The wave drag contributions of the wing and tails were determined from transonic data correlations and supersonic linear theory.

(U) Base drag coefficients were determined from the correlation shown in Figure 4-9. Three dimensional base drag contributions were included on the expansion nozzles of ramjet and scramjet configurations for engine-off operations.

(U) The drag attributed to the propulsion system during engine-on operation is composed of ram drag, spill drag, and bleed drag. These are accounted for as a reduction in the thrust available as described in Section 4.2.4. In all instances, external cowl drag is included in the aerodynamic drag analysis.

4.2.2.2.2 (U) Drag Due to Lift - The induced drag factor, L' , was calculated as follows:

$$L'_{M>1} = \frac{1}{C_{L\alpha}} - K_1 \left[\left(\frac{1}{C_{L\alpha}} \right)_{M<1} - \frac{1}{\pi A R e} - K \right]$$

where $L'_{M<1} = \frac{1}{\pi A R e} + K$

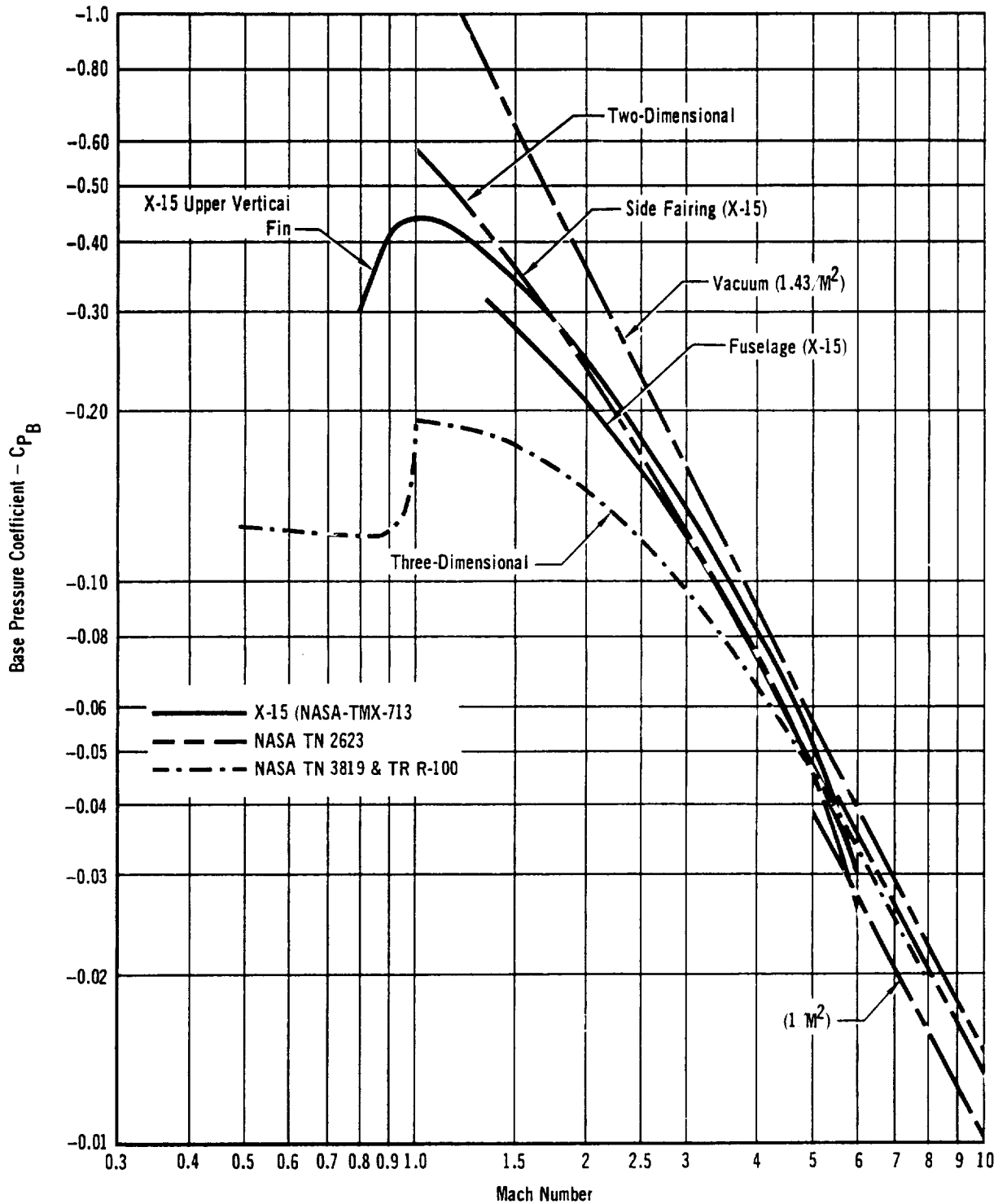
- o $C_{L\alpha}$ is the lift curve slope (Section 4.2.2.1)
- o $C_{L\alpha M<1}$ is the subsonic value of $C_{L\alpha}$ (Section 4.2.2.1)
- o A is the aspect ratio
- o e is Weissingers efficiency factor (Reference 6)
- o K is an additional induced drag factor (Figure 4-10) which accounts for the effects of wing taper ratio and leading edge radius.
- o K_1 is a correction factor (Figure 4-11) employed to adjust the wing leading edge suction and additional induced drag factor terms, inside the brackets, for configuration shape, Mach number, and leading edge sweep effects.

For all-body configurations, K_1 is assumed to be zero, as noted in Figure 4-11, thereby neglecting the leading edge suction and additional induced drag factor terms at all speeds. This reduces the calculation of L' for all-body shapes to

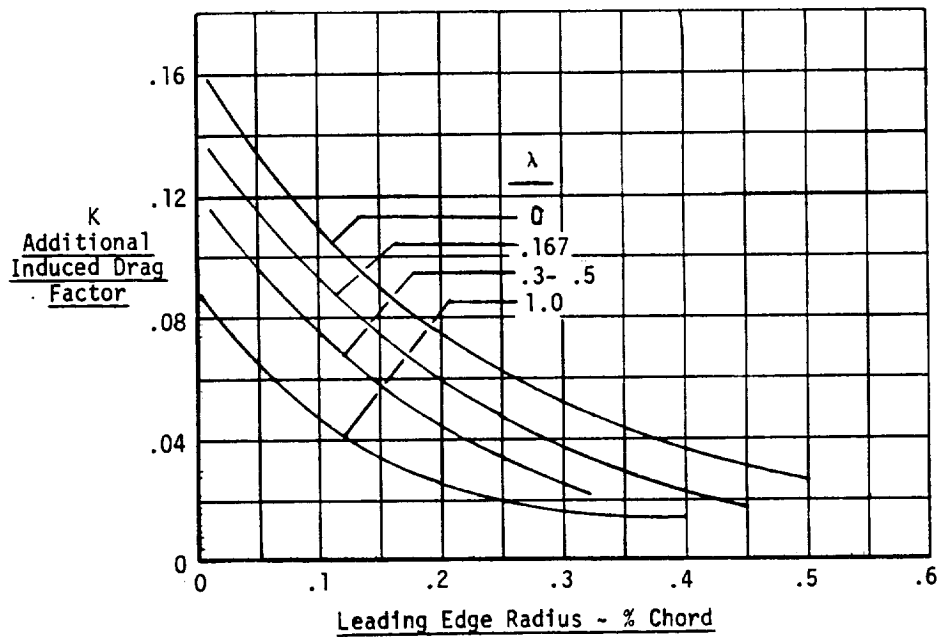
$$L' = \frac{1}{C_{L\alpha}}$$

which when combined with C_L and C_{D_0} yields an $(L/D)_{\max}$ value in good agreement with the aforementioned Phase I correlations of $(L/D)_{\max}$ with vehicle volume and planform area.

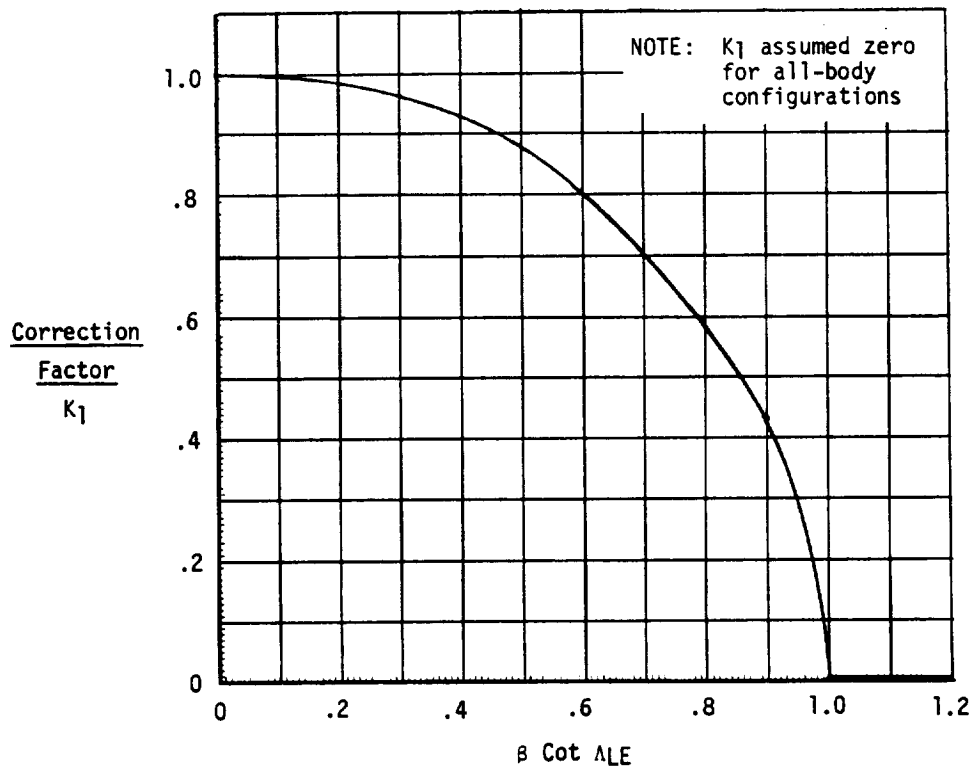
(U) FIGURE 4-9
BASE DRAG TURBULENT BOUNDARY LAYER



(U) FIGURE 4-10 ADDITIONAL INDUCED DRAG FACTOR (K)



(U) FIGURE 4-11 CORRECTION FACTOR vs $\beta \cot \Lambda_{LE}$



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4.2.3 (U) THERMODYNAMICS - During the Phase II vehicle tradeoff studies, primary effort was concentrated on defining vehicle changes, attributable to the thermal environment, which have a significant impact upon vehicle cost or performance. These can be categorized as follows:

- o Changes in moldline material selection and/or distribution
- o Changes in airframe insulation requirements for the structure and propellant tanks.
- o Changes in ramjet inlet Thermal Protection System (TPS) requirements.

Since the above vehicle changes are due to variations in the magnitude and/or duration of the aerodynamic heating environment, trade studies involving parametric variations of Mach numbers, test time, and load factor were examined. In addition, thermal analyses were conducted in support of the following tradeoffs:

- o Fuel - Normal Boiling Point (NBP) versus sub-cooled LH_2 (including slush).
- o Thermal Protection System (TPS).
 - (1) Insulated structure versus hot structure.
 - (2) Passive versus active TPS.
- o Tankage - Integral versus non-integral.

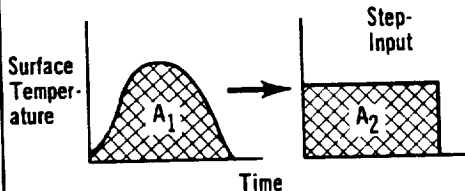
(U) Methodology used in conducting the analyses is summarized in Figure 4-12 and discussed below. The impact on vehicle size, weight, and cost are discussed in Section 4.4.

(U) Moldline Material Selection - Maximum external surface temperatures were determined for the various aircraft and missions under consideration to ensure the selection of moldline materials (shingles and insulation) with the appropriate temperature capability. Upper surface temperatures were computed at an end-of-boost/cruise insertion condition, that is, at cruise velocity and altitude, but at an angle-of-attack of zero degrees. Lower surface temperatures were computed at the cruise velocity and altitude, but at the assumed maximum angle-of-attack of 20 degrees. This latter condition is characteristic of a 3.5g maneuver which has been established as the design load factor at V_{max} conditions (see Section 4.4.4.1).

(U) Airframe Insulation Requirements - The baseline thermal/structural concept utilized during Phase II consists of:

- (a) (U) Radiation cooled external shingles which provide a smooth aerodynamic surface, transmit air loads via attachments to the primary structure, and protect the insulation from viscous shearing forces;
- (b) (U) Passive insulation which protects the primary structure from the external heating environment;
- (c) (U) Insulated load carrying aluminum structure (2 psf, 0.9 kg/m^2); and

(U) FIGURE 4-12
METHODOLOGY USED IN CONDUCTING
THERMODYNAMIC ANALYSIS

Vehicle Design	Driving Thermodynamic Variable	Analysis Conditions	Method of Analysis
Moldline Material Selection	Maximum Surface Temperatures	Maximum Use Temperature, °F (°K) a) Shingles Titanium, 1100°R (866°K) Rene'41, 1550°R (1120°K) TD NiC, 2400°R (1590°K) Columbium, 2800°R (1810°K) b) Min-K, 1800°R (1260°K) Dyna-Flex, 2700°R (1760°K)	(1) <u>Aerodynamic Heating Rates</u> • Stagnation - Fay & Riddell • Non-Stagnation - Turbulent Heating Correlations of Eckert (Mach 12) and Spalding and CHI (Mach 6) (2) <u>Surface Temperatures</u> • Radiation Equilibrium • 0.8 Emissivity
*Airframe Insulation Requirements *Based upon Requirements at Four Typical Locations, i.e., Lower and Upper Surface, Fuel and nonfuel areas.	Surface Temperature Histories	<u>LH₂ Fuel Tank</u> a) Lower Surface • 50 to 150°F (283 to 339°K) Structural Temperature Rise • Heat Leak to LH ₂ of of 100 $\frac{\text{Btu}}{\text{ft}^2 \text{ hr}}$ $\left(315 \frac{\text{watts}}{\text{m}^2} \right)$ b) Upper Surface • 50 to 250°F (283 to 394°K) Structural Temperature Rise • No Heat Transfer to LH ₂ <u>Non-Fuel Areas</u> • Upper and lower - 100 to 300°F (311 to 422°K) Structural Temperature Rise	(1) Surface Temperatures - Same as Above (2) Effective Step-Input Temperature - Equivalent to Area under Surface Temperature versus Time Curve, i.e., $A_1 = A_2$.  (3) Employ Two-Layer Plate Solution for One-Dimensional Heat Transfer (Reference 7) to size Insulation
Inlet Thermal Protection System (TPS) Requirements	<u>Uncooled Inlet</u> • Inlet Temperature Histories <u>Cooled Inlet</u> • Inlet Heating Rates	• Maximum Inlet Structural (Titanium) Temperature of 400°F (479°K) • Cooled Inlet Wall Temperature of 1600°F (1144°K)	<u>Uncooled Inlets</u> (1) Uncooled Inlet Wall Temperature Assumed equal to total Temperature (2) Insulation Thicknesses per Finite-Difference Computer Analysis <u>Regeneratively Cooled Inlets</u> (1) Heating Rates per Eckert's Turbulent Heating Expression (2) Coolant Flow Rates Based on a Heat Exchanger Efficiency of 80% (Reference 8.)

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- (d) (U) Integral tankage (aluminum) with internal cryogenic foam insulation to limit the heat transfer to cryogenic propellants.

(U) To determine overall airframe insulation, four representative locations were analyzed in detail, that is, a representative upper and lower surface location for both fuel and non-fuel areas. Local insulation requirements at these four locations were determined manually based upon an effective step-input of surface temperature (equivalent to area under actual temperature versus time curve, (see Figure 4-12) and a two-layer plate solution for one-dimensional heat transfer per Schneider (Reference 7).

(U) Insulation thicknesses in non-fuel areas were sized based upon an initial and final (maximum) structural temperature of 100°F (311°K) and 300°F (422°K), respectively. Passive external insulation requirements for upper fuel tank areas, likewise, were based upon the heat storage capacity of the aluminum structure only. That is, it was assumed that the internal tankage insulation was not wetted and hence the heat transfer to the fuel was negligible. Furthermore, because of temperature limitations of the cryogenic foam insulation, a maximum structural/bondline insulation temperature of 250°F (395°K) rather than 300°F (422°K) was used. External insulation requirements for lower surface fuel areas were determined assuming a 100°F (311°K) temperature rise of the tank wall plus a heat leak to the cryogenic hydrogen of 100 Btu/ft² hr (315 watts/m²). This heat leak was selected as a reasonable value for the short flight times of a research aircraft; the resultant final fuel temperatures and maximum tank pressure for a Mach 12 configuration are presented in Section 4.4.3 (Figure 4-99). Limiting the heat leak to the LH₂ fuel to 100 Btu/ft² hr (315 watts/in²) requires approximately 0.6 inch (1.52 cm) of cryogenic foam insulation.

(U) External insulation requirements for LOX tanks were arbitrarily set equal to LH₂ tankage values.

(U) Average passive insulation requirements as a function of Mach number and test time are summarized in Section 4.4.5.2. A comparison of active and passive requirements for the Mach 12 vehicle is discussed in Section 4.4.4.3.

(U) Ramjet Inlet TPS - Passive insulation is used both internally and externally to protect the titanium inlet structure (maximum temperature of 400°F, (478°K)) from aerodynamic heating effects. At Mach 6, aerodynamic friction results in external moldline temperatures of about 1000°F (810°K) and uncooled internal duct temperatures of about 2600°F (1700°K). Baseline Mach 6 aircraft were configured with uncooled inlets and thus the TPS was sized based upon the above internal duct temperature of 2600°F (1700°K).

(U) At flight speeds in excess of Mach 6, internal duct temperatures exceed the temperature capability (assumed to be 2700°F (1760°K)) of available load-bearing insulation materials and hence inlet wall temperatures were cooled to 1600°F (1144°K). To maintain consistency in conducting the Mach number trade study (see Section 4.4.5.1), regenerative cooling was also employed at Mach 6 as well as the higher Mach numbers considered, namely, Mach 7 and 8.

(U) Regenerative coolant flow (LH₂ fuel) requirements to maintain a 1600°F (1144°K) duct wall temperature were determined based upon turbulent aerodynamic heating conditions and a heat exchanger efficiency (η) of 80%.

$$\eta = (T_{c2} - T_{c1}) / (T_w - T_{c1})$$

where: T_{c1} and T_{c2} are coolant inlet and exit temperatures and T_w is the wall temperature.

A heat exchanger efficiency of 80% was selected based upon past studies, e.g., Reference (8).

4.2.4 (U) PROPULSION - The study involved numerous propulsion systems utilizing various engines including turbojets (TJ), turbofans (TF), ramjets (RJ), scramjets (SJ), convertible scramjets (CSJ), and rocket engines. The scope of the propulsion approach was further broadened by the consideration of various inlets and exits as well as the number of basic engines.

(U) To configure the baseline propulsion systems and perform the tradeoffs, analytical methods and configuration selection techniques appropriate for the scope of the study were used. The methodology and baseline configuration selection processes are described below.

4.2.4.1 (U) Methodology - This section discusses the methods used to determine inlet performance including recovery and additive drag, engine performance, nozzle performance, and engine scaling.

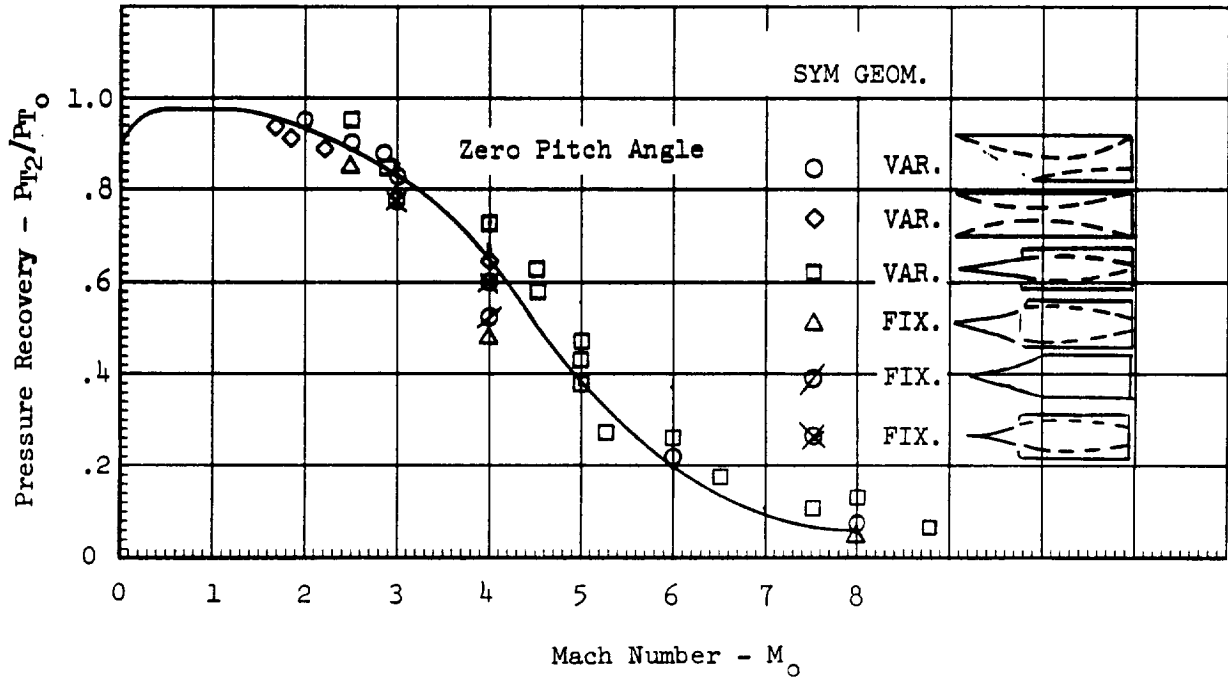
(a) (U) Inlet Performance - The two dominant parameters of inlet performance are (1) total pressure recovery, P_{T2}/P_{T0} , and (2) inlet drag.

(U) Pressure recovery is classically determined from analysis of the flow field or from test data. Analysis requires vigorous definition of the interacting viscous boundary layer and the primary inviscid flow field and a detailed definition of the inlet geometry and bleed system.

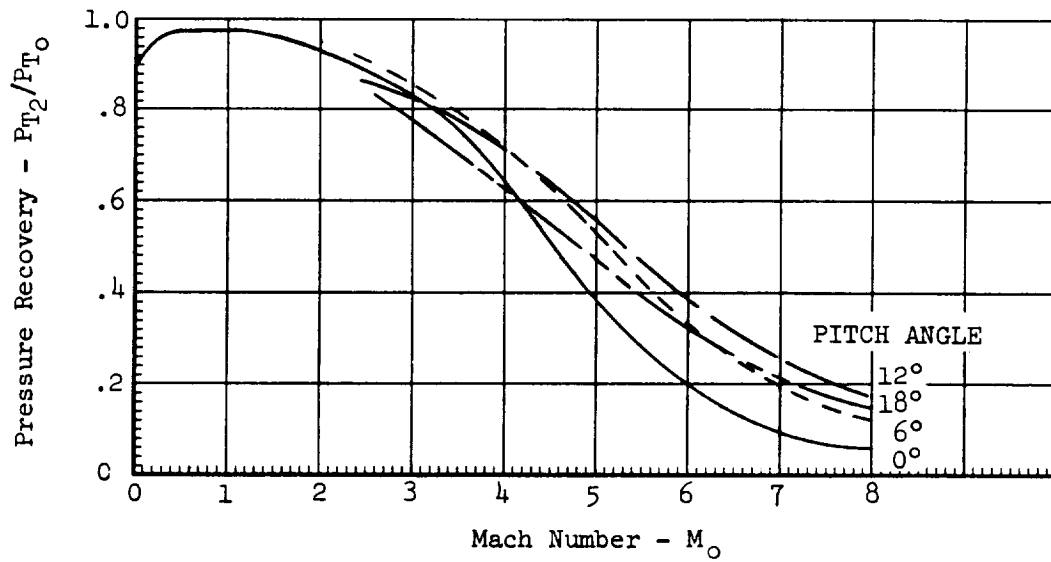
(U) For the TJ/TF and RJ inlets, in lieu of an analytical evaluation, pressure recovery was based on a compilation of experimental inlet test data which is representative of the inlets used on the research vehicles. Forebody pressure field effects and the pitch angle effects were included. The TJ/TF and RJ inlet pressure recovery data with angle-of-attack effects are shown on Figures 4-13 and 4-14.

(U) For the SJ and CSJ inlets, test data were not available. To define recovery levels representative of such designs, pressure recovery was estimated using inviscid, ideal gas analysis of the forebody and inlet shock structure. In this analysis, the shock-structure estimation incorporates an equivalent cone (forebody half angle plus angle-of-attack) forebody shock. The average Mach number in the captured stream tubes, downstream of the forebody shock, was estimated and the subsequent compression ramps were treated as two-dimensional with linear shock theory. The effects of boundary layer friction drag on inlet efficiency were estimated by means of the analytical model presented in Reference (9). In this method the momentum decrement of the boundary layer flow (relative to the inviscid flow field) is accounted for as a reduction in inlet efficiency. The SJ and CSJ inlet recovery data are shown in Figure 4-15.

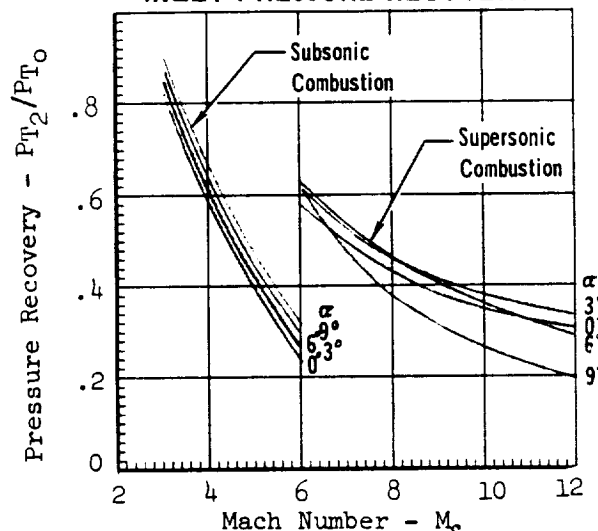
(U) FIGURE 4-13 TJ AND RJ INLET PRESSURE RECOVERY SOURCE



(U) FIGURE 4-14 TJ/TF AND RJ INLET PRESSURE RECOVERY Sensitivity to Pitch Angle



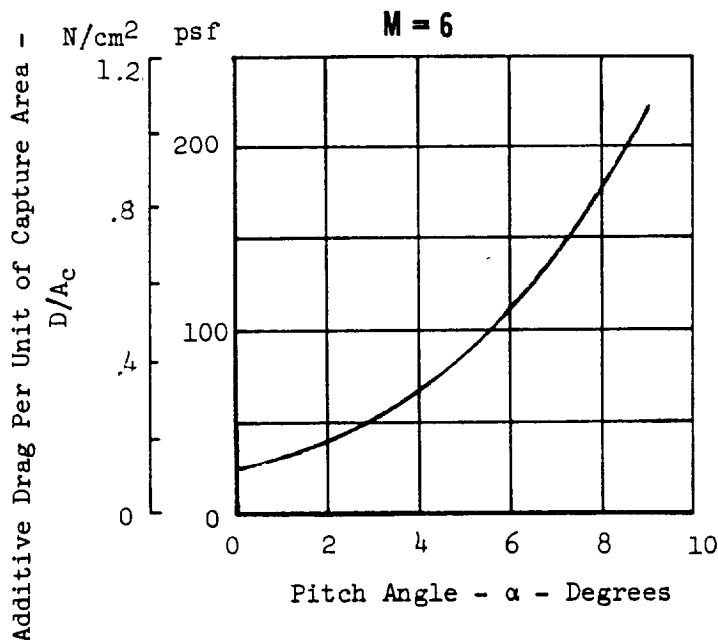
(U) FIGURE 4-15 SCRAMJET AND CONVERTIBLE SCRAMJET
INLET PRESSURE RECOVERY



(U) For all inlets, three elements of inlet drag were estimated:

- o Ram Drag - The stream thrust of the air captured by the inlet. This drag was estimated as equal to the momentum flux of the captured air.
- o Supersonic Spill Drag - The integrated drag force along the limiting streamline of the captured air, caused by the inlet not operating with shock-on-lip. This drag was estimated by predicting the flow field characteristics and integrating the pressure over the associated area, assuming two-dimensional inviscid flow fields. A typical result for the convertible scramjet vehicle is shown in Figure 4-16.

(U) FIGURE 4-16 CONVERTIBLE SCRAMJET ADDITIVE DRAG



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- o Bleed and Leakage Drag - The ram drag force associated with capturing the air which is bled from the inlet ramps to maintain proper flow on the ramps, and the air which is lost by leakage. An allowance of 10% of the total captured air was allotted for bleed and leakage, and was reflected in inlet sizing.

(b) (U) Engine Performance - The turbine engine (turbojet and turbofan) and ramjet engine performances were taken from engine specifications of uninstalled performance, provided by engine companies. These data were corrected for the vehicle inlet recovery and nozzle installation effects. In some cases, engine specifications were based on the use of JP fuels and corrections for the use of hydrogen were required. The design turbine inlet temperature of the basic turbine engines is the same for both H_2 and JP fuels. Therefore, no change in current turbine materials or cooling concepts was required. The SFC values for the H_2 turbine engines were obtained by ratioing the lower heating values of the H_2 fuel with that of JP fuel. The specific thrust (F_n/W_a) was corrected for the difference in fuel mass addition because of the differences in fuel air ratio at the same turbine inlet temperature level. Fuel flow for the ramjet was that required to cool the engine and nozzle, as determined by engine company specifications, plus that fuel needed to cool the inlet, as described in Section 4.2.3.

(U) SJ and CSJ engine performance were derived from a previous MCAIR study, Reference (1). That study used a one-dimensional cycle analysis program to define I_{sp} and F_n/A_c as functions of inlet, combustor, and nozzle efficiencies. Combustor efficiencies, representative of 1975 technology levels, were estimated, based on extrapolation of currently demonstrated values from several engine company tests.

(U) Engine-off drag was estimated for each of the airbreathing propulsion systems. For the vehicles with turbojets, doors were provided to close off the inlet. On the turbojet-ramjet vehicle the axisymmetric inlet configurations were provided with a petal-type expansion of the spike. On the turbojet-convertible scramjet vehicle the two-dimensional scoop ramp was pivoted to perform the closure. Drags of the closed inlets were estimated by procedures described in Section 4.2.2.2. For the ramjet inlet engine-off drag, an analysis developed in a previous MCAIR study, Reference (10), was used. That analysis assumed that air flows through the unlit RJ under these conditions:

- (1) Pressure recovery to nozzle entrance equal to MIL-E-5008 with an additional 6% loss.
- (2) Nozzle exhausts the air aft and parallel to vehicle axis, to an area equal to 85% of that required for full expansion to freestream static pressure.

These assumptions represent optimistic estimates of propulsion system performance based on limited test data.

(U) Rocket engine performance was taken directly from the engine company specifications, for the selected nozzle expansion ratio.

(c) (U) Nozzle Performance - Turbine engine internal nozzle performance, in terms of gross thrust coefficient, was taken from engine company specifications.

Installed nozzle boattail drag and/or base drag of the turbine engines was accounted for in the total vehicle drag analysis. Ramjet nozzle gross thrust coefficients were taken from the engine company specifications for the nozzle expansion area selected. Scramjet and convertible scramjet nozzle gross thrust coefficients were based on the amount of expansion associated with the individual research vehicle designs and the available nozzle pressure ratio. Chemical reactions were assumed to freeze in the SJ and CSJ nozzles at the area equal to 120% of the combustor exit area. Chemical kinetic analysis performed in a previous MCAIR study, Reference (1), indicated that this freeze point analysis produced performance estimates in agreement with the limited available test data. For the rocket nozzles, the expansion ratio chosen was the largest value that would not result in flow separation at maximum thrust, sea level operation. Engine company specification data were used to determine this separation limit.

(d) (U) Engine Scaling - The turbine engines and the RL10 (mod) and J2S rocket engines used in this study were used at full size as established by engine company specifications.

(U) Ramjets, scramjets, and convertible scramjets were scaled on the basis of capture area. The Marquardt MA 145 ramjet engine was used as a reference size for the ramjets. The SJ and CSJ engines were scaled from a reference size engine established in Reference (1). Scaling data for the LR 129 rocket engine was taken from the engine company specification, Reference (11).

(e) (U) Fuel Systems - The operational characteristics of a hypersonic aircraft fuel system are a function of engine requirements, thermal envelope, and fuel thermophysical properties. Tank outflow, line sizes, and delivery pressure at the engine pump or pre-valve were defined by engine requirements. Maximum bulk fuel temperatures were determined by the heating profile combined with flow rates and residence time. Maximum allowable fuel supply temperatures and corresponding pressures were established at levels where acceptable minimum net specific suction pressures were supplied at the engine pump inlets.

(U) Cryogenic fuels, especially liquid hydrogen, require specific attention to both local and bulk temperature/heating profiles. Without proper handling and insulation, the cryogen vaporizes readily and can lead to increased tank pressures and attendant structural weight increases.

(U) Fuel tank heating was studied to determine bulk propellant temperatures and vapor pressures at the end of the flight. The fuel vapor pressure (temperature dependent), in addition to pump inlet pressure, establishes minimum tank pressurization requirements. The effects of tank pressure levels on total pressurant gas weight and overall tank/structural weight were then evaluated. Design refinement studies were conducted to define the propellant initial temperature, flight heat leak, and operating pressure to assist in the selection of insulation, tank structure, and pressurization system for most efficient fuel tank and feed system design.

4.2.4.2 (U) Configuration Selection - The propulsion system selections were conducted in conjunction with the overall vehicle studies. Thus, mission performance, design arrangements, etc., were consistent for the baseline vehicle designs, providing a basis for accurate comparisons. This section describes the selection of the inlets, nozzles, and engines for the baseline vehicle configurations.

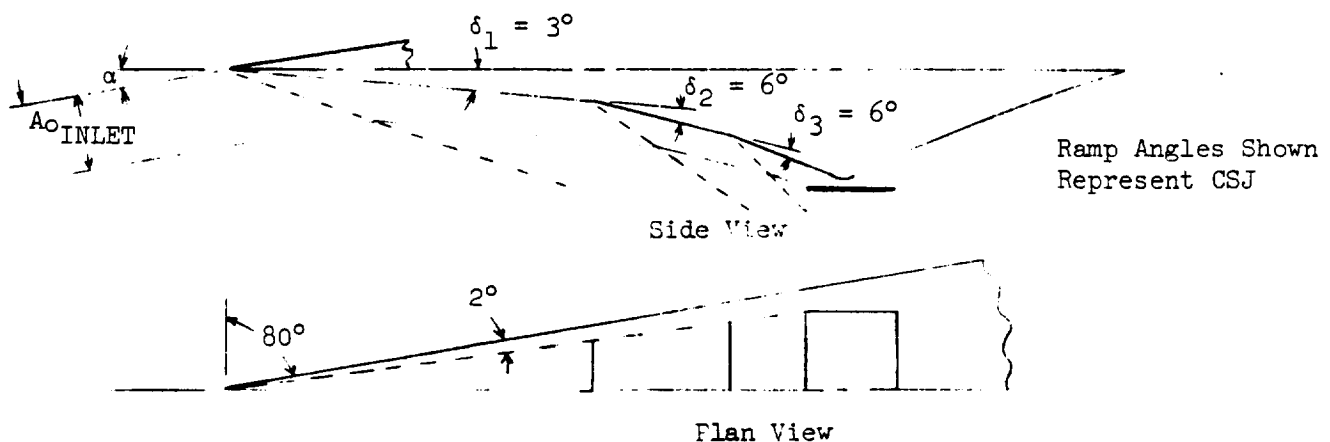
(a) (U) Inlet Selection - Inlet selections were based on performance analysis and limited model test data. The basic inlet types investigated were axisymmetric, semi-cone, vertical ramp two-dimensional (2-D), and horizontal ramp 2-D. These inlets can be integrated with research vehicles in various ways. The inlet configuration and installation, for the airbreathing engines of each baseline research vehicle, were selected on the basis of the following criteria:

- o Pressure recovery throughout the required operating range.
- o Practical installation of inlet on the airframe.
- o Inlet-engine integration.
- o Weight.

(U) The four flight research vehicle concepts incorporating airbreathers use three basic propulsion cycles: turbofans, ramjets, and scramjets or convertible scramjets. The Phase I vehicle developed for the Mach 6 rocket/RJ concept resulted in a very high operating weight empty (OWE) when using an axisymmetric RJ. To improve on this system for Phase II, a 2-D RJ was incorporated. For the 2-D RJ, only 2-D inlet designs were considered. The inlet was located within the forebody pressure field to obtain the benefit of the flow field on pressure recovery at the Mach 6 operating point.

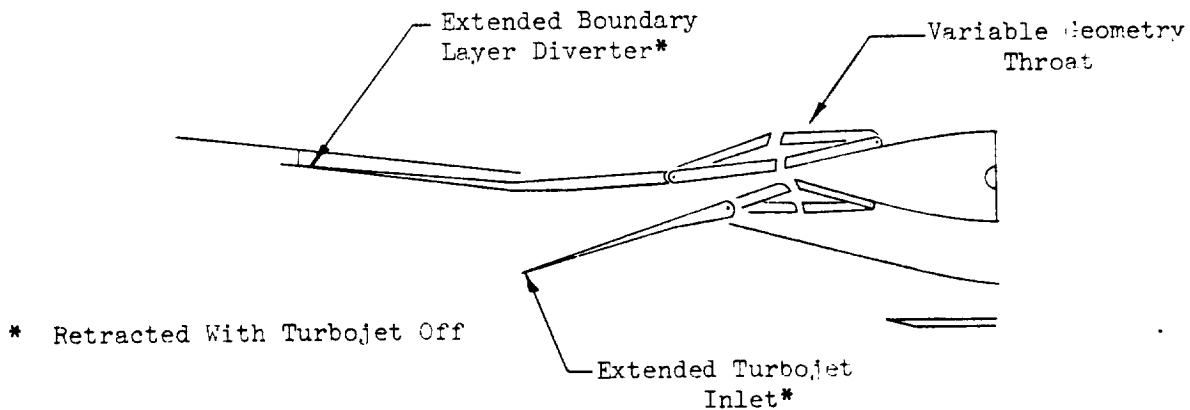
(U) For scramjet and convertible scramjet powered vehicles, previous studies such as Reference (1) have shown that the engine should be completely integrated with the airframe. In that study, the approach to achieving attractive scramjet cruise vehicles was developed in tradeoffs of numerous engine and installation concepts. In addition to indicating the need to integrate the inlet with the airframe lower surface, the study showed the advantages of inlet configurations that are essentially two-dimensional in shape. The research vehicles incorporate similar integrated inlet-combustor-nozzle-airframe scramjet designs. The resulting inlet design has the characteristics summarized in Figure 4-17. The turbojet accelerator inlet was also incorporated into the convertible-scramjet-powered research vehicle by the arrangement developed in Reference (1). This inlet configuration is shown in Figure 4-18.

(U) FIGURE 4-17 SCRAMJET AND CSJ INLET



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(U) FIGURE 4-18 TURBOJET INLET FOR CSJ CRUISE VEHICLE



(b) (U) Inlet/Engine Location - The B212 Mach 6 cruise vehicle incorporates two types of airbreathing engines, turbojet and ramjets. The most beneficial arrangement regarding inlet efficiency would involve mounting all these airbreathing engines in the lower, aft vehicle region. Sufficient room was not available in that region for all of these engines. Therefore a study was made to identify the best engine/inlet/nozzle arrangement. Since it was assumed that inlet pressure recovery did not differ among the various candidate arrangements, the primary considerations in the study were weight and engine-off performance. Length of the turbojet diffuser was used as a measure of weight differences. From the results summarized in Figure 4-19, the selected arrangement for this vehicle was a belly mounting for the 2-D ramjet with axisymmetric inlets for the wing-mounted turbojets.

(U) FIGURE 4-19 INLET/ENGINE LOCATION TRADE - TJ/RJ VEHICLE

		EVALUATION CRITERIA				
		Weight Difference	Controllability With One Engine Out		Pitch Angle Sensitivity	
			TJ	RJ	TJ	RJ
CONFIGURATION	RJ:2D, belly TJ:2D, wing	moderate	poor	moderate	good	excellent
	RJ:2D, belly TJ:AX, wing	small	poor	moderate	fair	excellent
	RJ:2D, wing TJ:2D, belly	moderate	moderate	worst	excellent	good
	RJ:2D, wing TJ:AX, belly	small	moderate	worst	fair-good	good

* Selected Arrangement

(c) (U) Nozzle Selection - Nozzle designs for the advanced propulsion cycles were based on concepts developed in engine company studies. The performance was based on analysis and limited model test data. For each propulsion cycle, certain types of basic nozzle shapes were considered to be appropriate:

- o Rockets: conventional bell.
- o Turbofan and turbojet engines: axisymmetric, with various means of changing throat and exit areas.
- o Ramjets (2-D): two-dimensional, with various means of changing throat and exit areas.
- o Scramjets and CSJ: basically 2-D, integrated with the vehicle lower surface contours.

The objective of this trade was to select the nozzle for each engine which best met these requirements:

- o Good gross thrust performance throughout the required operating range.
- o Compact integration with the airframe.
- o Minimum weight.

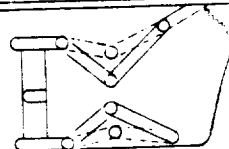
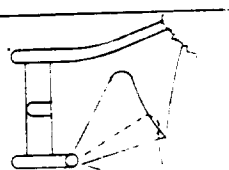
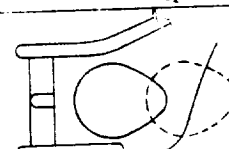
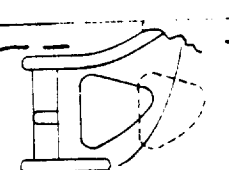
The considerations involved in the nozzle selection are described below.

(U) Three rocket engines were used with conventional bell nozzles, (their selection is described in the next subsection, (d)). For the rubberized rocket engines, scaled from the LR-129, an expansion ratio was selected as the maximum which should avoid nozzle flow separation at sea level maximum thrust operation. This expansion ratio, $\epsilon = 75$, was used for all applications of this engine. Two fixed size rocket engines were used, the J2S and the RL-10. The J2S nozzle, $\epsilon = 40$, precludes flow separation at sea level and was used unmodified for all applications. The RL10 (mod) nozzle, $\epsilon = 57$, does encounter nozzle flow separation throughout most of the lower flight envelope of the research vehicle. Therefore a shorter nozzle was selected, $\epsilon = 32$, to preclude separation at 35,000 feet (10.7 km), the lowest altitude for air launched application.

(U) The engine company specification nozzles were used with all the turbine engines. In the TJ/CSJ powered vehicle the turbofan engines were buried within the airframe. This installation did not affect the basic nozzle selection. However an exhaust duct and a closure door were incorporated into the airframe to seal off the TJ engine when it is not operating. The arrangement was developed in a previous study, Reference (1), for a similar vehicle.

(U) A trade study was made for the 2-D ramjet engine nozzle, to identify the most desirable configuration. Translating plugs and hinged flaps were considered. Internal performance was assumed to be equal for the various candidates. External drag and nozzle weight were qualitatively assessed. From the results, summarized in Figure 4-20, the flat-faced plug nozzle (Configuration 4) was selected.

(U) FIGURE 4-20 RAMJET NOZZLE TRADE

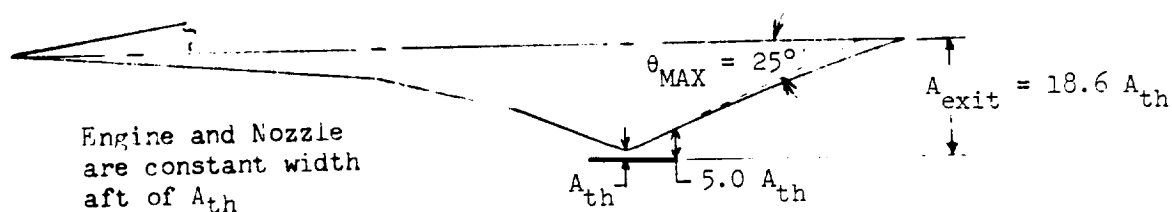
CONFIGURATION	EVALUATION CRITERIA			
	External Drag	Sidewall	Combustor Length Spraybar to Throat	Actuator Stroke
1 	Unaffected	Largest	Long	Shortest
2 	Bad when Throat Full Open	Moderate	Longest	Longest
3 	Unaffected	Large	Long	Intermediate
4 	Unaffected	Smallest	Smallest	Intermediate

*

*Selected Arrangement

(U) In the case of the scramjet and convertible scramjet engines, previous studies such as Reference (1) have indicated the desirability of completely integrating the propulsion system with the airframe. This includes the necessity of integrating the nozzle with the airframe lower surface. This study also showed the attractiveness of nozzle configurations that are essentially two-dimensional in shape. For the research vehicles, these integrated designs were used. The research vehicle nozzle design is summarized in Figure 4-21.

(U) FIGURE 4-21 SCRAMJET AND CSJ NOZZLE



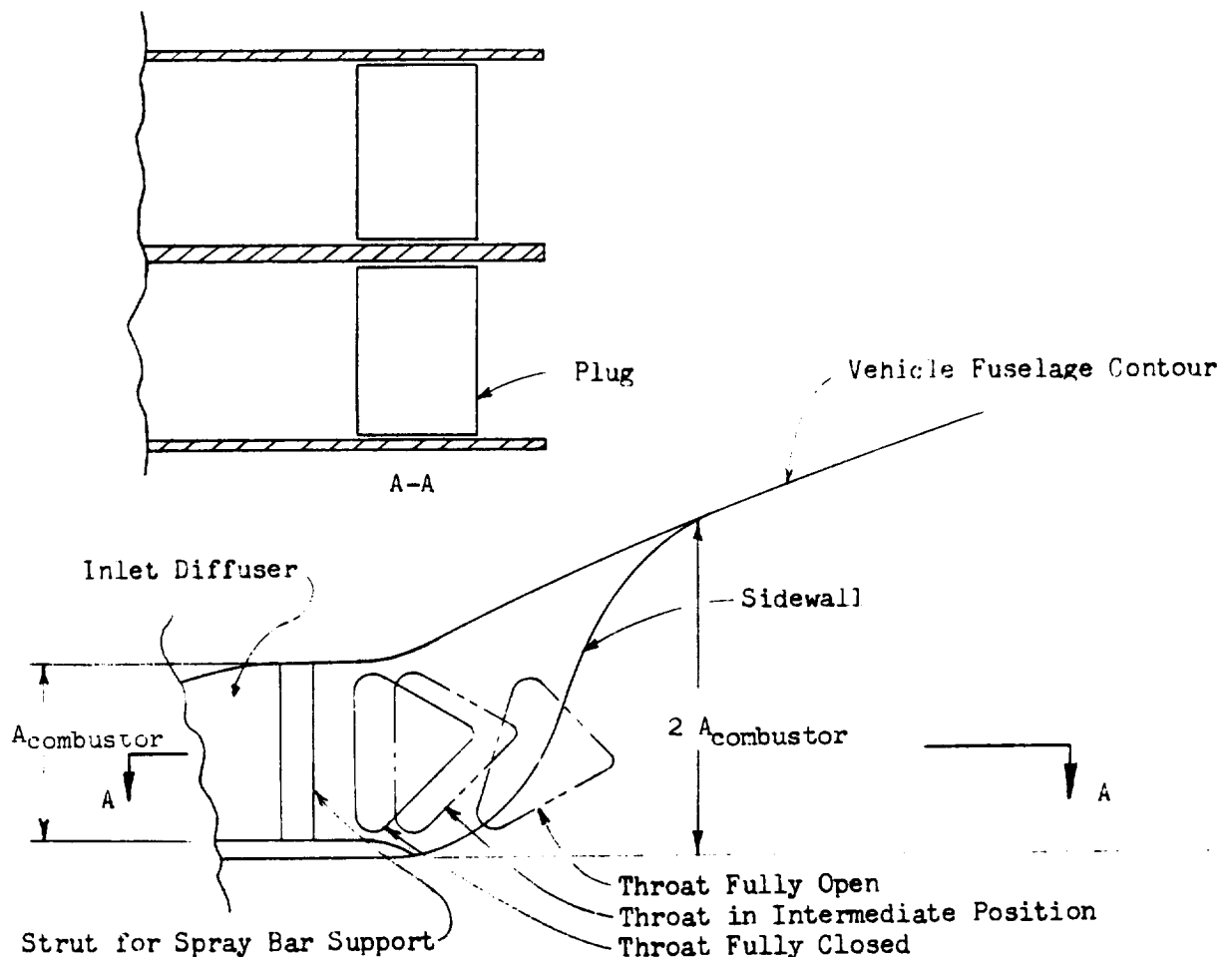
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(d) (U) Engine Selection - To choose the engine for the specified propulsion cycles of the seven baseline flight research vehicle concepts, both off-the-shelf and proposed (study) engines were evaluated. In the off-the-shelf category, only turbojets/turbofans and rockets were available. Proposed (study) engines were available for all propulsion cycles. The establishment of the selected engine configurations is discussed below.

(U) Turbine Engine - For the turbine engine applications, the F100 turbofan was selected. This choice resulted in a vehicle with less drag and more fuel volume than using the J58 engine, which has about the same thrust. Thrust of the GE4/J5P (SST engine) was too large for the research vehicle applications.

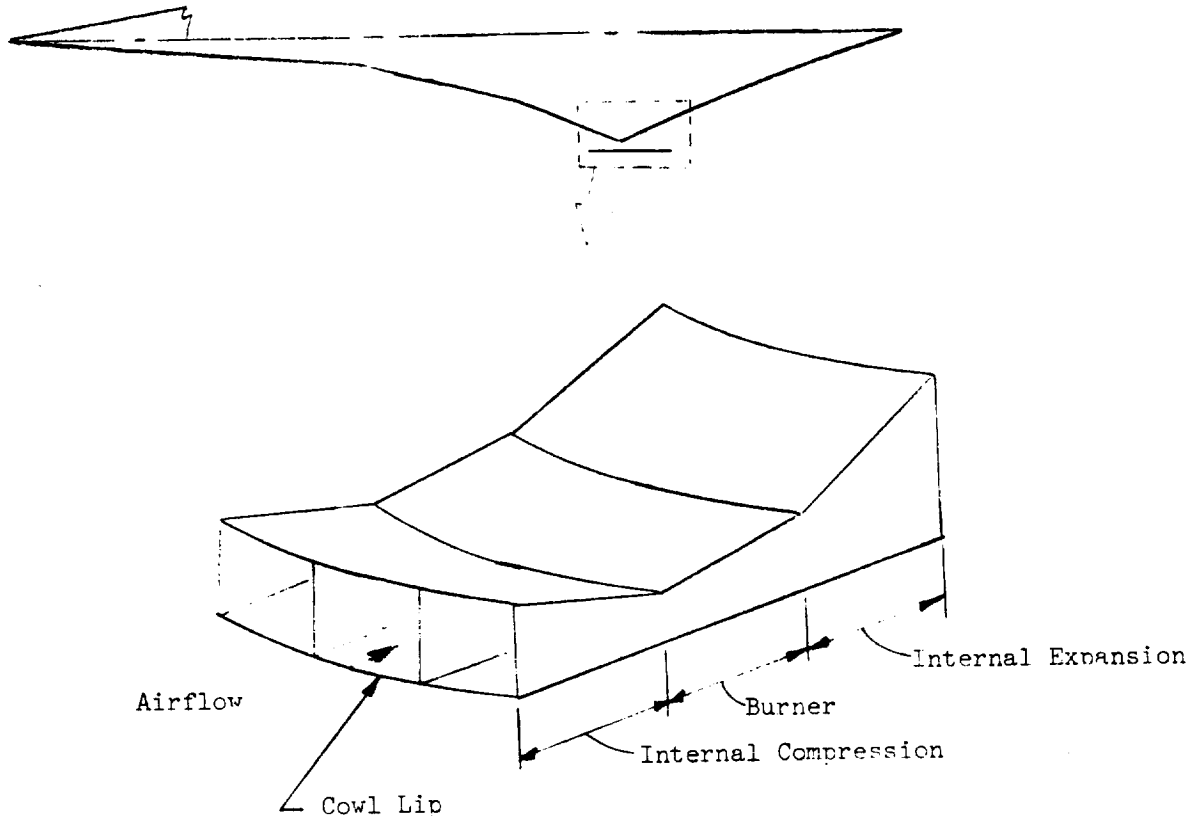
(U) Ramjet Engine - The Marquardt axisymmetric MA 145-XAB was the only ramjet engine available in the desired Mach number range. However in Phase I the integrations of this engine with the airframe produced vehicles with unattractive weight characteristics. Therefore, for Phase II, MCAIR configured a two-dimensional (2-D) ramjet engine, based on the MA 145-XAB concept. A sketch of this engine is shown in Figure 4-22.

(U) FIGURE 4-22 TWO-DIMENSIONAL RAMJET FEATURES



(U) Scramjet and Convertible Scramjet Engine - Propulsion system designs developed previously, Reference (1), were taken essentially unmodified and incorporated into the research vehicles. Figure 4-23 illustrates the research vehicle convertible scramjet engine.

(U) FIGURE 4-23 SCRAMJET AND CSJ ENGINES



(U) Rocket Engine - The high chamber pressure PWA LR-129 was the only available proposed advanced-technology rocket engine employing a conventional bell nozzle. Two off-the-shelf rocket engines were considered: The PWA RL10 (mod) and the Rocketdyne J2S. The tradeoff study to select among these three candidate engines is reported in Section 4.4.1. For the baseline vehicles, the major selection criterion was low OWE. The LR-129 engine was selected because it has a higher I_{sp} and thrust-to-weight ratio than the off-the-shelf engines. These characteristics are dominant when OWE is a major consideration. In applying this engine to the flight research vehicles, it was used in a rubberized fashion. That is, a thrust level was determined to match a desired value of takeoff thrust-to-vehicle-weight ratio. Then the associated engine length, diameter, and weight were determined from scaling data supplied by the engine company.

4.2.5 (U) STRUCTURES - Four basic variables that have a major impact on the structure and structural concepts were studied during Phase II. These are load factor, tankage, thermal protection system, and Mach number. During Phase II parametric trade-off studies were performed on each in an effort to select the best combination of variables. At the initiation of each tradeoff study, design options that satisfied the functional requirements, the design criteria, and the study ground rules were selected. Each design option was then investigated independently to establish the most attractive arrangement for the particular application. This was done to insure that each option was evaluated in its best configuration and that the selection was driven by performance and not by ground rule selection. Each design option that remained as a promising candidate was then incorporated into one or more research aircraft so that its impact on the total aircraft weight, size and cost could be evaluated.

(U) The overall vehicle comparisons were made using statistical weight estimating methods described in Section 4.2.6. These methods accurately reflect the changes that result from varying the design options. However, additional structural analysis was required for the integral/non-integral tank and the thermal protection system studies. These two tradeoff studies required preliminary analysis to be performed to assess the weight and volume variations associated with promising concepts since the estimation methods account only for the load carrying structure and not for extra heat protection or tankage material.

(U) The structural analysis and weight estimation were based on a "minimum risk" material and fabrication technology level. "Minimum risk" is defined as materials and fabrication methods that will be considered airworthy in 1975 with no more than normal development effort. Figure 4-24 presents a representative cross-section of the materials and their physical and mechanical properties that meet this requirement. Joining methods that are considered "minimum risk" include welding, brazing and limited diffusion bonding. The primary structure is considered as being mechanically joined.

(U) The non-metals that were included in the Phase II research aircraft were:

- o Windshield - 95% silica glass - flat panels, multi-glaze construction, 2000°F (1367°K) temperature capability.
- o Nose Cone - Zirconia ceramic modules with 4000°F (2475°K) temperature capability.
- o Insulation
 - Cryogenic
 - Polyvinylchloride (LO₂)
 - Polyurethane (LH₂)
 - High Temperature
 - Microquartz - 1800°F (1256°K) temperature limit
 - Flexible Min-K - 1800°F (1256°K) temperature limit
 - Dynaflex - 2700°F (1756°K) temperature limit
- o Vapor Barrier - Kapton-H

(U) FIGURE 4-24
PROPERTIES OF METALS AT ROOM TEMPERATURE

Alloy Designation	Condition (1)	Room Temperature Properties (Min. Guaranteed)								Temperature Limitation (2)			
		Ultimate Tensile Strength		Yield Tensile Strength		Elastic Modulus		Density		Primary Structure		Secondary Structure	
		KSI	N m ² x10 ⁶	KSI	N m ² x10 ⁶	10 ⁶ PSI	N m ² x10 ⁸	Lb/in. ³	Kg/m ³	°F	°K	°F	°K
2219-T81 (Aluminum)	STA	60	414	45	310	10.8	745	0.102	2820	400	477	500	533
2024-T62	STA	64	417	50	345	10.7	738	0.100	2768	325	436	375	464
7075-T76	STA	73	504	62	427	10.5	725	0.101	2795	250	394	275	408
Ti-6Al-6V-2Sn (Titanium)	STA	170	1171	160	1102	16.5	1138	0.164	4540	800	700	800	700
Ti-6Al-6V-2Sn	A	155	1069	145	1000	15.0	1034	0.164	4540	800	700	800	700
Ti-6Al-4V	STA	157	1082	143	986	16.4	1131	0.160	4425	800	700	900	755
Ti-6Al-4V	A	134	924	126	869	16.4	1131	0.160	4425	1000	810	1000	810
Ti-8Al-1Mo-1V	DA	133	917	121	835	18.0	1241	0.156	4320	1000	810	1100	866
Ti-5Al-2.5 Sn	A	120	827	113	780	15.5	1069	0.161	4460	900	755	1100	866
PH14-8Mo (Steel)	STA	240	1655	225	1550	28.0	1930	0.278	7700	1000	810	1100	866
A-286 (Iron)	A	140	965	95	655	29.1	2010	0.286	7910	1300	977	1300	977
L-605 (Cobalt)	A	130	896	55	379	32.6	2250	0.330	9130	1800	1256	2000	1367
Inconel 625 (Nickel)	A	120	827	60	414	29.8	2055	0.305	8440	1400	1032	1600	1144
Inconel 718	STA	180	1240	150	1034	29.0	2000	0.297	8210	1300	977	1800	1256
Rene 41	STA	170	1171	130	896	31.9	2200	0.298	8250	1550	1118	1800	1256
TD NiC	SR	110	758	75	517	21.9	1510	0.306	8460	2200	1480	2400	1590
Hastelloy X	A	100	689	45	310	28.6	1972	0.297	8210	2000	1367	2100	1421
B66 (Columbium)	A	90	621	70	483	14.6	1007	0.305	8440	2600	1700	2800	1810
FS-85	SR	75	517	60	414	20.0	1379	0.383	10060	2700	1756	2800	1810
Cb-752	A	75	517	60	414	17.0	1171	0.326	9020	2400	1590	2800	1810
T-222 (Tantalum)	A	110	758	105	725	29.0	2000	0.605	16740	3000	1920	3400	2140

(1) SR - Stress Relieved A - Annealed DA - Duplex Annealed STA - Solution Treated and Aged

(2) The selected maximum allowable service temperature limits are based primarily upon considerations of (a) competitive strength and/or stiffness at thermal exposures up to 100 hours and (b) adequate oxidation resistance either unprotected or with a suitable protective coating.

(U) The structural concept employed with the scramjet/convertible scramjet engines consists of a combination of primary structure, insulated radiation cooled external surfaces and regeneratively cooled internal panels. The primary load carrying structure which forms the shape of the engine is titanium structure maintained at 800°F (700°K). Protection from the external aerodynamic heating is provided by a thermal protection system (see Section 4.2.3 and 4.4.4). The internal wall of the engine forms the combustion chamber. This wall is made of TD-NiC, is regeneratively cooled thus acting as a heat exchanger, and in conjunction with insulation protects the primary structure from the combustion heating.

(U) Each of the thermal protection systems considered employs a radiation shingle for protection of the structure and the insulation. The results of the shingle optimization study (performed for an active system) are shown in Figure 4-25, as modified for the HYFAC Phase II parametric studies.

(U) FIGURE 4-25
SHINGLE DESIGN FEATURES

MATERIAL	TITANIUM	RENE' 41	TD NIC	COLUMBIUM	TANTALUM
Density (lb/in ³) (g/cm ³)	.160 4.5	.298 8.38	.306 8.6	.326 9.16	.606 17.0
Temperature (°F) Limit (°K)	1100 867	1550 1115	2200 1480	2800 1810	3400 2140
Panel Capability ⁽¹⁾ (psi) @ Elevated Temp. (N/cm ²)	16.5 11.3	30.3 20.8	4.5 3.1	5.1 3.5	8.6 5.9
Material Gage-Face (in) (cm)	.007 .0178	.007 .0178	.007 .0178	.008 .0203	.012 .0305
Construction Method ⁽²⁾	Diffusion Bond	Braze	Braze	Braze	Braze
Weight (lb/ft ²) (kg/m ²)	.68 3.32	1.41 6.90	1.45 7.07	1.57 7.67	4.52 22.05
Expansion @ Temp. (in) 24 Inch Length (cm)	.15 .381	.32 .813	.50 1.27	.30 .762	.42 1.07

(1) Based on post supported, 24" x 24" x 1/2" (61 cm x 61 cm x 1.27 cm)

(2) Honeycomb Core

4.2.6 (U) WEIGHTS - The basic weight estimation methodology used in Phase II is unchanged from that presented in Volume II. However, several of the Phase II parametric studies (Figure 4-2) required weight trend data whose source and methodology is discussed below.

(U) Rocket Motor Weight Trends - The weight of a wet, installed motor, including the necessary wire harness and instrumentation, thrust vector control system (if required), trapped propellants, spin-up system, and pressurant, varies as illustrated in Figure 4-26. Data is presented for:

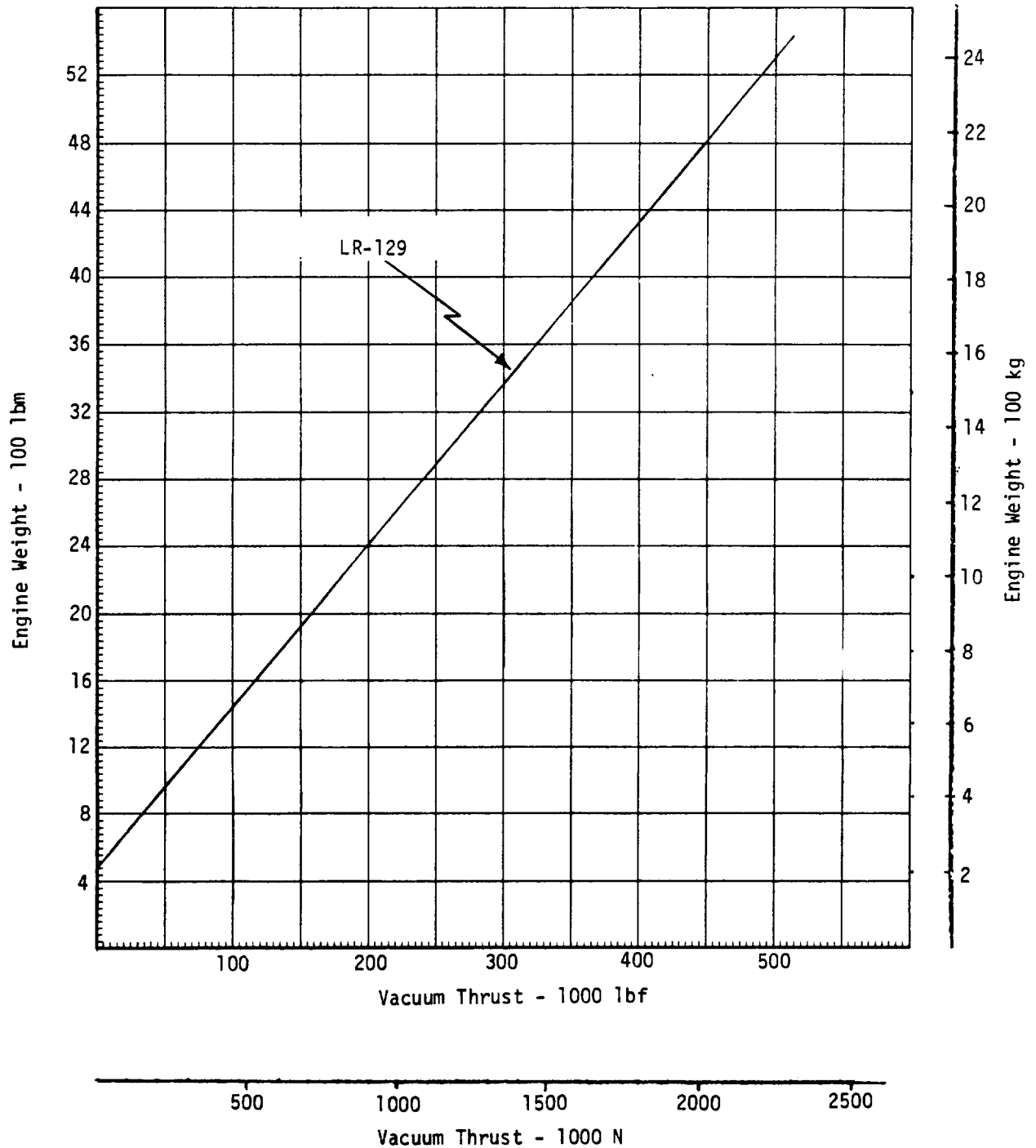
- o LR-129 HPC LO₂/LH₂ Translating Nozzle (Reference 11)

(U) Ramjet Weight Trends - The conventional circular-shaped ramjet was replaced by the two-dimensional (rectangular cross-section) ramjet concept to provide better airframe/inlet propulsion system integration. Weight trends were developed using a circular ramjet as a base. The base engine used was the Marquardt 145 circular ramjet. This engine was scaled as desired using the Marquardt scaling curves for capture area and minimum duct pressure (Reference 12). However, since the MCAIR ramjet is rectangular, a weight-strength correction factor was applied. This derives a rectangular pressure vessel from a circular vessel of the same cross-sectional area. Figure 4-27 presents the MCAIR two-dimensional ramjet weight as a function of capture area.

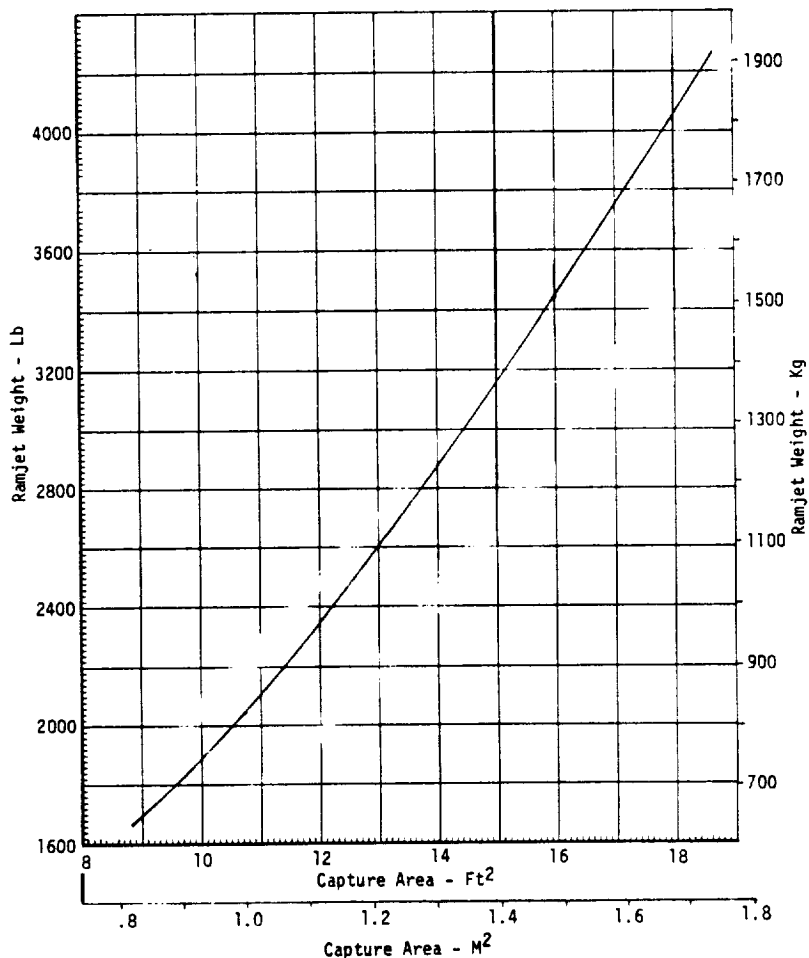
(U) Scramjet - Convertible Scramjet Weight Trends - Structural analyses have been performed on the scramjet-convertible scramjet engines. The results of these analyses have been modified and adapted to the Phase II research aircraft. Figure 4-28 shows the results of the preliminary structural analysis of the engine, which includes the weight of the structure and thermal protection system. Pressures in the inlet and nozzle sections are much less than those in the combustor. Therefore, the structural weight was computed separately for each section. For convenience, the weight is presented as a function of the combustor section pressure. The scramjet engines are designed for 100 psi (68.9 N/cm²) while the convertible scramjets are designed for 150 psi (103.2 N/cm²) reference Figure 4-3. The weight is also presented as a function of module frontal area, since the engine size is dependent upon the particular configuration being studied. The weight of the modules required for adequate thrust can be assembled by summing the individual values.

(U) The engine weight includes the structure, regenerative cooling system, fuel injectors, and ramp mechanism, with the back-up and mounting structure being carried in the main body weight. Trapped fluids were accounted for separately in the useful load section. Engine weights are estimated on the basis of weight trends and methodology described in Volume II, Part 2.

(C) FIGURE 4-26
ENGINE WEIGHT vs VACUUM THRUST

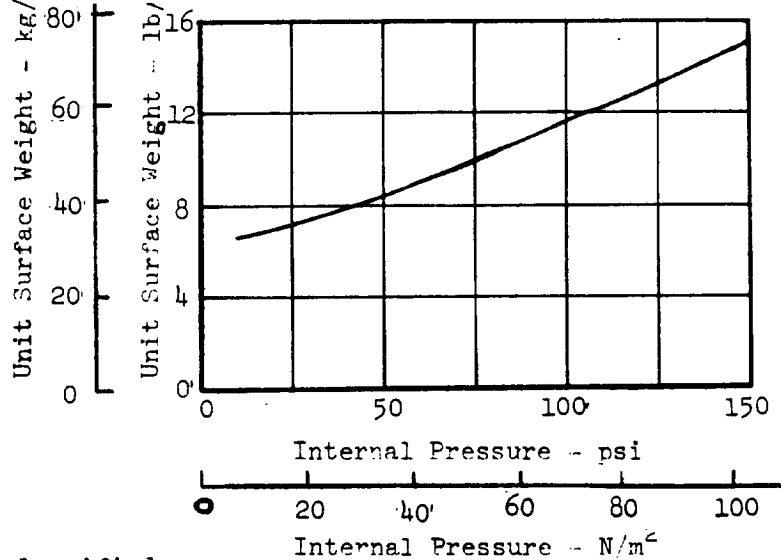


(U) FIGURE 4-27
2 DIMENSIONAL RAMJET WEIGHT SCALING



(U) FIGURE 4-28

SCRAMJET-CONVERTIBLE SCRAMJET WEIGHT SENSITIVITY

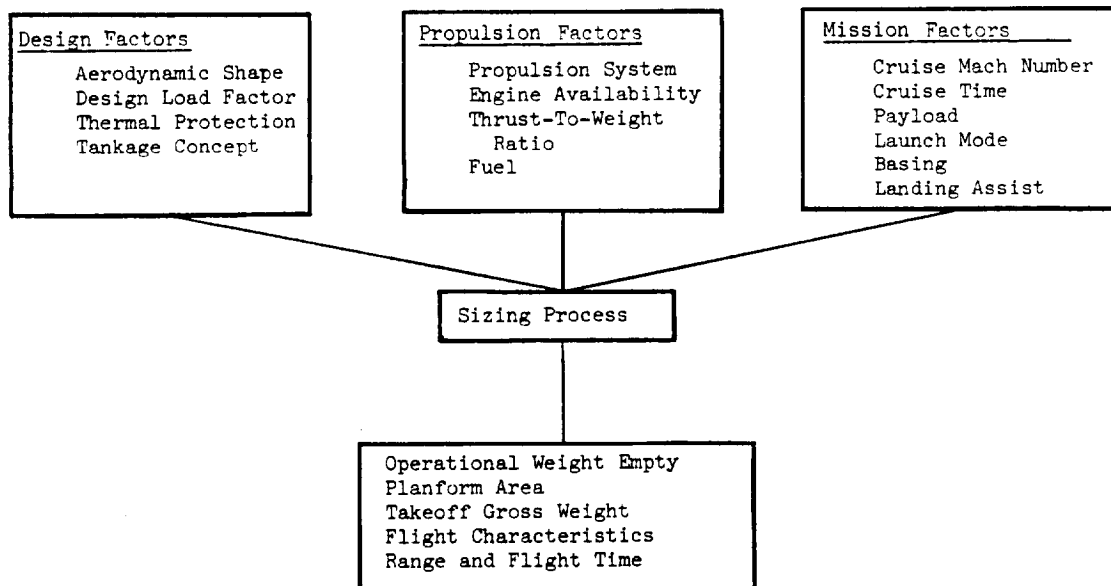


4.2.7 (U) PERFORMANCE METHODOLOGY - The methods and approaches used in analyzing the performance capability of the flight research vehicles are presented in this section. Each of the noted performance subjects are discussed in the following sections.

- o Airbreather Accelerator Methodology
- o Rocket Accelerator Methodology
- o Nominal Glide Path
- o Landing Characteristics
- o Landing Assist
- o Single Base Operation

(U) The general vehicle sizing technique employed for Phase II is the same as that developed for Phase I studies wherein a balance is obtained between the fuel available in a given vehicle and the fuel required to perform the mission. A schematic of the input requirements for the sizing process and the output are shown in Figure 4-29.

(U) FIGURE 4-29 POINT DESIGN SIZING PROCESS

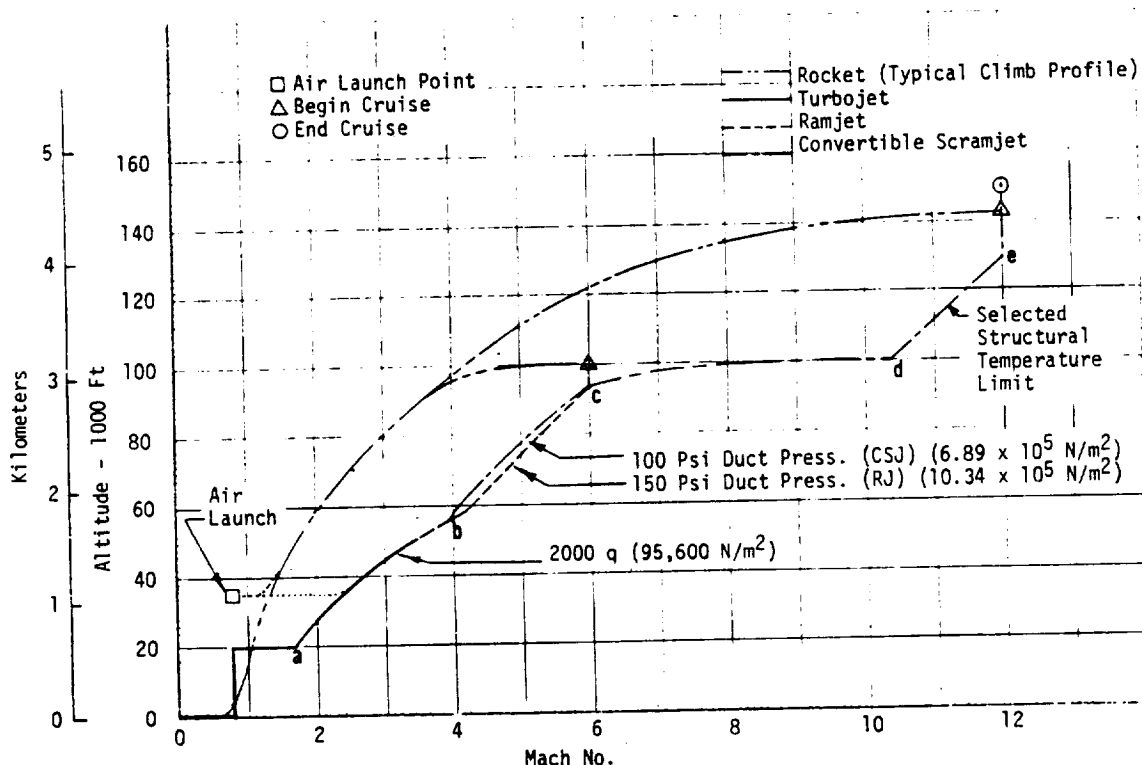


(U) The mission consists of an acceleration-climb, cruise, and an unpowered glide descent and landing. The climb phase of flight is performed differently for airbreathing and rocket accelerator engines. Figure 4-30 defines the climb profiles. The rocket accelerators employ a climb path for minimum fuel usage. For the airbreathing accelerator vehicles, various structural limitations defined the method of climb. These constraints are defined on the profile:

- (a) to (b) - dynamic pressure boundary, 2000 lbs/ft² (95,600 N/m²)
- (b) to (c) - engine duct pressure boundary, 100 psi (6.89×10^5 N/m²) for CSJ and 150 psi (10.3×10^5 N/m²) for RJ
- (c) to (d) - combustion mode of operation switched from subsonic to supersonic and a constant altitude acceleration is employed.
- (d) to (e) - selected structural temperature limit.

(U) For mission planning purposes, a nominal glide path was defined. The glide maneuver utilizes a speed brake system for glide path control (Section 4.2.7.3). Flight research vehicle landing studies were performed assuming acceptable vehicle handling qualities and a landing assist technique was defined. Basing concept studies were conducted considering, as an alternate, single base operation.

(U) FIGURE 4-30 PHASE II CLIMB PROFILES
Baseline Systems (Nominal)



4.2.7.1 (U) Airbreather Accelerator Methodology - HYFAC vehicles with airbreathing accelerator engines were sized by matching fuel available on a given size vehicle with fuel required to perform the mission. The climb and the cruise portions of the mission were considered separately in calculating the fuel required.

(U) The climb fuel was calculated using the KC6G Advanced Design Mission Program. Ground rules for using the program for HYFAC are that Isp is reduced by 5%, turbojet and ramjet thrust are assumed to be independent of angle of attack, and scramjet thrust is input as a function of angle of attack. Briefly, this program uses a closed form iterative procedure to calculate time and fuel required to accelerate and climb. With a known gross weight at the beginning of climb an estimate was made of the fuel required to traverse a segment of the climb path. Based on initial and final weights, the fuel required was calculated and the iteration (step size, $\Delta M = 0.5$) continued until the estimated and the required fuel converge. This process was repeated along the entire climb path.

(U) The cruise phase of flight is at $(L/D)_{\max}$ equilibrium altitude. For this phase of flight, the following ground rules were defined for determining cruise fuel requirements:

$$\text{At } (L/D)_{\max} \quad C_{D_0} = L' C_L^2$$

$$\text{therefore} \quad C_L = \sqrt{\frac{C_{D_0}}{L'}}$$

(U) For vehicles with scramjets, the propulsive lift increment was added to the aerodynamic lift. The equation defining cruise fuel requirements is then:

$$W_{\text{FUEL}} = \frac{D}{I_{\text{sp}}} (t)$$

The specific impulse, Isp, of the airbreathing engines is arbitrarily reduced by 5% to account for differences between estimated engine performance and flight test operation performance. Cruise fuel is then:

$$W_{\text{FUEL}} = \frac{2 C_{D_0} q S p}{.95 (I_{\text{sp}})} (t)$$

$$\text{where } q = \frac{W \left[1 - \left(\frac{v^2}{\sqrt{(R_e + h)g}} \right)^2 \right]}{(C_{L\text{AERO}} + C_{L\text{PROP}}) S p}$$

$C_{L\text{AERO}}$ = aerodynamic lift coefficient

$C_{L\text{PROP}}$ = propulsion system lift coefficient

Cruise altitude versus weight with required fuel was compared to an altitude versus weight curve during climb for a given Mach number. The intersection of these two curves was used as the initial cruise condition.

(U) Fuel volume available versus planform area and empty weight were obtained from the configuration design curves. For the all hydrogen fueled vehicles, the takeoff gross weight was readily obtained by plotting fuel available and fuel required versus empty weight; the intersection determined the size of the design.

(U) For dual fueled (JP-hydrogen) vehicles, an iterative procedure must be used since fuel weight is not simply a function of volume. A takeoff gross weight was assumed and the JP required to accelerate to Mach 3, the shutdown point of the turbojets, was calculated using the KC6G Program. The volume of the JP used was subtracted, from the total volume, leaving the remaining volume available to carry hydrogen. This resulted in a new takeoff gross weight. The process was repeated until the weights converged. This takeoff gross weight was used to calculate mission fuel required and size the aircraft as described above.

4.2.7.2 (U) Rocket Accelerator Methodology - HYFAC vehicles with rocket accelerators were sized to provide a balance between fuel available and fuel required to perform the design mission. The fuel required to perform the mission was determined by using a closed form iterative solution. The solution was found to be adequate for sizing rocket accelerator vehicles when compared to point mass trajectory results, and enabled a large number of vehicles to be evaluated.

(U) The closed form technique considered fuel requirements for a climb and a constant Mach number cruise phase of flight. The cruise requirements were determined initially from the Breguet Range Equation solving for the cruise weight fraction (W_c/OWE),

$$\ln \frac{W_c}{OWE} = \frac{\Delta t_{TEST}}{(L/D)_{max}} \left\{ \frac{1}{1 - \left[\frac{V}{\sqrt{(Re + h)g}} \right]^2} \right\} I_{Sp} (.99)$$

The specific impulse is arbitrarily reduced 1% for rocket engines, as indicated, to account for differences between estimated and operational engine performance.

(U) The acceleration-climb fuel requirements were determined from the classical rocket equation solving for the climb weight fraction ($\frac{TOGW}{W_c}$),

$$\ln \frac{TOGW}{W_c} = \frac{\Delta V_{ideal}}{g I_{sp}} (.99)$$

where: TOGW = take-off gross weight (lb)

$$\Delta V_{ideal} = \Delta V_{actual} + \Delta V_D + \Delta V_G + \Delta V_M + \Delta V_P$$

and $\Delta V_D = \text{aerodynamic drag velocity loss} = \int_{t_1}^{t_2} \frac{D}{m} dt$

$\Delta V_G = \text{gravity velocity loss} = \int_{t_1}^{t_2} g \sin \gamma dt$

$\Delta V_M = \text{maneuver velocity loss} = \int_{t_1}^{t_2} \frac{(T_{VAC} - P A_e)}{m} (1 - \cos \alpha) dt$

$\Delta V_P = \text{engine exit nozzle atmosphere back pressure loss} = \int_{t_1}^{t_2} \frac{P A_e}{m} dt$

The ΔV losses during the climb phase were evaluated using correlations derived from approximately 50 calculated acceleration-climb profiles for minimum fuel. The velocity loss correlations used in the performance calculations are shown in Figures 4-31 through 4-33 for both air and horizontal ground launch. These curves are the same as those used in Phase I studies with the exception of the air launch drag losses which were refaired at T/W ratios less than 1.5 when additional data for the correlation became available.

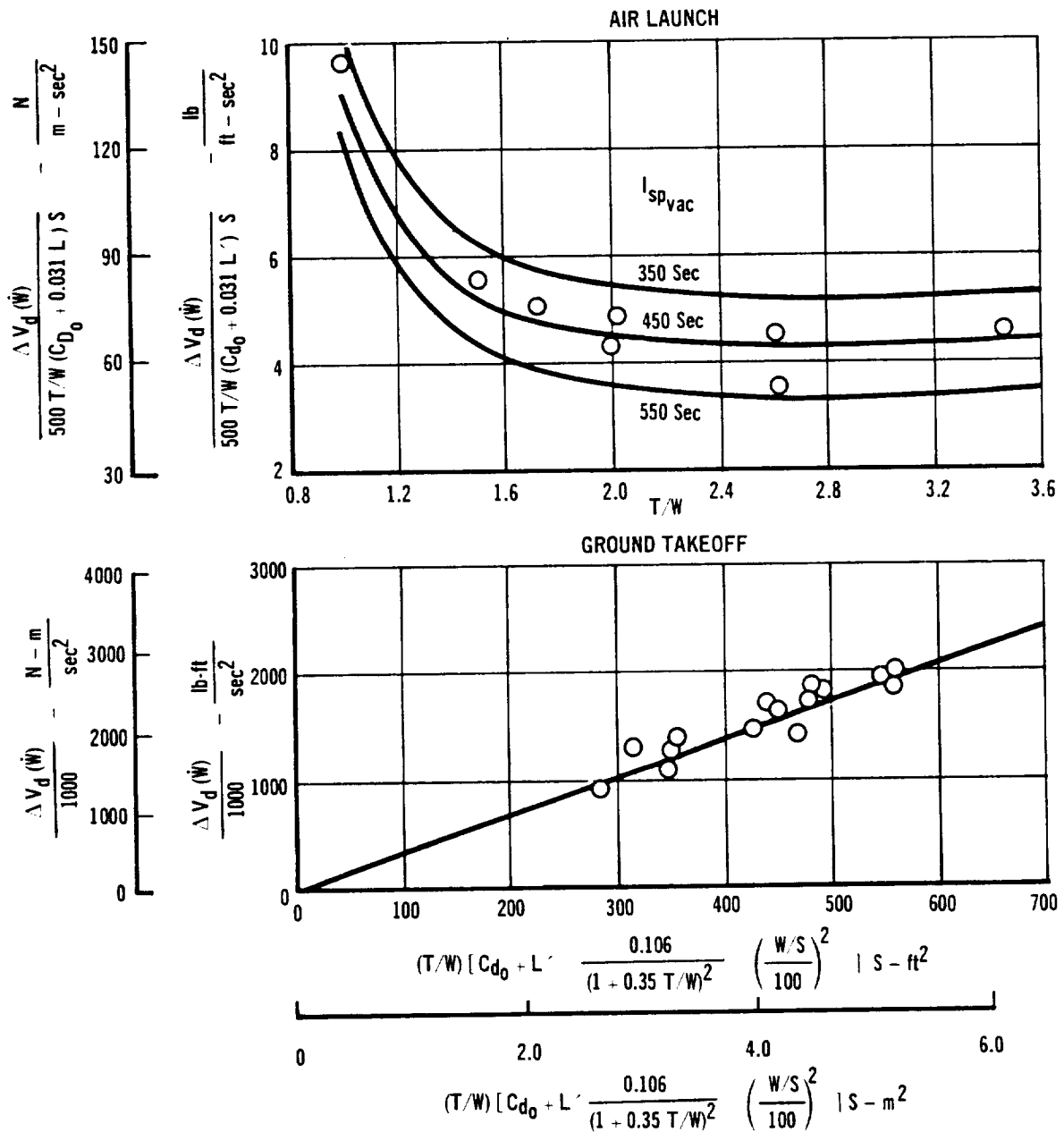
(U) The fuel required curve and the size of the vehicle were determined as follows:

- (a) Assume weight empty (OWE)
- (b) Solve for cruise weight (W_C) using Breguet Range Equation iterating until the assumed cruise weight (used to obtain cruise altitude and test velocity) agrees with the resultant cruise weight.
- (c) Solve for take-off gross weight using the classical rocket equation iterating until the assumed TOGW (used to determine velocity losses) agrees with the resultant TOGW.
- (d) Repeat (a) through (c) to obtain another point. The resultant two points define the relationship of the fuel required with empty weight. A linear relationship results from the assumption of a constant value of Isp.
- (e) The sized vehicle is at the intersection of the fuel required curve with the fuel available curve (obtained from design data).

4.2.7.3 (U) Nominal Glide Path - It is necessary to be able to control the vehicle flight path during the power-off glide following cruise in order to provide the pilot with flexibility in the selection of a landing site. Range studies have been conducted assuming the use of partially deflected speed brakes during the glide phase thereby providing a "nominal" glide range that is the mean value of the maximum and minimum glide ranges.

(U) FIGURE 4-31 DRAG LOSSES
Rocket Boost

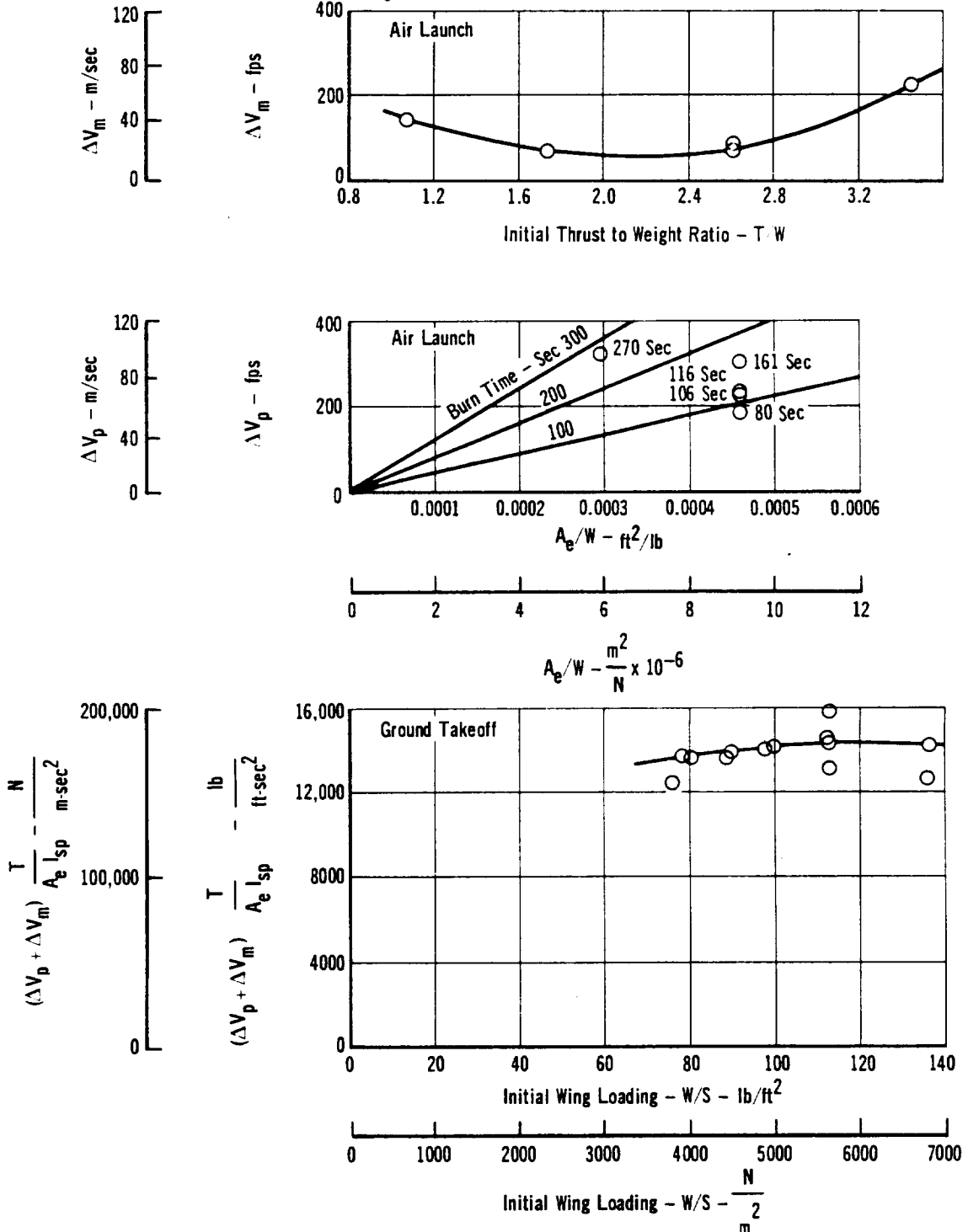
ΔV_d	Drag Loss - fps (m/sec)	C_{d0}	Zero Lift Drag Coefficient
T	Vacuum Thrust - lb (N)		at Mach = 1.2
W	Initial Weight - lb (N)	L'	Induced Drag Factor
$I_{sp_{vac}}$	Vacuum Specific Impulse - sec		$\Delta C_d / \Delta C_L^2$ at Mach = 1.2
$\dot{w}$	Propellant Flow - lb/sec ($\frac{N}{sec}$)	S	Reference Area - ft ² (m ²)



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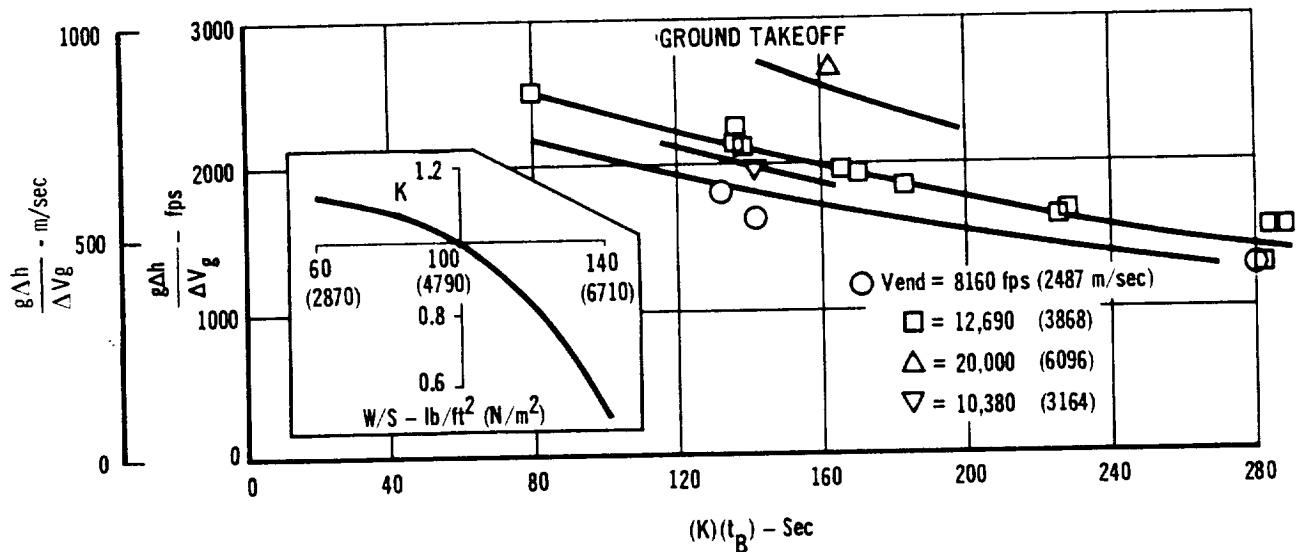
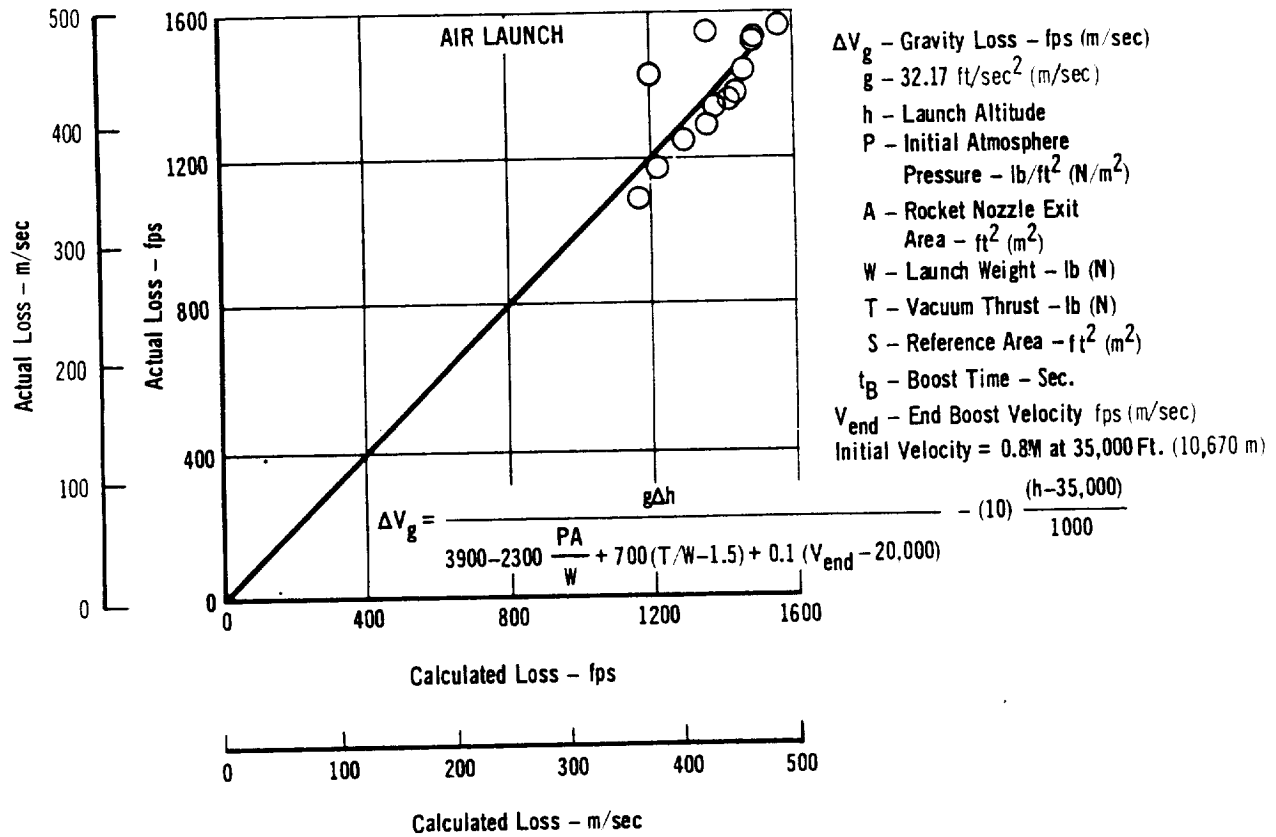
(U) FIGURE 4-32
PRESSURE AND MANEUVERING LOSSES
Rocket Boost

ΔV_p - Pressure Losses - fps (m/sec)
 ΔV_m - Maneuver Losses - fps (m/sec)
 A_e - Rocket Nozzle Exit Area - ft² (m²)



MCDONNELL AIRCRAFT

(U) FIGURE 4-33
GRAVITY ROCKET BOOST
Rocket Boost



(U) To size speed brakes, a survey of fuselage mounted speed brake data was conducted and the results are shown in Figure 4-34. The fairing of data is arbitrary. The asymptotic value indicated in Figure 4-34 results from modified Newtonian theory with a correction factor (K) from Hoerner (Reference 5) to account for the fullness of the pressure distribution over the face of the speed brake, as a function of deflection. Speed brakes were sized using the data fairing of Figure 4-34 together with the ground rule that they be effective enough to provide a minimum $\Delta(L/D)$ of at least -1.0 throughout the vehicle speed range. Typical results, obtained for an all-body configuration (-250), are as follows:

$$\text{Area Ratio, } S_B/S_p = .0206$$

$$\text{Max Deflection, } \delta_B = 50^\circ$$

where: S_B = speed brake flat plate area

δ_B = speed brake deflection angle, degrees

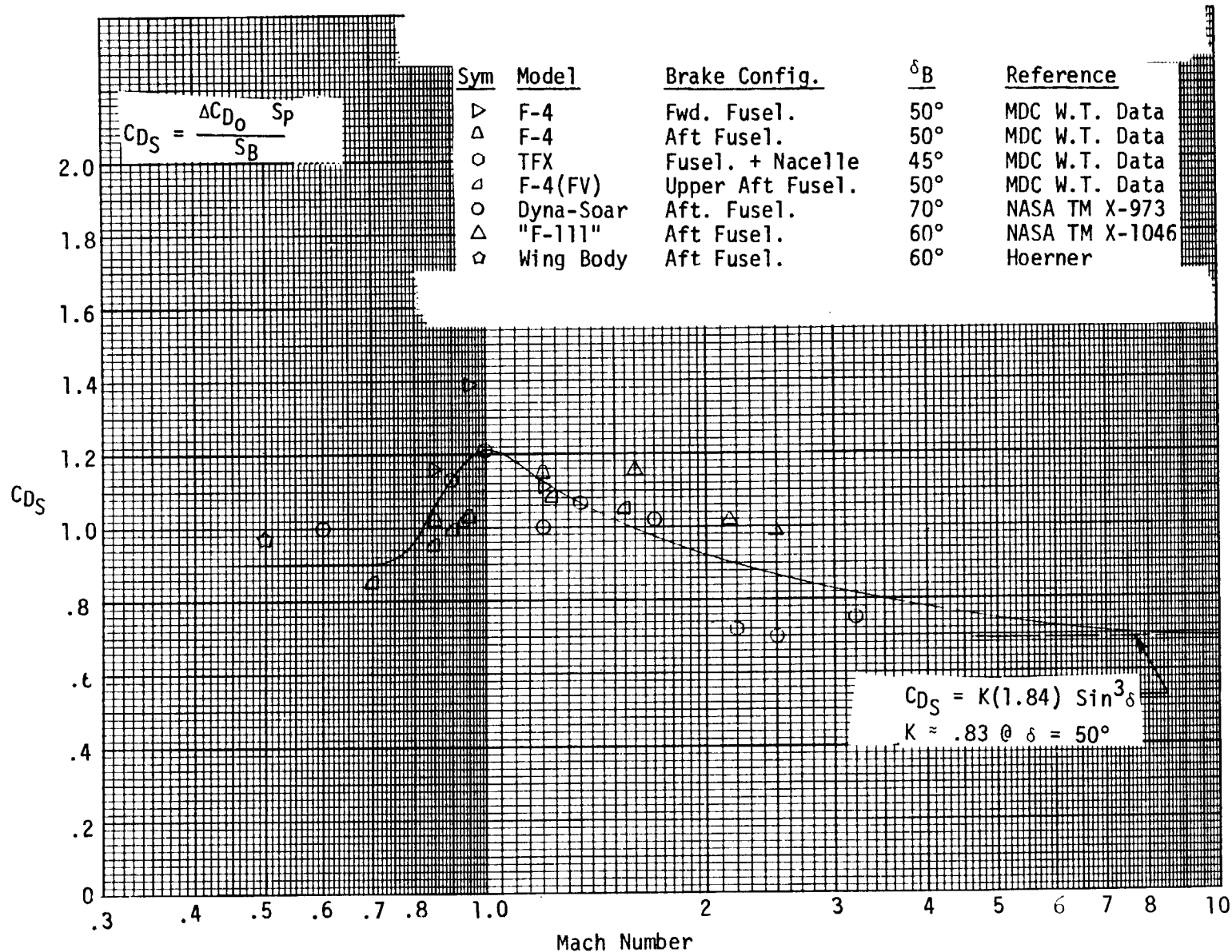
S_p = vehicle planform area

The maximum glide range used for all HYFAC test vehicles assumes flight at aerodynamic $(L/D)_{\max}$. Minimum glide range is obtained with full speed brake deflection and angle of attack modulation. An angle of attack of 3 degrees was found to provide a minimum range without exceeding a dynamic pressure limitation of 2000 lb/ft² (95,600 N/m²).

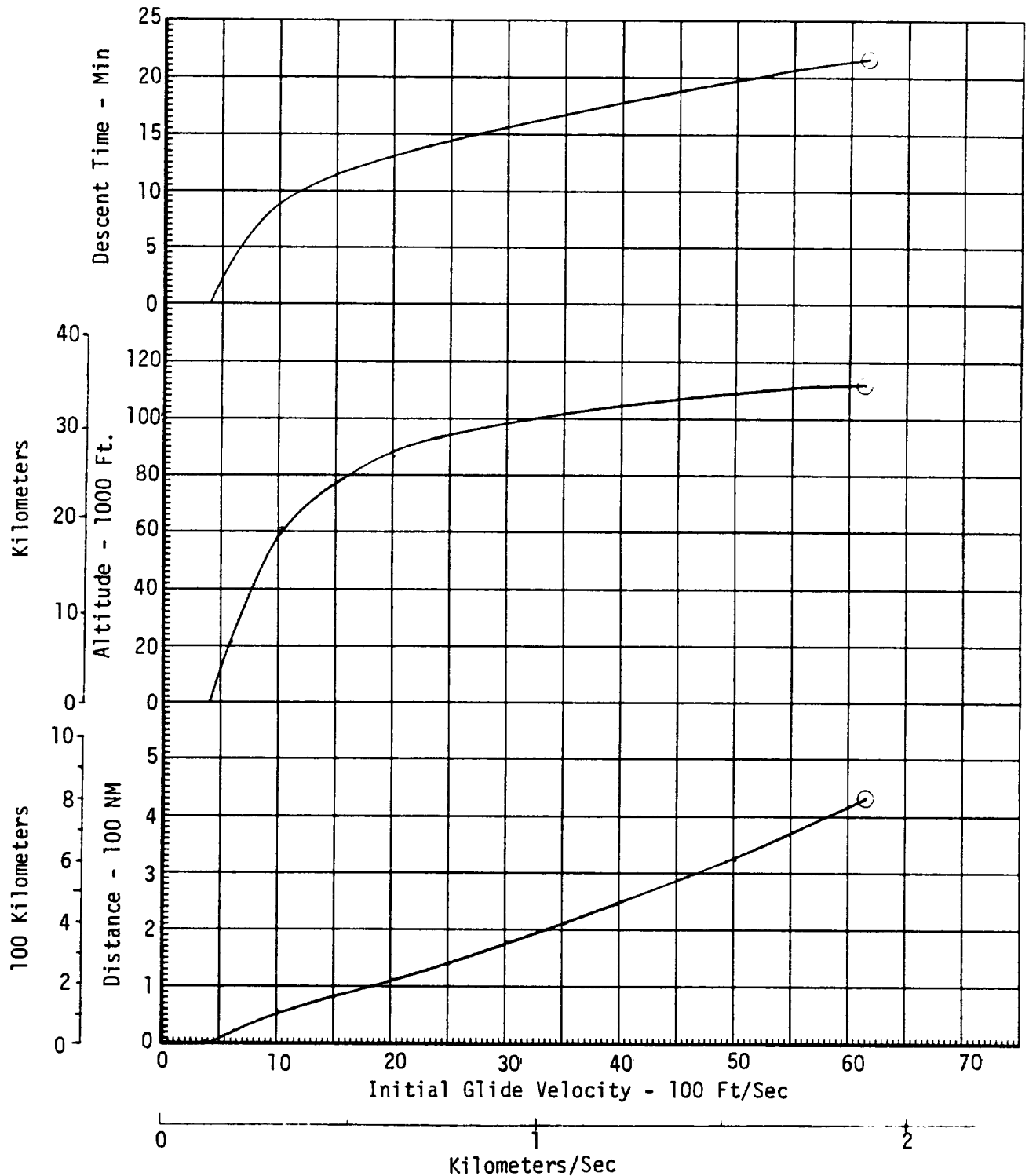
(U) Representative glide parameters for the baseline HYFAC test vehicles are presented in Figures 4-35 through 4-37 where maximum glide distance, altitude and time are shown as a function of velocity at end of cruise. These trajectory parameters were obtained using the MDC three degrees of freedom point mass trajectory program, (deck KSDF). The nominal glide range is approximately 65% of the maximum glide range in all cases investigated.

4.2.7.4 (U) Landing Characteristics - Approach and landing characteristics for a typical all-body configuration (B233A) are shown in Figure 4-38. These characteristics are based on a power-off high energy approach technique, wherein the initial approach speed is sufficiently in excess of the touchdown velocity to permit a flare to a shallow final glide path and while decelerating provide the pilot with enough time-on-final to make corrective alignment adjustments, extend the landing gear, and negotiate a smooth touchdown. The chief parameters influencing these characteristics are $(L/D)_{\max}$, CL_{TD} , CL_{OPT} , CL_α , and W/S . The boundaries of maximum and minimum values and the preferred conditions shown in Figure 4-38 are based on the results of a McDonnell pilot-in-loop landing simulation study, Reference (13). The preferred conditions denote those conditions for which the best pilot ratings were obtained. The maximum and minimum boundaries correspond to a Cooper Rating of 3.5, or the pilot rating separating Satisfactory and Unsatisfactory conditions on the Cooper Scale.

(U) FIGURE 4-34
SPEED BRAKE DRAG
Fuselage Mounted Brakes

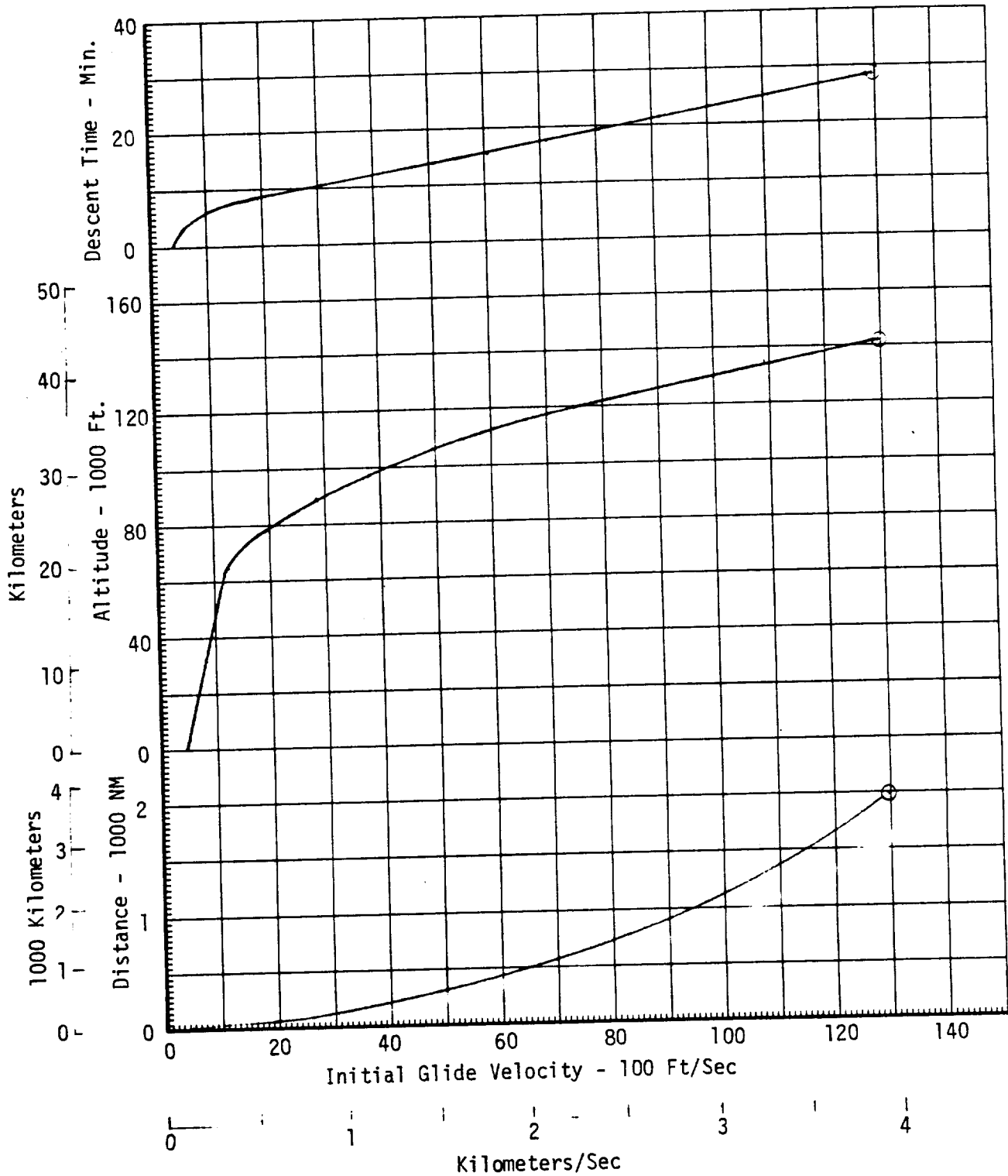


(U) FIGURE 4-35
(L/D)_{MAX} EQUILIBRIUM GLIDE DESCENT
TJ/RJ Wing Body Configuration (B212A)



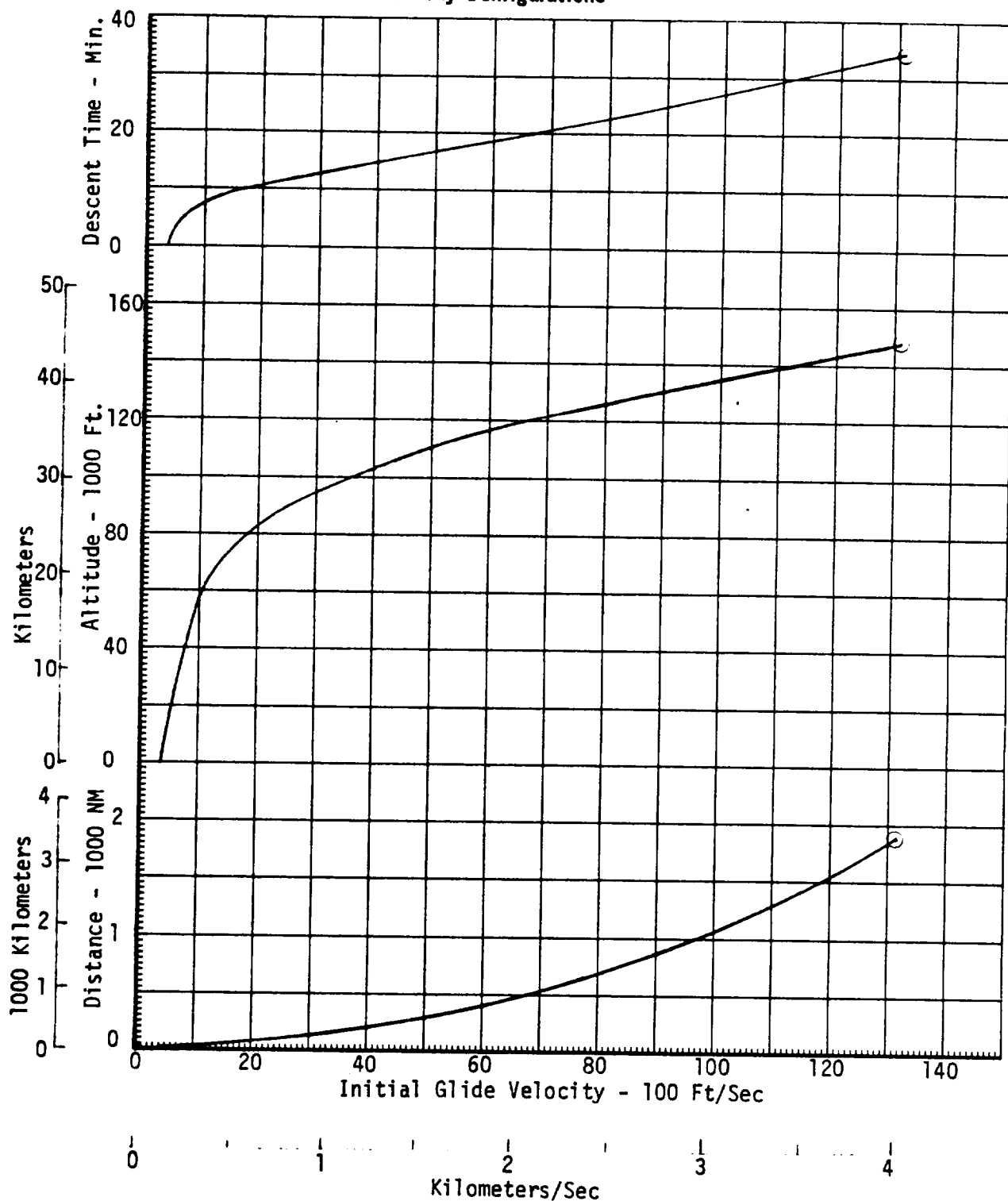
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(U) FIGURE 4-36
(L/D)_{MAX} EQUILIBRIUM GLIDE DESCENT
TJ/CSJ Blended Wing Body Configuration (B257F)

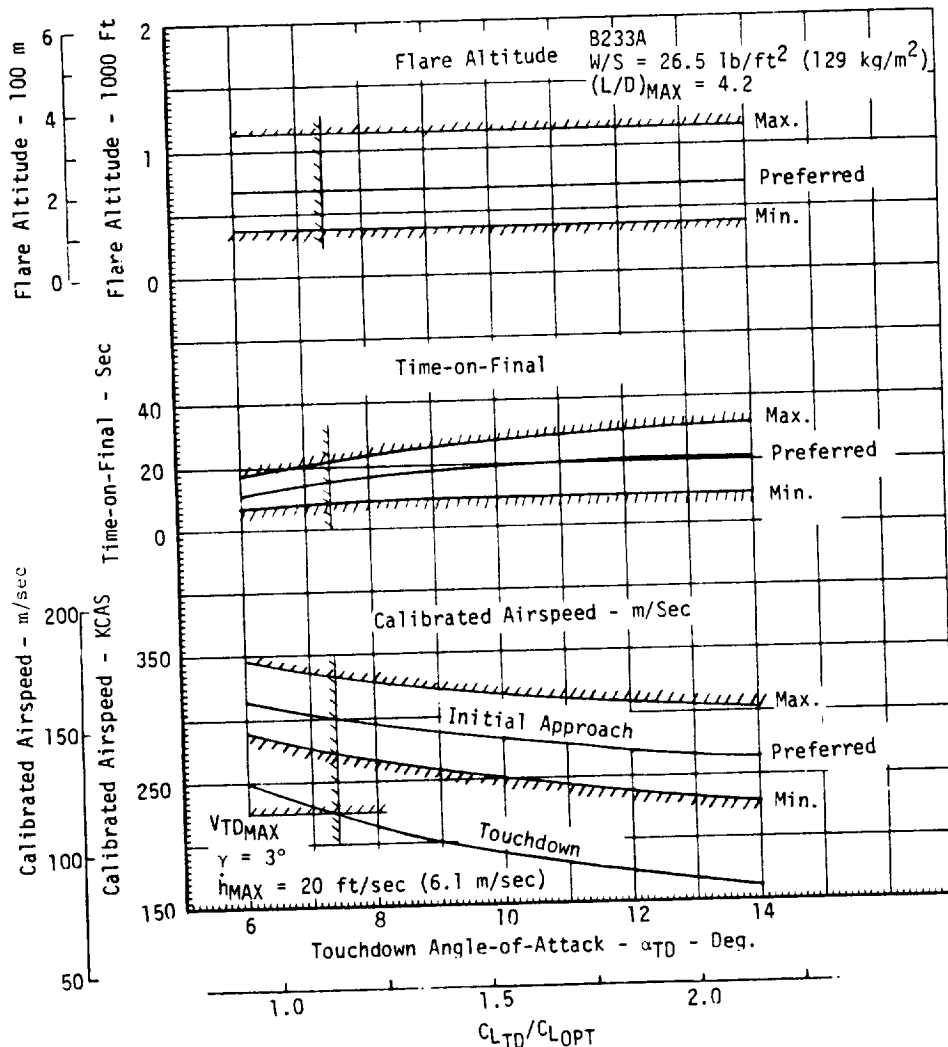


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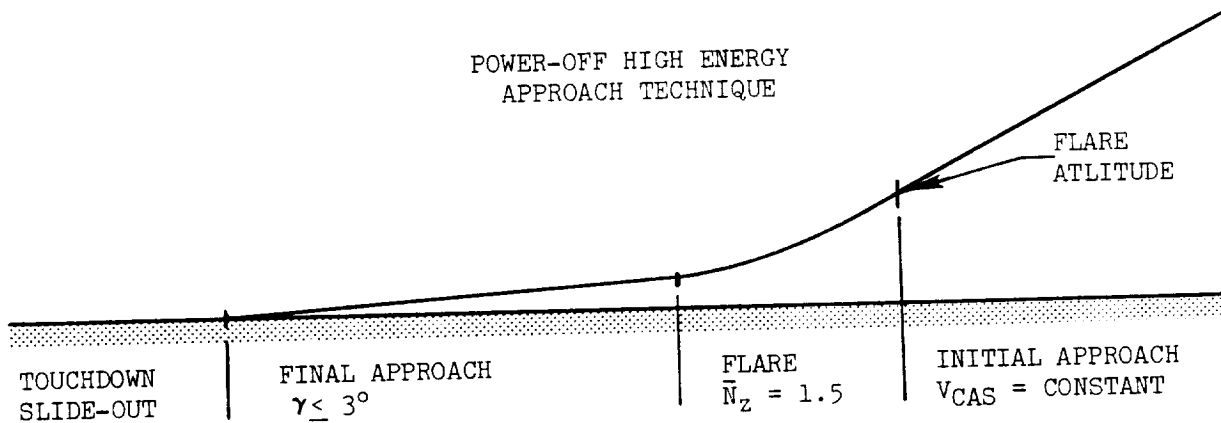
(U) FIGURE 4-37
(L/D)_{MAX} EQUILIBRIUM GLIDE DESCENT
Rocket All Body Configurations



(U) FIGURE 4-38
APPROACH AND LANDING CHARACTERISTICS
Power-Off



POWER-OFF HIGH ENERGY
APPROACH TECHNIQUE



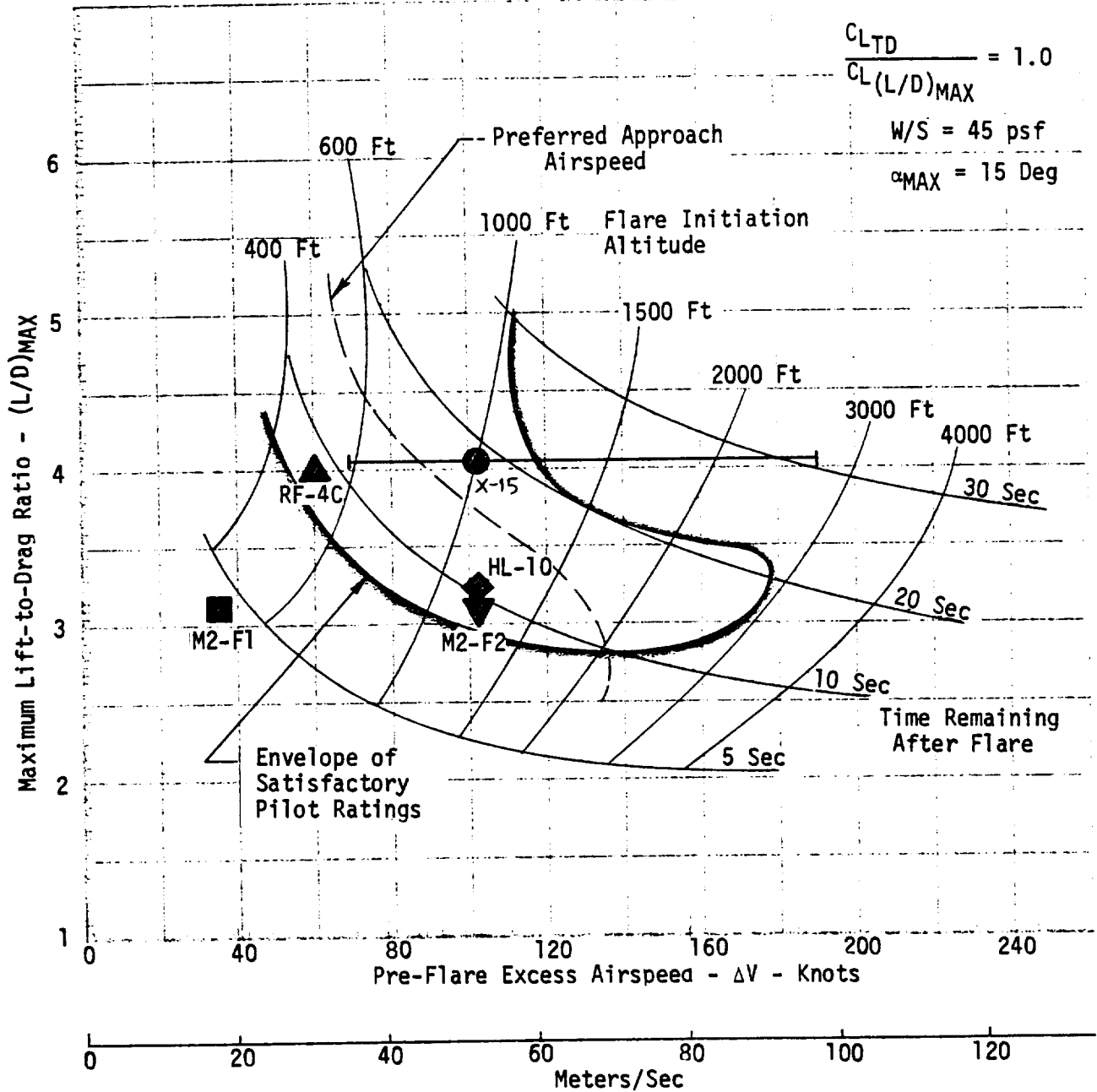
(U) A comparison of actual flight experience with the envelope of Satisfactory Pilot Ratings is presented in Figure 4-39. The X-15 was the first research vehicle to employ a power-off high energy approach as a part of its normal flight operation. In the course of this program, the X-15 investigated a wide range of approach conditions as illustrated in Figure 4-39. The M2-F1 vehicle was considered to exhibit unsatisfactory landing characteristics because of the lack of time-on-final, which agrees well with the simulation study results. The RF-4C data point represents landings performed by one of the evaluating McDonnell test pilots at Edwards Air Force Base to familiarize himself with the high energy approach technique prior to the simulator study.

(U) If the landing gear of the all-body configuration employed in Figure 4-38 is designed for a touchdown sink rate of 20 ft/sec (6.1 m/sec), then touchdown velocities must be limited to about 225 knots (417 kilometers/hr) assuming a -3 degree flight path angle at touchdown. If it is further assumed that the angle of attack at touchdown should not exceed 15 degrees because of pilot over-the-nose vision, stability and control considerations, etc., the minimum touchdown velocity should be about 160 knots (297 kilometers/hr). This provides a useable range of touchdown speeds of approximately 65 knots (120.5 kilometers/hr). Nominal touchdown conditions could be assumed to be a velocity of 190-195 knots (352-362 kilometers/hr) and an angle of attack of about 10° requiring a C_{LTD} approximately 50% greater than C_{LOPT} . To provide these conditions at touchdown, the preferred approach would be at an initial calibrated airspeed of about 285 knots (528 kilometers/hr) followed by a flare of about 1.5 g initiated at an altitude of 700 feet (214 m) and a final deceleration time of about 20 seconds.

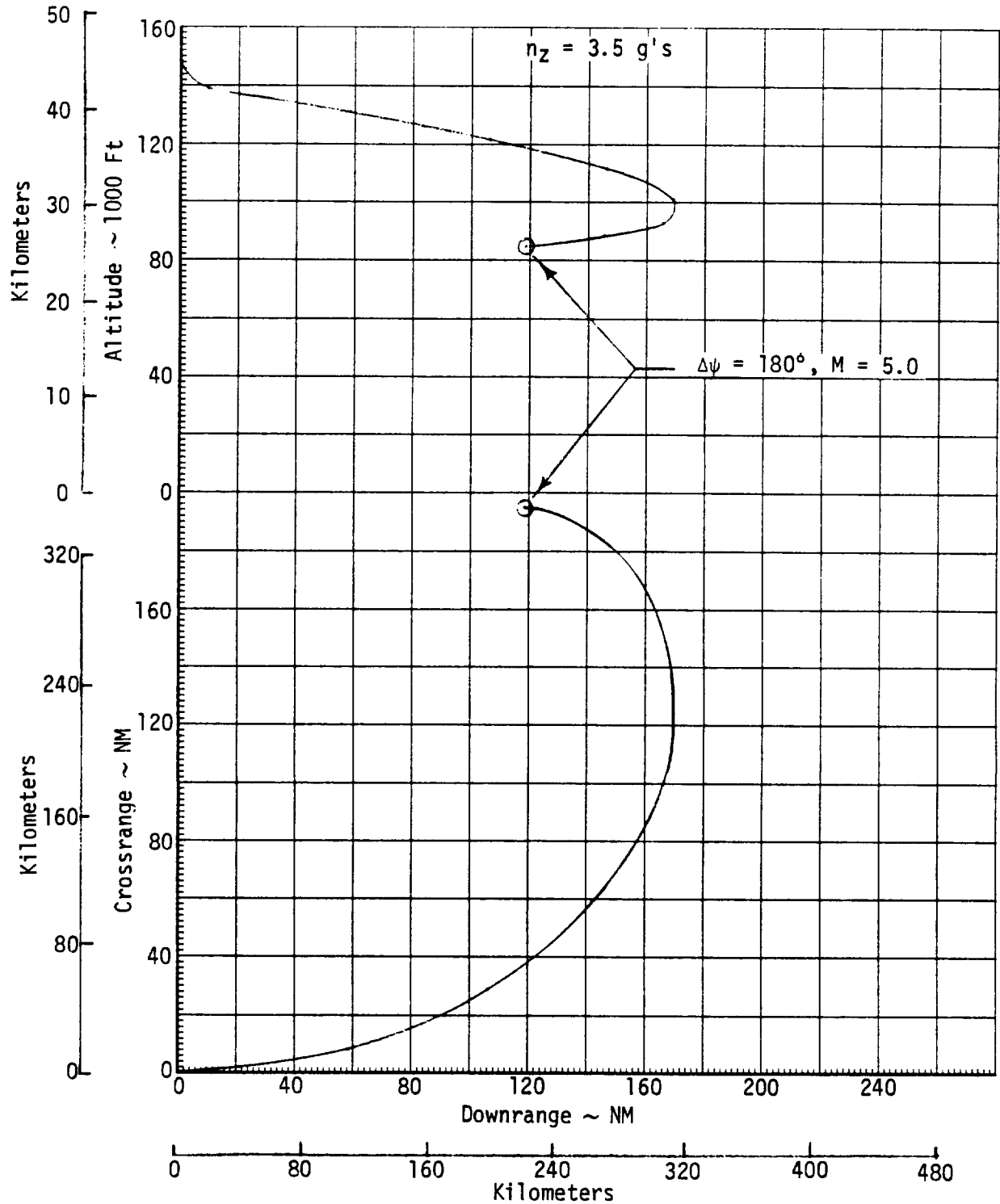
4.2.7.5 (U) Landing Assist - Landing assist capabilities were incorporated in the B233 Configuration by resizing to include an additional fuel allotment for flight path realignment during the landing approach.

4.2.7.6 (U) Single Base Operation - The return phase of the mission for Mach 12 rocket accelerator test vehicles sized for single base operation, was initiated after the 5 minute test cruise with a 3.5 g unpowered windup turn. The trajectory for this maneuver is shown in Figures 4-40 and 4-41. A 20° maximum angle of attack limitation was imposed at the initiation of the windup turn. The Mach number attained at completion of the windup turn was held constant during the powered cruise back. For the Mach 12 vehicle, a 200 nautical mile (370.6 km) unpowered glide to the landing base was allotted. For the Mach 6 vehicles, the cruise to landing base is powered.

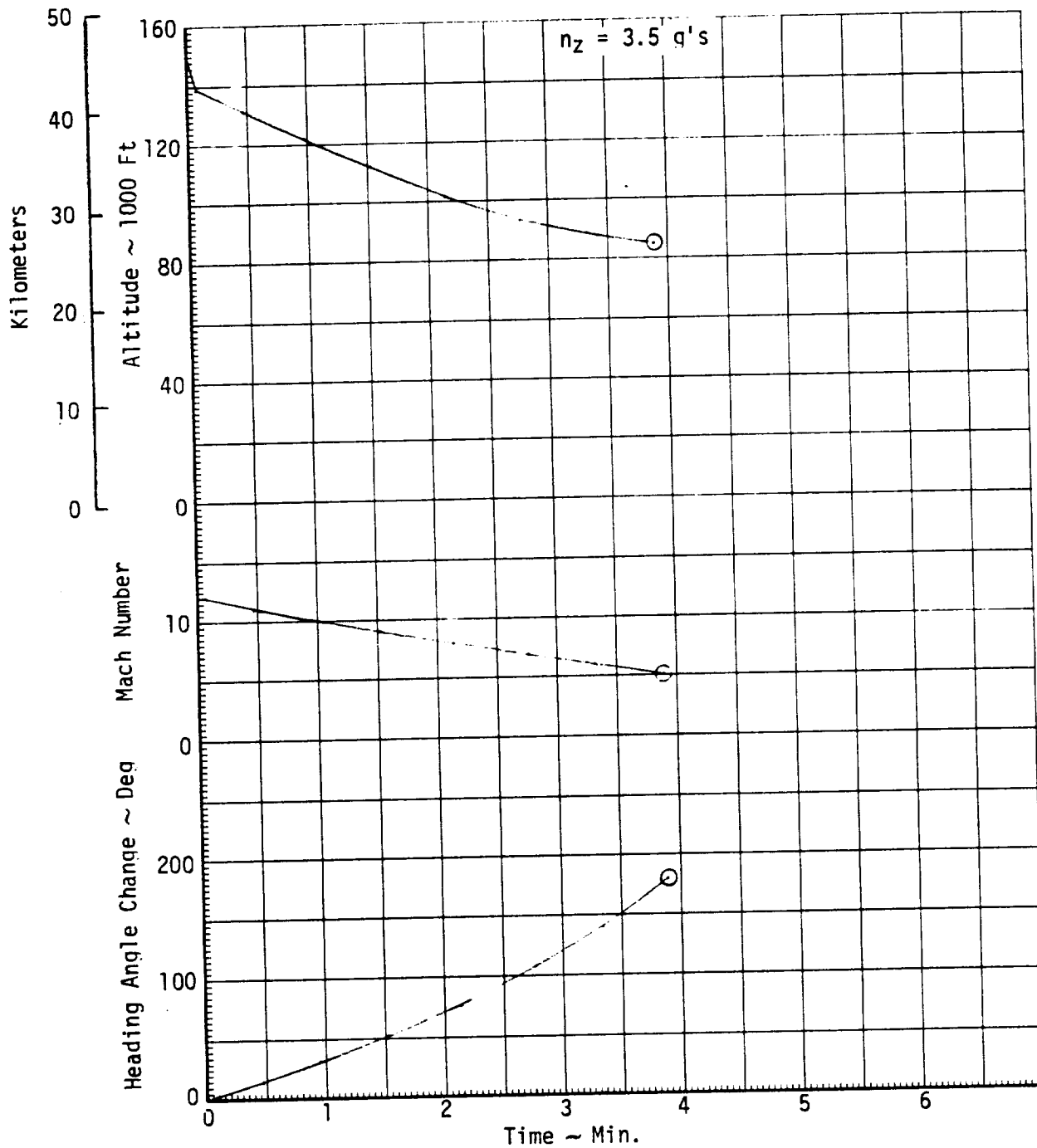
(U) FIGURE 4-39
FLARE ALTITUDE AND TIME ON FINAL REQUIREMENTS



(U) FIGURE 4-40
MACH 12 UNPOWERED WINDUP TURN TRAJECTORY
Rocket All Body Configuration



(U) FIGURE 4-41
MACH 12 UNPOWERED WINDUP TURN TRAJECTORY PARAMETERS
Rocket All Body Configuration



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4.2.8 (U) FLIGHT VEHICLE COSTS - Total system costs were derived for the baseline Phase II flight research vehicles, their parametric variations, and for the seven near-optimum configurations which evolved from the trade-off studies. The costing methodology developed and employed during Phase I, as described in Volume II, Part 2, Section 4.12, was modified and employed throughout Phase II, as described in the remainder of this section.

(U) The scope of effort to obtain and compare both total costs and cost elements during Phase II, prompted the mechanization of the estimating procedure. To this end a computer program was devised which directly parallels the methodology described in Volume II, as modified here. As methodology changes or changes in cost element definitions evolved during Phase II, these changes were incorporated into the computer program.

(U) Remaining paragraphs in this section deal with changes in methodology which occurred during Phase II, ground rule changes and comparison of Phase I and II costs for the same system.

4.2.8.1 (U) Changes In Cost Methodology - A significant change in procedure was instituted to insure development of accurate program cost estimates for aircraft designed to operate at hypersonic speeds. Where former methods estimated the RDT&E and investment costs by lumping airframe design, miscellaneous subsystems design and development and system development testing costs, the revised system estimates these costs individually. While total program costs are comparable using these two systems, element costs differ greatly. These two estimating techniques are compared and results are shown in Section 4.2.8.3.

(U) In Phase II, some changes were made to facilitate the costing process and to provide more visibility in the cost analysis. These changes involved: (1) number of models, (2) format, and (3) parametric cost data and are discussed in the following sections.

4.2.8.1.1 (U) Cost Models - Due to the elimination of the staged VTO and unmanned HTO configurations in Phase II, the number of cost models were reduced from six to four models. These models distinguished by control mode, launch concept, and recovery mode are shown in Figure 4-42.

(U) FIGURE 4-42 FLIGHT RESEARCH VEHICLE COST MODELS - PHASE II

<u>Cost Model</u>	<u>Control Mode</u>	<u>Launch Concept</u>	<u>Recovery Mode</u>
(1)	Manned	HTO	Land Recovery
(2)	Manned	Air Launch	Land Recovery
(3)	Manned	V/HTO	Land Recovery
(4)	Unmanned	Air Launch	Land Recovery

4.2.8.1.2 (U) Cost Format - Two cost elements were removed from the cost format as a result of the elimination of the staged VTO configurations from the spectrum of flight research vehicles and are as follows: (1) the launch investment cost, and (2) the launch service operating cost.

(U) The cost format for Phase II remained basically the same as for Phase I. Phase II factors and cost parameters were refined where possible and further break-outs of appropriate elements in the RDT&E and Investment Costs were made. RDT&E elements are broken out as follows:

- o Airframe Engineering
- o Subsystems Engineering
- o Development Testing
- o Avionics Development
- o Pre-Delivery Flight Test
- o Test Hardware
- o Tooling
- o Equipment Development
- o Propulsion Development
- o Support Equipment Design & System Integration
- o Ground Test Facilities

(U) Investment cost elements are broken out as follows:

- o Airframe Construction
- o Subsystems Construction
- o Avionics
- o Propulsion
- o Support Costs
- o Launch Platform Costs

(U) Operating cost elements are broken out as follows:

- o Range User Cost
- o Escort Aircraft and Logistics
- o Vehicle Refurbishment Cost
- o Propellant Cost
- o AGE Maintenance
- o General Purpose Maintenance Support
- o Transportation Cost
- o Pilot Pay and Support Personnel Pay
- o Launch Platform Operating Cost

4.2.8.1.3 (U) Parametric Cost Data - Major changes were employed in the derivation of the RDT&E and investment costs associated with the airframe and miscellaneous subsystems. In Phase II the method of deriving the RDT&E and investment costs associated with the airframe and miscellaneous subsystems included use of parametric data in the following cost and sub-cost elements: (1) airframe engineering, (2) subsystem engineering, (3) development testing, and (4) production manhours.

(U) The curves presented in Figure 4-43 were used to derive the airframe engineering manhours associated with the design of the airframe structure. The hard point, proprietary, data used to generate these curves are presented in Section 3 of Volume V. Engineering manhours obtained from Figure 4-43 were converted to dollars by application of the engineering rate shown in Section 3 of Volume V. The design engineering costs for the airframe structure vary with Mach number.

(U) In Figure 4-44, the curves used to derive the subsystems engineering manhours required for the design of the subsystems are presented. These curves were derived from the hard point, proprietary, data presented in Section 3 of Volume V. Engineering design costs required for the subsystems are not influenced by Mach number and were converted to dollars by the previously stated engineering rate.

(U) Development testing costs were obtained from the curves presented in Figure 4-45. These curves were derived from the hard proprietary data points presented in Section 3 of Volume V. The engineering rate, shown in Section 3 of Volume V, was used to convert manhours to dollars.

(U) Production manhours required for the airframe structure and subsystems are presented in Figure 4-46. A production labor rate, shown in Section 3 of Volume V, was used to convert manhours to dollars.

4.2.8.2 Cost Estimating Ground Rule Changes - Changes in estimating ground rules for Phase II fall into two categories which deal with (1) aircraft operational cost considerations, and (2) procurement cost considerations.

4.2.8.2.1 (U) Aircraft Operational Cost Considerations - All flight vehicle operating costs are based on a 200-flight program extending over five years. Consideration was given in Phase II to the reliability and maintenance aspects of manned and unmanned vehicles.

(U) In-house studies conducted at McDonnell Douglas show that the total system reliability is less for unmanned vehicle configurations than for manned vehicle configurations, resulting in a lower probability of vehicle recovery. For example, studies conducted for the "Low Speed Unmanned Test Vehicle", Reference (14), show that for a program consisting of 50 flights, three manned vehicles are required, compared to five unmanned vehicles. These quantities are based on the following estimated probabilities:

- o Probability of mission success = .9188 (unmanned vehicle), .9406 (manned vehicle)
- o Probability of vehicle recovery = .9541 (unmanned vehicle), .9816 (manned vehicle)
- o Probability of human success = .995.

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(U) In-house studies have also shown that maintenance requirements are greater for unmanned vehicles, due to the increased complexity and quantity of unmanned system equipment, as well as more stringent equipment reliability requirements.

(U) When the flight vehicle was sized for a single-base and dual-base operation, several factors, besides weight, were found to affect the flight vehicles' total system cost. For example, the AGE, AGE spares, and general purpose maintenance support costs for a dual base operation are twice those for a single base operation. Additionally, transportation costs for dual base configurations are not applicable when single base operations are involved.

4.2.8.2.2 (U) Procurement Cost Considerations - The major changes made in the cost data employed in Phase II are as follows:

(a) (U) Trainer costs were reduced from \$5,000,000 to \$750,000 based on HL-10 trainer costs.

(b) (U) Material cost of columbium was reduced from \$600 per lb (\$1325/KG) to \$500 per lb (\$1104/KG).

(c) (U) Tantalum material (density = 0.606 lbs/in^3 ($.017 \text{ KG/cm}^3$)) was priced at \$600 per lb, (\$1325/KG) with a material complexity factor of 4.0. This material was not used in Phase I.

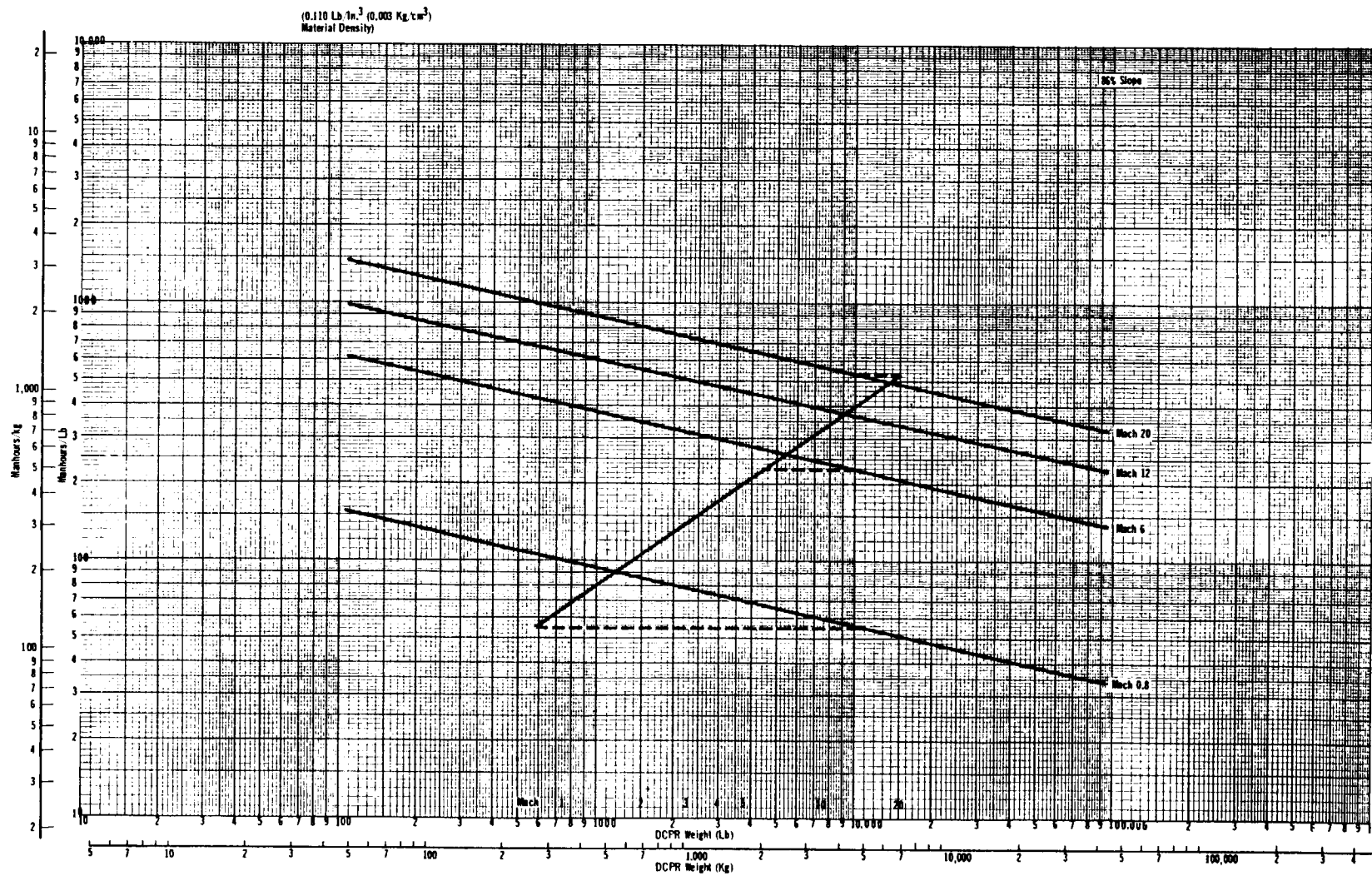
(U) Several major changes were made in costing the propulsion systems. Off-the-shelf engines were considered in the Phase II trade-off studies. Initial spare engines were priced on the following basis:

(a) (U) Five spare ship-sets were allocated for configurations requiring multiple acceleration and cruise engines. It is unlikely that more than one or two engines would require replacement at any one time and that a total of 25 engines would therefore be adequate.

(b) (U) Seven spare ship-sets were allocated for configurations requiring one acceleration and cruise engine similar to the X-15 program.

(U) Vehicle refurbishment costs were computed at 2.5% of their corresponding flight vehicle costs for all configurations, except on the unmanned configurations B-284A and 284B, where they were computed at 2.75% of their corresponding flight vehicle costs, because of increased maintenance requirements.

(U) In Phase II, two C-5A aircraft were utilized for each air-launched candidate program. It was found that research vehicles having a wing span of 32 ft or more required a major modification of the C-5A launch aircraft. The cost is based on relocating the right inboard engine, beefing up the wing, and modifying the wheel pod. This modification was priced at \$1.5M per launch aircraft. The C-5A modification cost associated with the pylon installation was priced at \$75 per lb of attached weight (TOGW of research vehicle).

(U) FIGURE 4-43
REQUIRED ENGINEERING MANHOURS PER LB/KG FOR AIRFRAME STRUCTURE

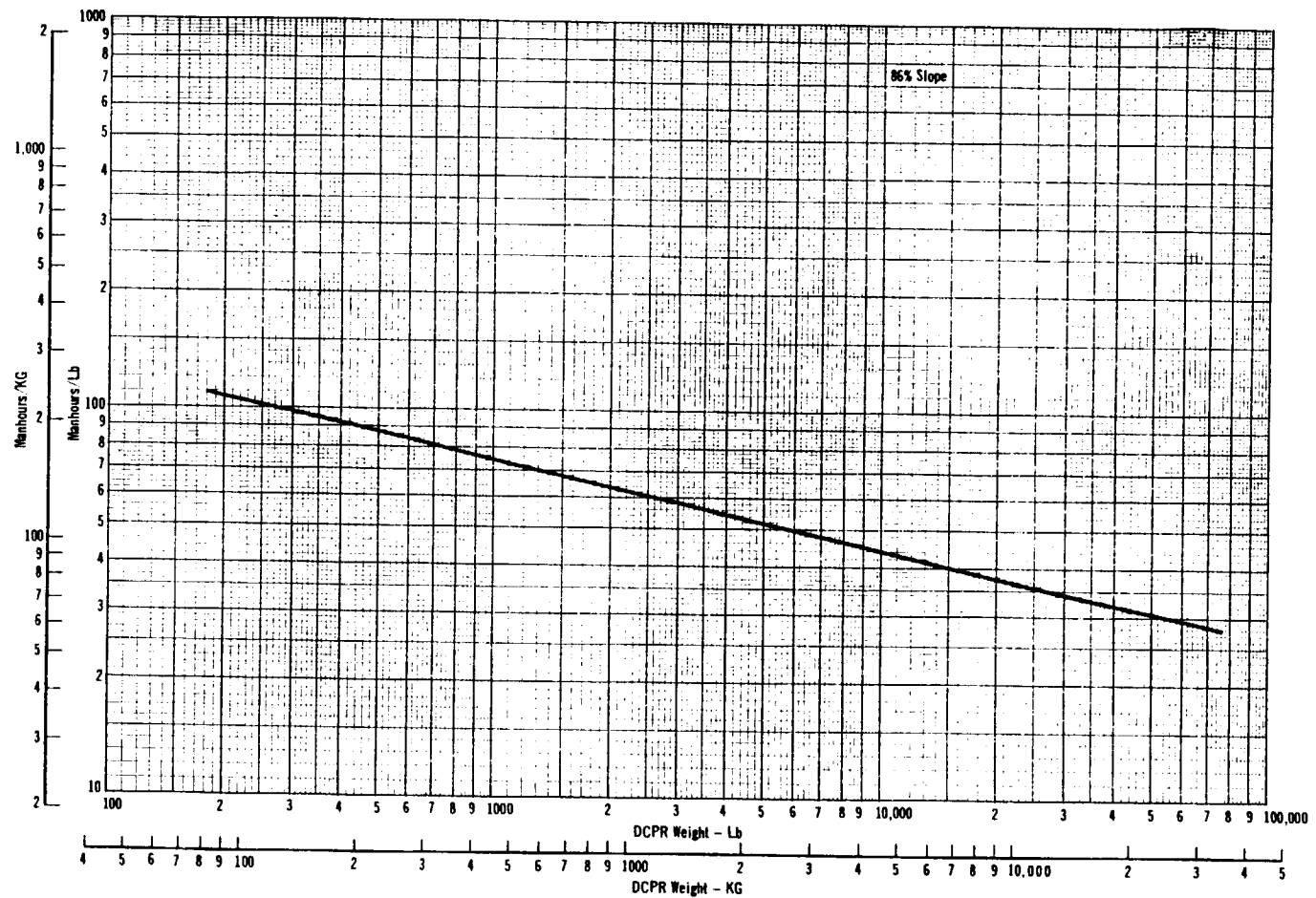
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11-59

FOLDOUT FRAME 1

FOLDOUT FRAME 2

(U) FIGURE 4-44
REQUIRED ENGINEERING MANHOURS PER LB/KG FOR AIRCRAFT SUBSYSTEMS

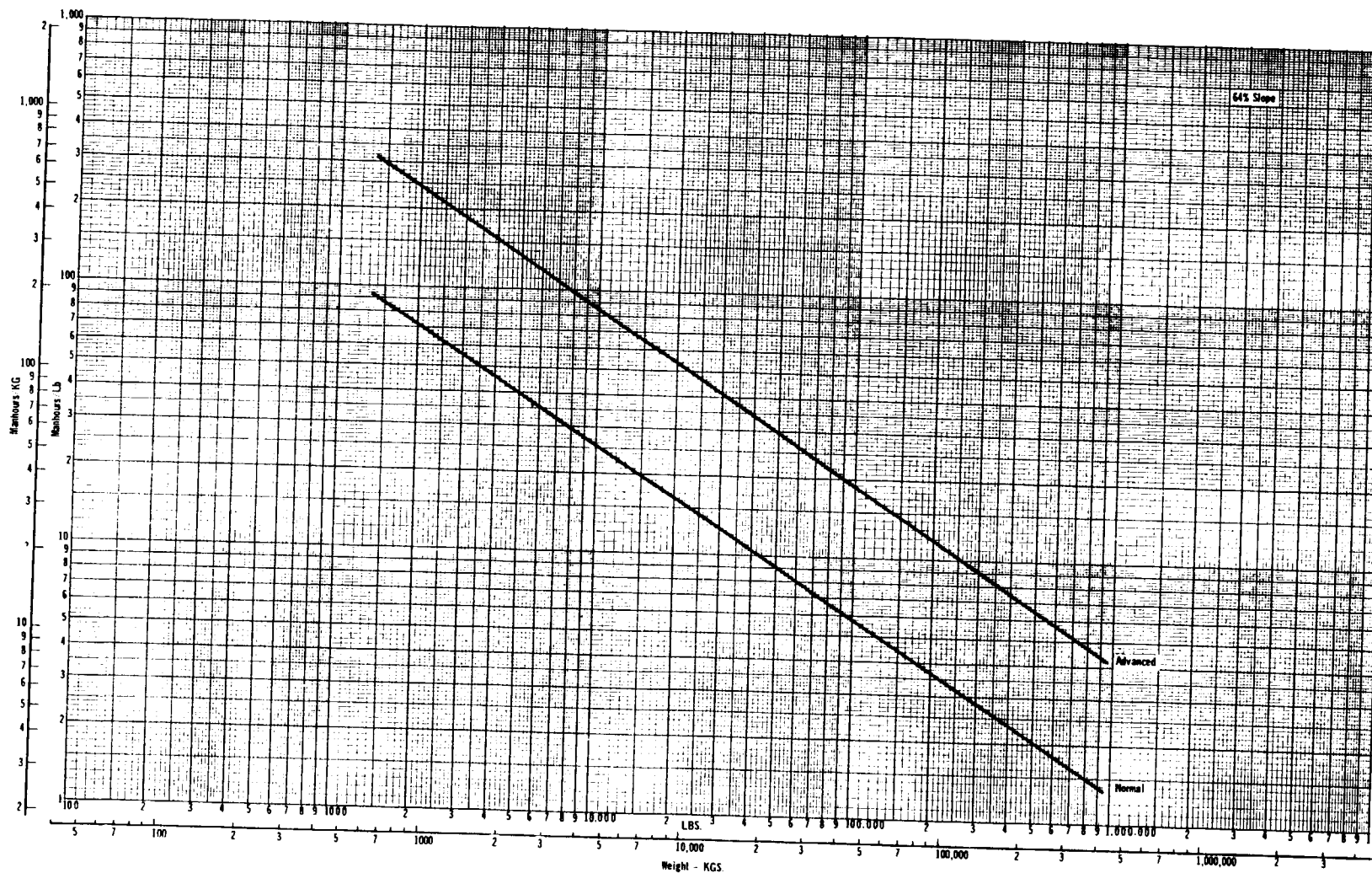


MCDONNELL AIRCRAFT
11-60

FOLDOUT FRAME 1

(U) FIGURE 4-45
REQUIRED ENGINEERING MANHOURS PER LB/KG FOR AIRFRAME TESTING

FOLDOUT FRAME 2

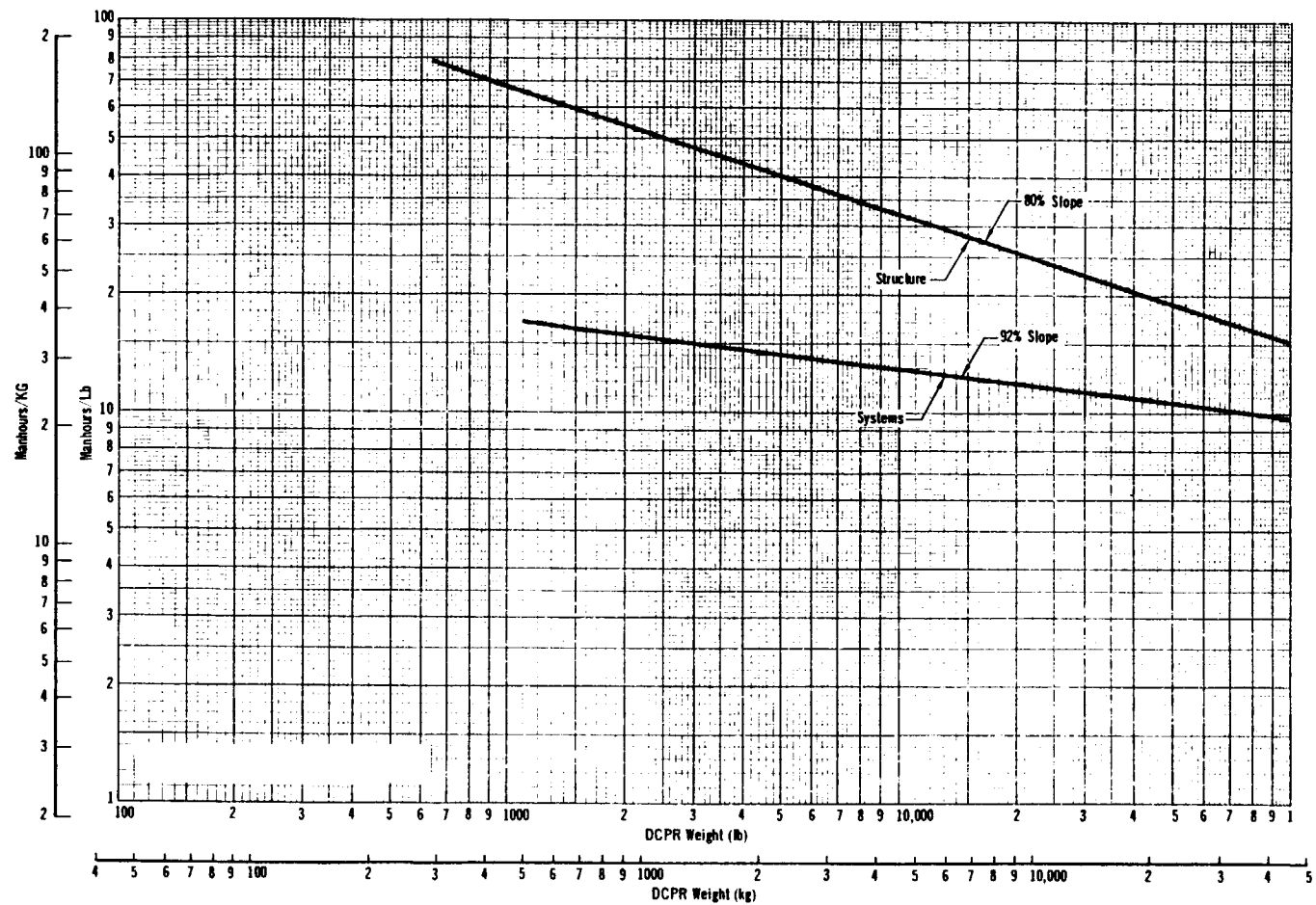


MCDONNELL AIRCRAFT

FOLDOUT FRAME 1

FOLDOUT FRAME 2

(U) FIGURE 4-46
REQUIRED EXPERIMENTAL PRODUCTION MANHOURS PER LB/KG
FOR AIRFRAME STRUCTURE AND SHINGLES



4.2.8.3 (U) Comparison of Estimating Methods - As indicated in Section 4.2.8.1, significant changes in the estimating procedure were made during Phase II. In this section a comparison is made between the Phase I and Phase II cost estimating procedures. Additionally, these two methods are compared as they apply to a specific example and thereby results are compared.

4.2.8.3.1 (U) Method Changes - Each step in estimating the cost of a flight vehicle program is outlined in Figure 4-47. RDT&E, Investment, and Operating Cost are listed, relationships and data sources are shown for both the Phase I and Phase II methods. In each case where a change in procedure is made the rationale concerning the change is given.

(U) Computation of the Airframe Engineering portion of RDT&E costs during Phase I was based on a parametric relationship between the total DCPR weight and engineering manhours. During Phase II, Airframe Design and Development Costs were revised by separately computing airframe engineering cost on the basis of the DCPR structural weight and by computing miscellaneous subsystems costs on the basis of systems weight and development testing costs for airframe and systems separately. Parametric relationships were employed in each case.

(U) These changes reflect the fact that while many cost factors in aircraft development are direct functions of the maximum speed capability of the aircraft, certain other factors are not affected to the same degree. Avionic equipment and displays, controls, hydraulic systems, etc., are relatively independent of speed. Structures, fuel systems, engines and environmental control systems, on the other hand are more directly affected by increases in speed.

(U) A lump sum was added during Phase II to account for the conversion of turbojet engines from JP fuel to hydrogen.

(U) Manpower rates were reviewed and adjusted to coincide with the revised cost element breakout. Consequently, these revised rates require a nearly complete set of computations as indicated by the Phase II column of Figure 4-47.

(U) Investment Cost was revised in a manner similar to the RDT&E cost. Labor and material costs are parametrically keyed to the DCPR structural weight and to the miscellaneous subsystems DCPR weight, as opposed to being collectively related to the DCPR weight. In addition to the more detailed breakdown of cost elements, the estimation of investment costs was revised due to revised cost data, and labor rates as shown by the second and third pages of Figure 4-47.

(U) Single base operation requires 1/2 the AGE and spares required for two base operation. When two base operation is involved, spare parts and equipment are stocked on the basis of aircraft maintenance being required at either base. With one base operation this requirement is halved.

(U) The methods used in developing Operating Cost estimates during Phase I were retained without change during Phase II. Only the LOX utilization factor was revised during Phase II from 1.54 to 1.50.

FOLODOUT FRAME 1

FOLODOUT FRAME 2

(U) FIGURE 4-47 SYSTEMS COST COMPARISON SUMMARY

Cost Categories and Cost Elements	Phase I		Phase II		Differences (Rationale)
	Cost Estimating Relationships and Data Employed				
	Relationships and Data	Computation, Data Source, and Comments	Relationships and Data	Computation, Data Source, and Comments	
I. RDT&E Costs					
1. Airframe Design and Development					
A. Airframe Engineering	Total DCPR Weight \$ Lb Mach No. Material Density	Engr. Price per Lb vs DCPR Wt Figure 4-99 Vol. II, Part 2	Structural DCPR Weight Man Hours Lb and Mach No. Material Density	DCPR Weight Summary Engr. Man Hours/Lb (Struct.) Figure 4-43 Material Density Summary	Phase II Data Refined to Allow for Separate Airframe, Systems, and Development Test Costs
B. Misc. Subsystem Des. & Dev.	Combined with Airframe Engineering and Computed in Total		Subsystem DCPR Weight Man Hours Lb	DCPR Weight Summary Engr. Man Hours/Lb (Systems) Figure 4-44	
a. Subsystem Engineering	Eq. Wt. x \$ Lb x Rate Factors	CFE Cost Summary	Eq. Wt. x \$ Lb x Rate Factors	CFE Cost Summary	
b. Equipment Development	Combined with Airframe Engineering and Misc. Subsystem Engineering and Computed in Total		Total DCPR Weight Man Hours Lb for Conv. and Adv. Material x Wt of Conv. and Adv. Material	DCPR Weight Summary Engineering Man Hours/Lb Figure 4-45 (Testing)	Rate Factors Changed in Phase II
C. Development Testing	% of Flight Vehicle Inventory Cost	Direct Computation	Same as Phase I.	Same as Phase I	Phase II Data Refined to Allow for Separate Airframe, Systems, and Dev. Test Costs. Man Hours Lb Converted to \$ Lb by Machine Program.
D. Test Hardware					No Change
E. Pre-Delivery Flight Test					
a. Engineering	\$ Man Hour x No. of Man Hours Required	Man Hours Based on F-4 Experience	\$ Man Hours x No. of Man Hours Required	Man Hours Based on F-4 Experience	\$ Man Hours Changed in Phase II
b. Production	\$ Man Hour x No. of Man Hours Required	Man Hours Based on F-4 Experience	\$ Man Hours x No. of Man Hours Required	Man Hours Based on F-4 Experience	\$ Man Hour Changed in Phase II
c. Material	F-4 Material x Ratio of Total \$ Lb to Base \$ Lb x Proc. Exp. Factor x G&A Factor	Direct Computation	F-4 Material x Ratio of Total \$ Lb to Base \$ Lb x Proc. Exp. Factor x G&A Factor	Direct Computation	Procurement and G&A Factors Changed in Phase II
d. Travel and Per Diem	F-4 Dollars x Econ. Adj. Factor x G&A Factor	Direct Computation	F-4 Dollars x Econ. Adj. Factor x G&A Factor	Direct Computation	G&A Factor Changed in Phase II.
e. Pilot's Salary	F-4 Dollars x Econ. Adj. Factor x G&A Factor	Direct Computation	F-4 Dollars x Econ. Adj. Factor x G&A Factor	Direct Computation	G&A Factor Changed in Phase II.
f. Propellants	\$ Lb x Lbs Ftl x No. of Ftl's	Direct Computation	Same as Phase I.	Same as Phase I	No Change
g. Transportation	\$ Round Trip x No. of Round Trips	Direct Computation	Same as Phase I.	Same as Phase I	No Change
2. Tooling	Total DCPR Weight \$ Lb and Rate Factors Wt Complexity of Conventional and Advanced Materials	DCPR Weight Summary	Total DCPR Weight Man Hours Lb and Rate Factors Wt Complexity of Conventional and Advanced Material	DCPR Weight Summary	In Phase II, Man Hours Lb are Converted to \$ Lb in the Machine Program. Rate Factors were Changed in Phase II
3. Avionics Development	Avionics Wt x \$ Lb x Rate Factors	DCPR Weight Summary	Avionics Wt x \$ Lb x Rate Factor	CFE Cost Summary	Rate Factors Changed in Phase II.
4. Propulsion Development					
A. J2S Rocket	\$10,000,000	Direct Computation	Same as Phase I.	Same as Phase I	No Change
B. Turbojet	None	Development Cost Paid for in USAF Program	None (JP) \$25,000,000 (LH2)	Direct Computation	\$25,000,000 Added for LH2 Conversion
C. Ramjet	\$170,000,000	Direct Computation	Same as Phase I.	Same as Phase I	No Change
D. CSJ	\$265,000,000	Direct Computation	Same as Phase I.	Same as Phase I	No Change
E. Storable Fuel Rocket	\$ Lb of Vacuum Thrust	Rkt Eng. Prop. Costs	Not Applicable	Not Applicable	No Change
F. LR-129 Rocket	\$ Lb of Vacuum Thrust	Rkt Eng. Prop. Costs	Same as Phase I.	Same as Phase I.	Storable Rockets Not Employed in Phase II.
G. RL-10 Rocket	Not Applicable	Not Applicable	\$24,000,000	Direct Computation Pratt & Whitney Est.	No Change
H. Turboramjet	\$ Lb of Sea Level Thrust	Turboramjet Prop. Costs 1.5 Factor Applied to LH2 Engines Vol. II, Part 2	Not Applicable	Not Applicable	Turboramjet not Employed in Phase II
5. Support Equipment Design & System Int.	(CER) 0.047 (OWE) 0.59	Direct Computation	Same as Phase I.	Same as Phase I	No Change
6. Ground Test Facilities					
A. Turbojet (LH ₂)	\$3,500,000	Direct Computation	Same as Phase I.	Same as Phase I	No Change
B. Ramjet	\$8,000,000	Direct Computation	Same as Phase I.	Same as Phase I	
C. CSJ SJ	\$50,000,000	Direct Computation	Same as Phase I.	Same as Phase I	
D. Turboramjet	\$5,000,000	Direct Computation	Not Applicable	Not Applicable	

EXHIBIT FRAME 2

EXHIBIT FRAME 1

(U) FIGURE 4-47 SYSTEMS COST COMPARISON SUMMARY (Continued)

Cost Categories and Cost Elements	Phase I		Phase II		Differences (Rationale)
	Cost Estimating Relationships and Data Employed				
	Relationships and Data	Computation, Data Source, and Comments	Relationships and Data	Computations, Data Source, and Comments	
II. Investment Costs					
I. Flight Vehicles					
A. Airframe					
a. Labor	Total DCPR Weight \$ Lb Weighted Complexity	Production Price per Lb vs DCPR Wt Vol. II, Part 2	Structure DCPR Weight Man Hours Lb Weighted Complexity Factor	DCPR Weight Summary Prod. Man Hours Lb Figure 4-46 Production Complexity	Phase II Data Refined to Allow for Separate Airframe and Systems Costs. Man Hours Lb Converted to \$ Lb by Machine Program.
b. Material	Total DCPR Wt x Material Cost (\$ Lb) by Material Type	Direct Computation	Structure DCPR Weight x Material Cost (\$ Lb.) by Material Type	Direct Computation	Phase II Data Refined to Allow for Separate Airframe and Subsystem Material Costs.
B. Misc. Subsystems					
a. Labor	Combined with Airframe Labor and Computed as a Total Labor Cost		Subsystems DCPR Weight x Man Hours Lb	DCPR Weight Summary Prod. Man Hours Lb Figure 4-46	Phase II Data Refined to Allow for Separate Airframe and Systems Labor Costs. Man Hours Lb Converted to \$ Lb by Machine Program
b. Material	Combined with Airframe Material and Computed as a Total Material Cost		Subsystem DCPR Weight x Material Cost (\$ Lb) by Material Type	Direct Computation	Phase II Data Refined to Allow for Separate Airframe and Subsystem Material Costs.
c. Equipment	\$ Lb x Eq. Wt x Rate Factors	Equipment Cost Summary	\$ Lb x Eq. Wt x Rate Factors	Equipment Cost Summary	Rate Factors Changed in Phase II
C. Propulsion					
a. J2S Rocket	\$2,500,000	Direct Computation	\$1,600,000	5% Factor Applied for Gimbaling	\$900,000 Decrease in Phase II.
b. Turbojet	\$ Lb of Sea Level Thrust	Turbojet Prop. Costs	\$583,000	GE Cost for Qty. Prod. F-100-GE-100	
c. Ramjet	\$550 Lb of Dry Engine Weight	Direct Computation	Same as Phase I	Same as Phase I	No Change
d. CSJ SJ	\$ Module	Direct Computation	Same as Phase I	Same as Phase I	No Change
e. Storable Fuel Rocket	\$ Lb of Vacuum Thrust	Rkt. Eng. Prop. Costs	Not Applicable	Not Applicable	Storable Rkts. not Employed in Phase II.
f. LR-129 Rocket	\$ Lb of Vacuum Thrust	Rkt. Eng. Prop. Costs	Same as Phase I	Same as Phase I	No Change
g. RL-10 Rocket	Not Applicable	Not Applicable	\$460,000	Direct Computation P&W Estimate	
h. Turboramjet	\$ Lb of Sea Level Thrust	Turboramjet Prop. Costs	Not Applicable	Not Applicable	Internal Estimate
D. Avionics	Avionics Equip. Wt. x \$ Lb x Rate Factor \$100,000 Added to Unmanned Vehicle	Direct Computation	Avionics Equip. Wt. x \$ Lb x Rate Factor \$100,000 Added to Unmanned Vehicle	Direct Computation	Rate Factors Changed in Phase II
2. Support Costs					
A. AGE					
a. Single Base Operation	Not Applicable	Not Applicable	7.5% x Σ Flight Vehicle Inv. Cost	Direct Computation	Single Base Operations Employed in Phase II.
b. Two Base Operation	15% x Σ Flight Vehicle Inv. Cost	Direct Computation	Same as Phase I.	Same as Phase I.	No Change
B. Training Equipment	\$5,000,000	Obtained from Reference 17	\$750,000	Obtained from NASA FRC	\$4,250,000 Diff.
C. Initial Stocks					
a. Engines					
Single Accelerator Cr. Eng.	7 Eng. Sets x Cost Per Engine	Direct Computation	Same as Phase I.	Same as Phase I.	No Change
Multiple Accelerator Cr. Eng.	Not Applicable	Not Applicable	5 Eng. Sets x Approp. Cost per Engine	Direct Computation	5 Spare Sets Employed for Multiple Engine Configurations
b. AGE Spares	10% x Σ AGE Investment Cost	Direct Computation	Same as Phase I	Same as Phase I.	No Change
D. Initial Training	20 Pilots in Program 20 x 50,000 Dollars per Pilot	Direct Computation	Same as Phase I	Same as Phase I.	No Change

FOLDOUT FRAME 1

FOLDOUT FRAME 2

(U) FIGURE 4-47 SYSTEMS COST COMPARISON SUMMARY (Continued)

Cost Categories and Cost Elements	Phase I		Phase II		Differences (Rationale)
	Cost Estimating Relationships and Data Employed				
	Relationships and Data	Computation, Data Source, and Comments	Relationships and Data	Computation, Data Source, and Comments	
II. Investment Costs					
E. Initial Transportation	2% x AGE, AGE Spares, Eng. Spares, Refurb. Materials, Prop. and Training Equipment Costs	Direct Computation	Same as Phase I	Same as Phase I	No Change
3. Launch Platform Cost					
A. C-5A B-52 Modification					
a. Pod Installation	\$75/Lb x Flight Vehicle (TOGW)	Direct Computation	Same as Phase I	Same as Phase I	No Change
b. Major Mod. (C-5A)	Not Applicable	Not Applicable	\$1,500,000 per C-5A MCAIR Estimate	Direct Computation	No Change
B. VTO Launcher	\$1,000,000 per Launcher	Direct Computation	Same as Phase I	Same as Phase I	No Change
4. Launch Vehicle Cost					
A. Thor	0.5 M Dollars x No. of Thors	Direct Computation Douglas Estimate	Not Applicable	Not Applicable	No Change
B. SLV-3	1.65 M Dollars x No. of SLV-3's	Direct Computation GD Estimate	Not Applicable	Not Applicable	No Change
III. Operating Costs					
1. Range User Cost					
A. Range Operating Cost	21.5 R Dollars Flight x No. of Flights	Direct Computation	Same as Phase I	Same as Phase I	No Change
B. Base Engr. Support	No. of Years x 25.31 K Dollars Year	Direct Computation	Same as Phase I	Same as Phase I	No Change
2. Escort Aircraft and Logistics	11.5 K Dollars Flight x No. of Flights	Direct Computation	Same as Phase I	Same as Phase I	No Change
3. Vehicle Refurbishment Cost	1.5% - 2.5% x Flight Vehicle Unit Investment Cost	1.5% (Mach 4.5), 2.0% (Mach 6), 2.5% (Mach 12)	2.5-2.75% x Flight Veh. Cost x No. of Flts	Same as Phase I	Refined Estimate in Phase II / (Inv. Cost), (No. of Flights)
4. Propellant Cost					
A. LH2	\$ 13851 Lb x 1.5 (Utilization Factor) x Lbs of Fuel Req. Flt x No. of Flights	Direct Computation	Same as Phase I	Same as Phase I	No Change
B. LOX	\$ 0061 Lb x 1.54 (Utilization Factor) x Lbs of Ox. Req. Flt x No. of Flights	Direct Computation	1.5 Utilization Factor Used	Direct Computation	Utilization Factor Changed in Phase II
C. JP	\$ 015 Lb x 1.1 (Utilization Factor) x Lbs of Fuel Req. Flight x No. of Flights	Direct Computation	Same as Phase I	Same as Phase I	No Change
D. Aero 50	\$ 47 Lb x 1.1 (Utilization Factor) x Lbs of Fuel Req. Flt x No. of Flights	Direct Computation	Not Applicable	Not Applicable	Storable Rkt Fuel not Employed in Phase II.
E. N ₂ O ₄	\$ 12 Lb x 1.1 (Utilization Factor) x Lbs of Ox. Req. Flt x No. of Flights	Direct Computation	Not Applicable	Not Applicable	Storable Rkt Oxidizer not Employed in Phase II.
5. AGE Maint. (Labor Only)	3% x AGE Inv. Cost x No. of Bases	Direct Computation	Same as Phase I	Same as Phase I	No Change
6. General Purpose Maint. Sup.	100K Dol. Yr x No. of Yrs x No. of Bases	Direct Computation	Same as Phase I	Same as Phase I	No Change
7. Transportation Cost					
A. Phase I					
a. Configs. 200, 205, 207	\$2,482 Rd. Trip x No. of Rd. Trips	Edwards to Walker AFB and Return	Not Applicable	Not Applicable	Based on \$811 per Flying Hour (MATS Airlift Industrial Fund Concept) for C-5A Transport.
b. Configs. 210, 213, 214	\$3,438 Rd. Trip x No. of Rd. Trips	Edwards to Dyess AFB and Return			
c. Configs. 220, 221	\$2,578 Rd. Trip x No. of Rd. Trips	Barksdale AFB to Kennedy and Return			
d. Configs. 204, 234, 232, 233, 250, 251, 250, 252, 253, 254, 255, 256, 284, 285	\$7,932 Rd Trip x No. of Rd. Trips	Edwards to Homestead AFB and Return			
e. Config. 257	\$12,440 Rd Trip x No. of Rd. Trips	Edwards to Loring AFB and Return	None	Vehicles Return to Launch Site	Based on \$811 per Flying Hours (MATS Airlift Industrial Fund Concept) for C-5A Transport.
f. Configs. 270, 271, 280, 281 & 282	\$7,736 Rd Trip x No. of Rd. Trips	Edwards to Kennedy and Return			
B. Phase II					
a. Configs. B207 & B212	Not Applicable	Not Applicable	\$7,932 Rd Trip x 200 Trips	Edwards to Homestead AFB and Return	
b. Configs. B232, B233, B257 & B284	Not Applicable	Not Applicable	\$7,735 Rd Trip x 200 Trips	Edwards to Kennedy and Return	
c. Config. B260	Not Applicable	Not Applicable			

MCDONNELL AIRCRAFT

1-64

FOLDOUT FRAME 1

FOLDOUT FRAME 2

(U) FIGURE 4-47 SYSTEMS COST COMPARISON SUMMARY (Continued)

Cost Categories and Cost Elements	Phase I		Phase II		Differences (Rationale)
	Cost Estimating Relationships and Data Employed				
	Relationships and Data	Computation, Data Source, and Comments	Relationships and Data	Computation, Data Source, and Comments	
III. Operating Cost					
8. Pilot Pay and Support Per Pay					
A. Pilot Pay	4 Pilots per Yr x 50K Dol. Pilot x 5 Yrs	Direct Computation	Same as Phase I	Same as Phase I	No Change
B. Support Per Pay	2 NMPR x 20K Dol. Yr. x No. of Yrs. NMPR = .50120 LPM (TOGW 344,000) 0.33	Direct Computation	Same as Phase I	Same as Phase I	No Change
9. Launch Platform Op. Cost					
A. C-5A	Cost per FIL x No. of FILs.	Direct Computation Based on \$1,284 Fly. Hr., Ref. AFM-172-3, 27 Apr. 69	Same as Phase I	Same as Phase I	No Change
B. B-52	Cost per FIL x No. of FILs.	Direct Computation Based on \$1,396 Fly. Hr., Ref. AFM-172-3, 27 Apr. 69	Not Applicable	Not Applicable	No Change
C. VTO Launcher	5% x Launcher Inv. Cost x No. of Yrs.	Direct Computation	Same as Phase I.	Same as Phase I	Ref. Rand Rpt. R-287 No Change
10. Launch Service Cost					
A. Thor	.195K Dol. x No. of Flights	Direct Computation Based on Douglas Estimates	Not Applicable	Not Applicable	No Change
B. SLV-3	.2479K Dol. x No. of FILs.	None Based on GD Estimates	Not Applicable	Not Applicable	No Change

Abbreviations: DCPR - Defense Contractor's Progress Report
K - Thousand
M - Million
NMPR - No. of Maintenance Personnel Required
LPM - Launches per Month

AFM - Air Force Manual
Wt - Weight
GD - General Dynamics
TOGW - Takeoff Gross Weight

Eq - Equipment
Adj. - Adjustment
Proc - Procurement
Exp - Expense

Rkt - Rocket
G&A - General and Administration
FIL - Flight

4.2.8.3.2 (U) Cost Comparison - Changes in the cost estimating procedure for Phase II were discussed in the previous section. Costs relative to Configuration 233 were computed using both the Phase I and Phase II methods and are compared in Figure 4-48. Reductions in RDT&E costs are nearly matched by increases in the investment and operating costs. Note that the change in operating cost is a direct function of the change in flight vehicle unit investment cost.

(U) FIGURE 4-48
COST COMPARISON SUMMARY - PHASE I TO PHASE II METHODS
(Millions of 1970 Dollars)

Cost Categories and Cost Elements		Configuration 233 (Phase I)	
		Phase I Method	Phase II Method
I. RDT&E Costs	No. of Flights	180	180
1. Airframe Design and Development			
A. Airframe Design			55.332
B. Miscellaneous Subsystem Design and Development		110.070	6.512
C. Development Testing			2.635
D. Test Hardware		3.221	4.428
E. Pre-Delivery Flight Test		3.552	3.336
Sub Total		116.843	72.243
2. Tooling		8.124	7.862
3. Avionics Development		3.154	3.118
4. Propulsion Development		127.199	127.199
5. Support Equipment Design and System Int.		17.136	17.136
6. Ground Test Facilities		-	-
Total		272.456	227.558
II. Investment Costs			
1. Flight Vehicles			
A. Airframe		16.787	21.119
B. Miscellaneous Subsystems			1.659
C. Propulsion		1.299	1.299
D. Avionics		1.015	0.947
Unit Cost (1) Vehicle		19.101	25.024
Unit Cost (3) Vehicles		57.303	75.072
2. Support Costs			
A. AGE		8.595	11.261
B. Training Equipment		5.000	0.750
C. Initial Stocks (Engs. and AGE Sps.)		9.953	10.219
D. Initial Training		1.000	1.000
E. Initial Transportation		1.168	1.355
Sub Total		25.716	24.585
3. Launch Platform Cost		11.205	11.205
Total		94.224	110.862
III. Operating Costs (5 Years)			
1. Range User Costs		5.136	5.136
2. Escort Aircraft and Logistics		1.998	1.998
3. Vehicle Refurbishment Cost		85.954	112.608
4. Propellant Cost		0.494	0.494
5. AGE Maintenance (Labor Only)		1.290	1.290
6. General Purpose Maintenance Sup.		0.500	1.000
7. Transportation Cost		1.428	1.428
8. Pilot Pay and Sup. Per Day		18.200	18.200
9. Launch Platform Operation Cost		2.286	2.286
Total		117.286	144.440
Grand Total		483.966	482.860

4.3 BASELINE VEHICLE DESIGN AND PERFORMANCE

(U) The Phase I screening of configurations resulted in the selection of seven vehicle concepts for further study during Phase II. These basic concepts were refined in accordance with the design requirements set forth in Section 4.1 and the methods and approaches described in Section 4.2 and used as the baseline for Phase II trade-off studies. The mission employed in sizing these aircraft consists of an acceleration - climb to the test Mach number at an altitude corresponding to equilibrium for $(L/D)_{\max}$, a five minute constant Mach number cruise, and an unpowered descent. The climb profiles employed are shown in Figure 4-49. A summary of pertinent vehicle sizing information is presented in Figure 4-50. A description of the key design characteristics, the vehicle performance and flight characteristics, and the vehicle weight and cost, for each of the seven baseline aircraft follows.

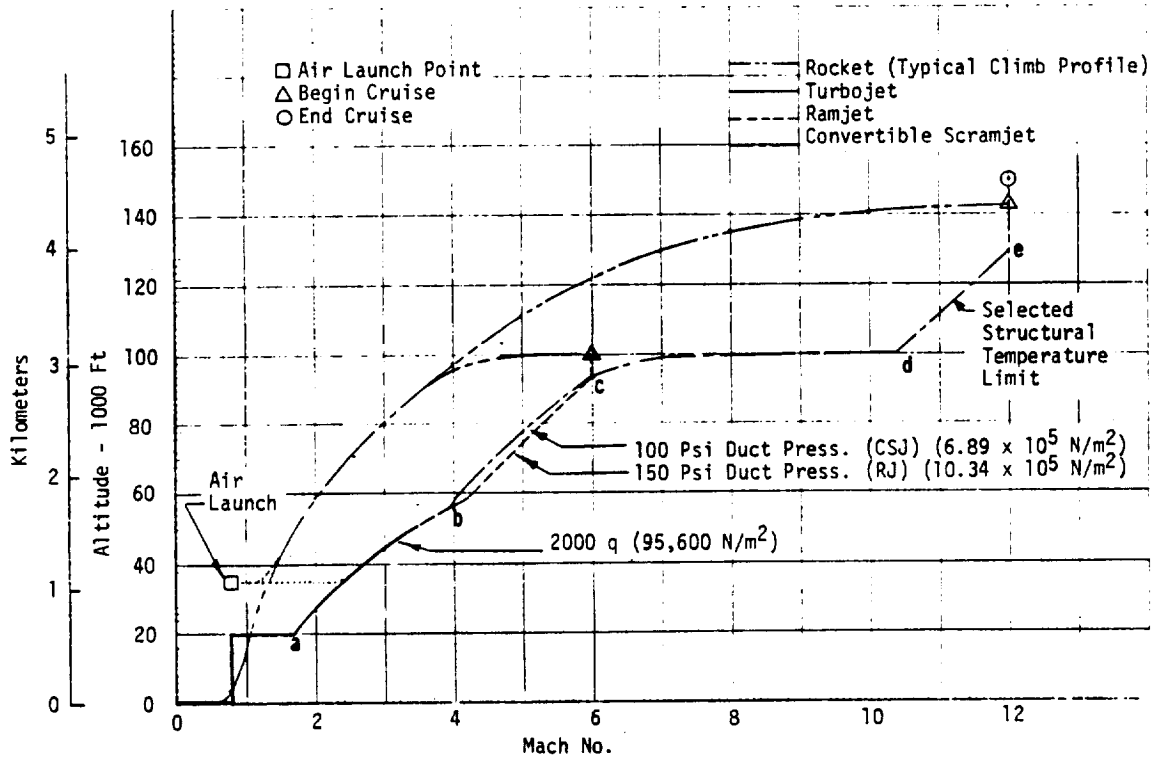
4.3.1 (U) ROCKET - RAMJET CONFIGURATION (CLASS B207) - Design B207A, shown in Figure 4-51, is the baseline Mach 6 configuration representing an all-body, air launched, manned vehicle approach. It is powered by a rocket engine during climb and acceleration and a ramjet for the five minute cruise. The aerodynamic characteristics, performance capabilities, weights and cost information are summarized in Figure 4-52. The change from rocket power to ramjet operation for Mach 6 cruise results in a net drag reduction, due to the fact that the base drag increase incurred at the rocket nozzle is lower than the ramjet engine off drag. For the Mach 6 test condition, a maximum range of 700 nmi (1300 km) results corresponding to a total flight time of 22 minutes.

(U) The propulsion system for the B207A aircraft consists of a rocket engine for acceleration, and a mixed compression inlet and ramjet engine for Mach 6 cruise. A scaled Pratt and Whitney Aircraft (PWA) LR129 High Chamber Pressure rocket is the selected acceleration engine; a two-dimensional hydrogen-fueled ramjet based on the axisymmetric Marquardt MA 145-XAB design is the selected cruise engine.

(C) The scaled version of the PWA LR-129 rocket engine is 60 in (152 cm) long, and has a conventional bell nozzle with an expansion ratio of 75 and an exit diameter of 30 in (75 cm). The propellants are LO_2 and LH_2 . The engine incorporates both a pre-burner and a main burner in the basic cycle and is regeneratively and transpiration cooled with hydrogen fuel. At full power, the rocket operates at 3000 psi (2060N/cm²) chamber pressure at a mixture ratio of 6.0. The engine is operated at full power from launch to the start of cruise and is shut down during cruise. The engine is scaled to produce 57,600 lb (256,000N) vacuum thrust which results in a vehicle vacuum thrust/takeoff gross weight ratio (TVAC/TOGW) of 1.5. The vacuum propellant specific impulse is 451 seconds.

(U) The inlet for the cruise propulsion system is a two-dimensional, mixed compression configuration. The inlet has a single, overhead, external, insulated (uncooled) ramp located in the vehicle compression field. Throat area variation is provided by hinged ramps in the inlet and diffuser.

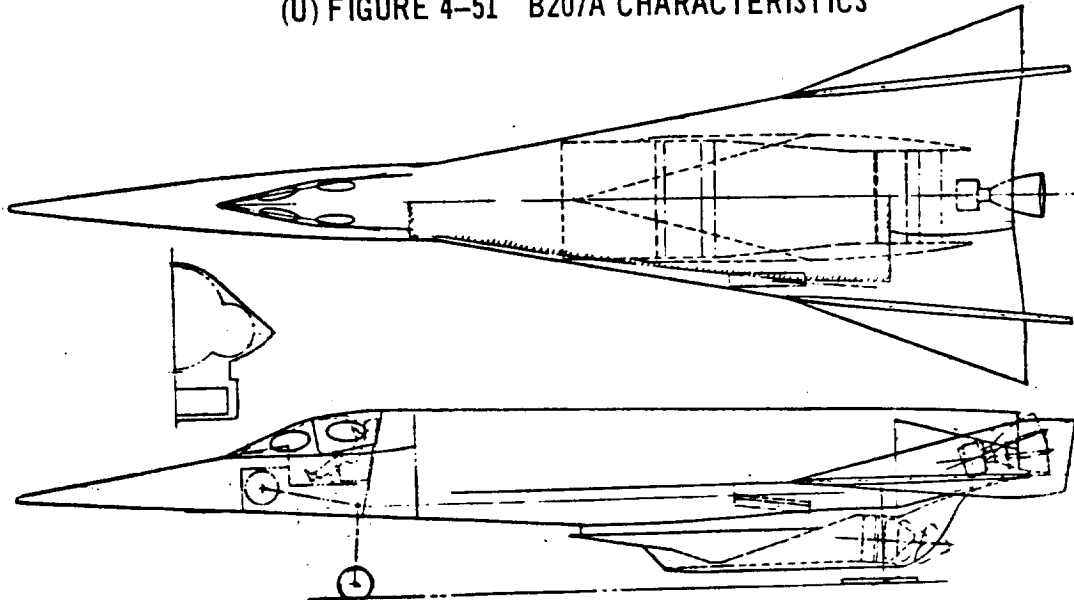
(U) FIGURE 4-49 PHASE II CLIMB PROFILES
Baseline Systems (Nominal)



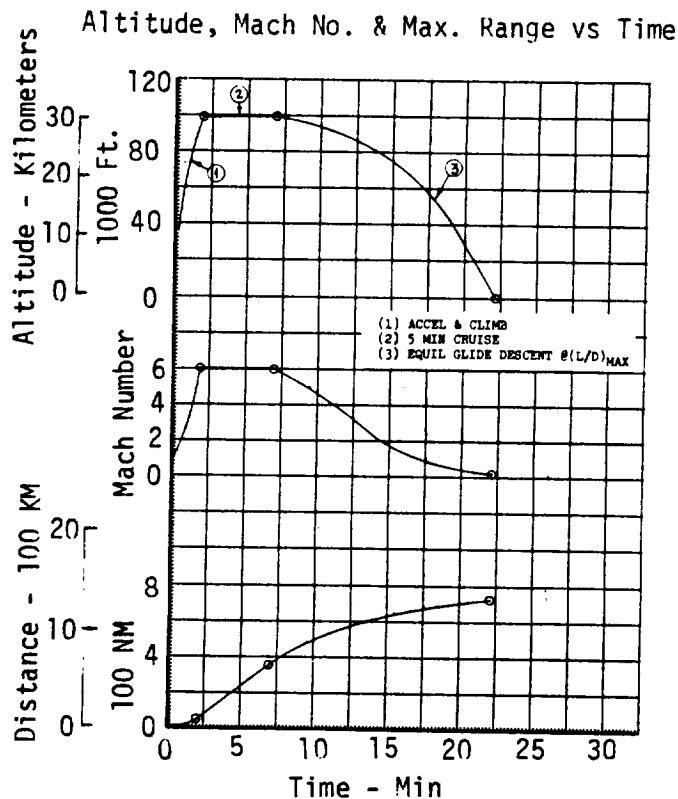
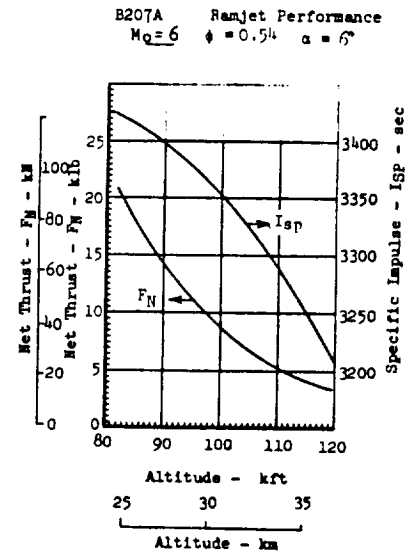
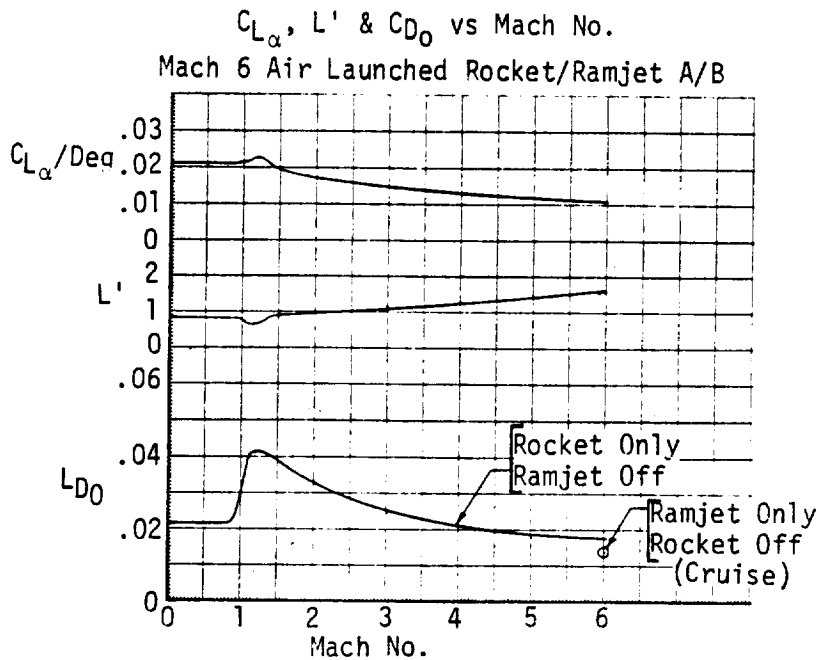
(U) FIGURE 4-50
PERFORMANCE COMPARISON BASELINE HYPERSONIC RESEARCH VEHICLES

Model	Design Mach No.	Launch Concept	Type Config	Propulsion System		Fuel	Sp F ² (m ²)	Volume Total Ft ³ (m ³)	Volume Fuel Ft ³ (m ³)	$\frac{2/3}{S_p}$	L/D _{Max} at Design Mach	Owe lb (kg)	TOGW lb (kg)	Propellant Wt		W _F : W _{TO}	W/S _p TO lb/ft ² (kg/m ²)	Remarks
				Accel and Climb	Cruise									Accel and Climb-lb (kg)	Cruise lb (kg)			
B207A	6	AIR	AB	RKT	RJ	LH ₂	426 (39.6)	1,700 (48.11)	775 (21.9)	0.335	3.19	22,860 (10,369)	38,430 (17,432)	14,920 (6,766)	650 (295)	0.405	90.3 (440.9)	Manned
B212A	6	HTO	WB	TJ/RJ	RJ	LH ₂	1,004 (93.27)	3,048 (86.26)	1,005 (28.44)	0.210	3.25	32,540 (14,760)	37,223 (16,884)	3,573 (1,621)	1,110 (503)	0.126	37.1 (181.1)	
B232A	12	AIR	AB	RKT	SJ	LH ₂	947 (87.98)	4,260 (120.6)	2,418 (68.43)	0.278	2.98	26,730 (12,125)	78,010 (35,385)	49,900 (22,634)	1,380 (626)	0.658	82.4 (402.3)	
B233A	12	AIR	AB	RKT	RKT	LH ₂	929 (86.30)	3,840 (108.7)	2,276 (64.41)	0.263	3.10	24,600 (11,158)	77,950 (35,358)	49,030 (22,240)	4,320 (1,960)	0.685	84.0 (410.1)	
B257F	12	HTO	BWB	TJ/CSJ	CSJ	LH ₂	2,860 (266.00)	17,400 (492)	7,560 (214)	0.234	3.69	82,600 (37,466)	117,830 (53,447)	32,000 (14,515)	3,230 (1,465)	0.299	41.2 (202.0)	
B260A	12	HTO/VTO	AB	RKT	RKT	LH ₂	1,405 (130.5)	6,990 (197.8)	4,655 (131.7)	0.260	3.14	37,900 (17,191)	147,250 (66,791)	102,800 (46,629)	6,550 (2,971)	0.742	105.0 (512.7)	Manned
B284A	12	AIR	AB	RKT	RKT	LH ₂	881 (81.8)	3,810 (107.8)	2,190 (61.98)	0.277	3.00	23,190 (10,519)	74,590 (33,833)	47,180 (21,400)	4,220 (1,914)	0.688	84.7 (413.5)	Unmanned

(U) FIGURE 4-51 B207A CHARACTERISTICS



<u>PARAMETERS</u>		<u>PHYSICAL CHARACTERISTICS</u>			
Mach No.	6	SIZE:	ft	(m)	
		Overall Length	58.3	(17.77)	
TVAC/TOGW	1.5	Total Span	20.9	(6.37)	
		Height	10.9	(3.32)	
Test Time	5 min	AREAS:	ft ²	(m ²)	
Payload	1000 lb (453.6 kg)	Planform	426	(39.58)	
Load Factor	5.0(Struct) 3.5 (TPS)	Wing	331	(30.75)	
Basing	Dual	Tips, Total	50	(4.65)	
		Vert.Tails, Total	81	(7.52)	
Therm.Prot.Sys.	Passive	VOLUME:	ft ³	(m ³)	
		Total	1700	(48.11)	
Tankage	Integral	Propellant	775	(21.93)	
Fuel Type	Subcooled LH ₂	PROPULSION:	ACCEL.	CRUISE	
		Engine	LR 129 RKT	Ramjet	
Wing Loading	90.3 lb/ft ² (440.88 kg/m ²)	Thrust (VAC) lb	57,645	-	
		(N)	(256,405)	-	
		Thrust(Mach 6 @ 1b	-	7,160	
		Cruise Alt) (N)	-	(31,848)	
LE Sweep	80°	Acapt ft ²	0	13.2	
		(m ²)	0	(1.23)	

(C) FIGURE 4-52
B207A CAPABILITIES

B207A WEIGHT

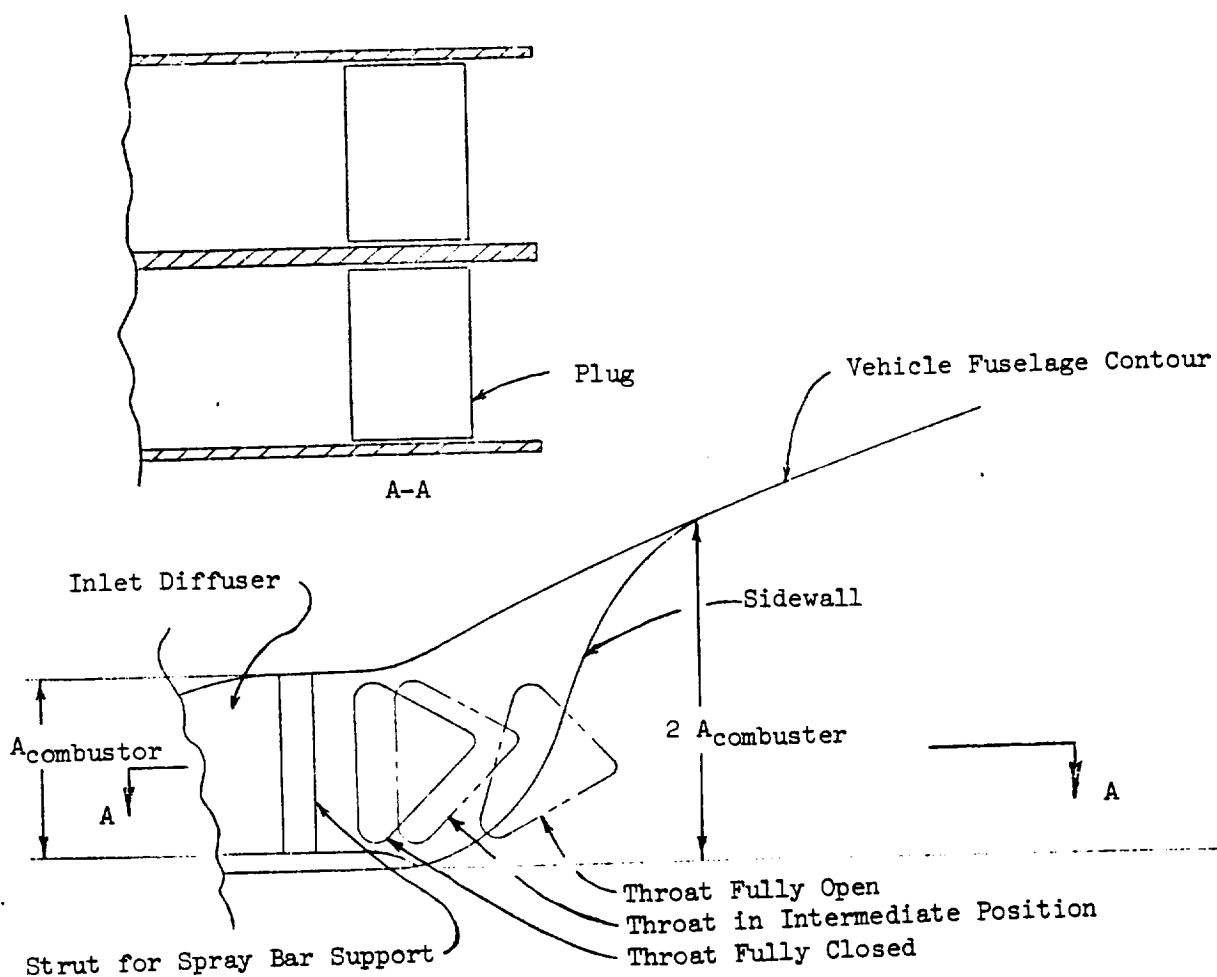
	lbm.	kg
STRUCTURE	8531	3869.6
EQUIPMENT	4448	2017.6
PROPULSION	9881	4481.9
PROPELLANT	15570	7062.4
OWE	22860	10369.1
TOGW	38430	17431.6

B207A

	COST (3 VEHICLES) (MILLIONS OF 1970 DOLLARS)
RDT&E	332
INVESTMENT	121
OPERATING	164
TOTAL	617

(U) The cruise engine is a two-dimensional, hydrogen-fueled ramjet based on the axisymmetric Marquardt MA 145-XAB design, Reference (15), and configured to the specific integration requirements of this vehicle. A translating plug nozzle, Figure 4-53, is incorporated in the engine to provide the required variation in nozzle throat area. The engine and nozzle are regeneratively cooled with hydrogen fuel. The ramjet capture area is 13.2 ft² (1.23 m²) and is sized at Mach 6.0 cruise to provide thrust equal to vehicle drag while operating at minimum SFC (maximum specific impulse). The fuel flow at this condition is compatible with engine and nozzle cooling requirements. The ramjet is ignited at Mach 6.0 for cruise. The performance for these operating conditions is presented in Figure 4-52.

(U) FIGURE 4-53 TWO-DIMENSIONAL RAMJET FEATURES



4.3.2 (U) TURBOJET - RAMJET CONFIGURATION (CLASS B212) - Design B212A, shown in Figure 4-54, is the baseline Mach 6 configuration representing the wing-body, horizontal takeoff, manned vehicle approach. For acceleration to Mach 3, this aircraft is powered by two wing-mounted turbojet engines and is powered by a two-dimensional hydrogen-fueled ramjet engine from Mach 1 to Mach 6 cruise. The drag rise associated with the turbojet engine-out condition above Mach 3 is evident in Figure 4-55. Additional performance, weight, and cost data for this design are also shown in Figure 4-55. The maximum range and total flight time associated with this vehicle are 900 nmi (1670 km) and 30 min respectively.

(U) The inlet configuration for the turbojet engines is a mixed compression, axisymmetric design. External compression is achieved on a single cone. Throat geometry is varied by translating the centerbody.

(C) The General Electric F100-GE-100 engine is a dual rotor, mixed flow afterburning turbofan having a 0.89 bypass ratio and an overall pressure ratio of 23.8. The GE-TFDC-10C translating flap convergent-divergent nozzle is incorporated and the engines have been converted to use hydrogen fuel. The engines are operated at maximum afterburner power setting for acceleration to Mach 3.0 after which they are shut down. The performance characteristics for these engines are presented in Figure 4-55.

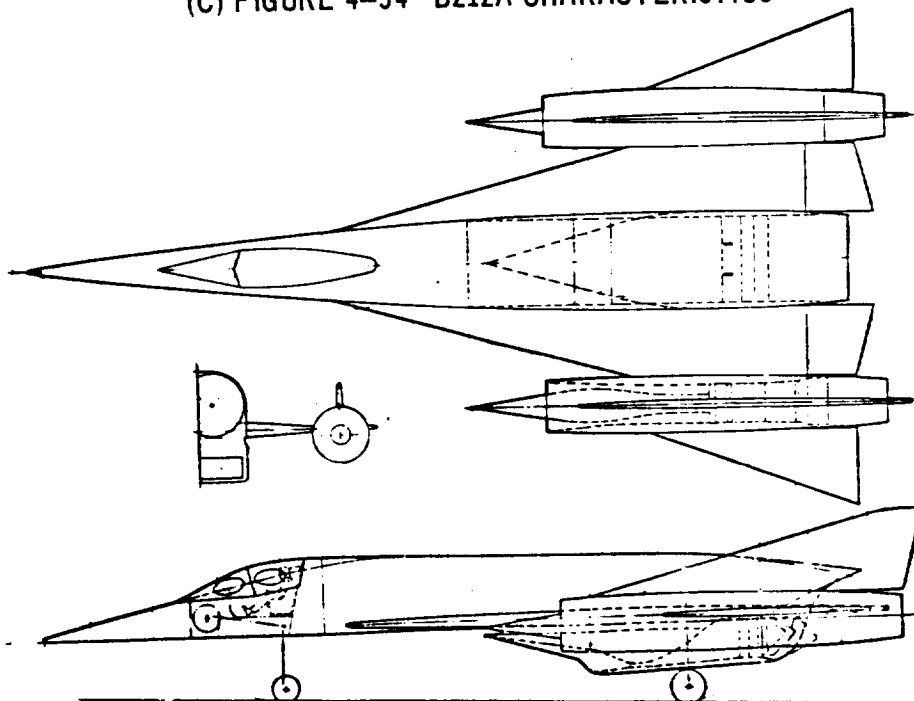
(U) The inlet for the cruise engine is a two-dimensional, mixed compression configuration. The inlet has a single, overhead external compression ramp located in the vehicle compression field. Throat area variation is achieved by hinged ramps in the inlet and diffuser. Both the inlet and engine are integrated with the lower fuselage moldline.

(U) The cruise engine is a two-dimensional, hydrogen-fueled ramjet based on the axisymmetric Marquardt MA-145-XAB design and configured to the specific integration requirements of this vehicle, Figure 4-53. A translating plug nozzle was incorporated in the engine to provide the required variation in nozzle throat area. The engine and nozzle are regeneratively cooled with hydrogen fuel. The ramjet capture area of 21.45 ft² (1.99 m²) was sized at Mach 6.0 cruise to provide thrust equal to vehicle drag while operating at minimum SFC (maximum specific impulse). It was then confirmed that this sized engine produced satisfactory acceleration. The fuel flow at this condition is compatible with engine and nozzle cooling requirements.

4.3.3 (U) ROCKET-SCRAMJET CONFIGURATION (CLASS B232) - Design B232A, shown in Figure 4-56 is the baseline Mach 12 configuration representing an all-body, air launched, manned vehicle approach. A rocket engine is employed for acceleration and a scramjet for cruise. The resulting performance, weight, and cost data are shown in Figure 4-57. At Mach 12 the vehicle drag is less for cruise than in the acceleration mode because base drag is less with SJ on and the rocket off, than with rocket on and SJ off. The maximum range and flight time are 3000 nmi (5560 km) and 40 min respectively.

(C) The scaled version of the PWA LR-129 rocket engine is 84.5 in (214 cm) long, and has a conventional bell nozzle with an expansion ratio of 75.0 and an exit diameter of 43.5 in (110 cm). The engine uses LO₂ and LH₂ propellants, incorporates both a pre-burner and main burner in the basic cycle, and is

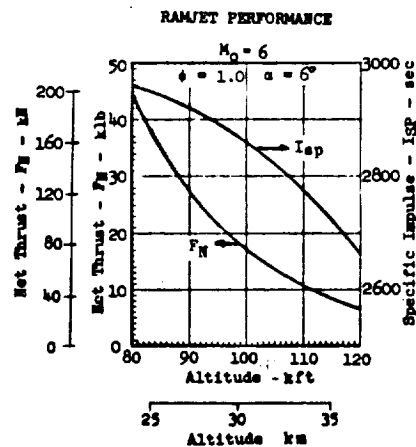
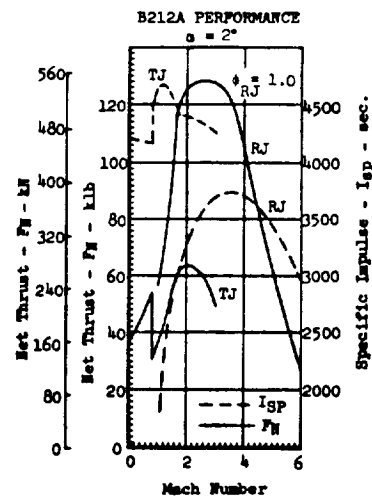
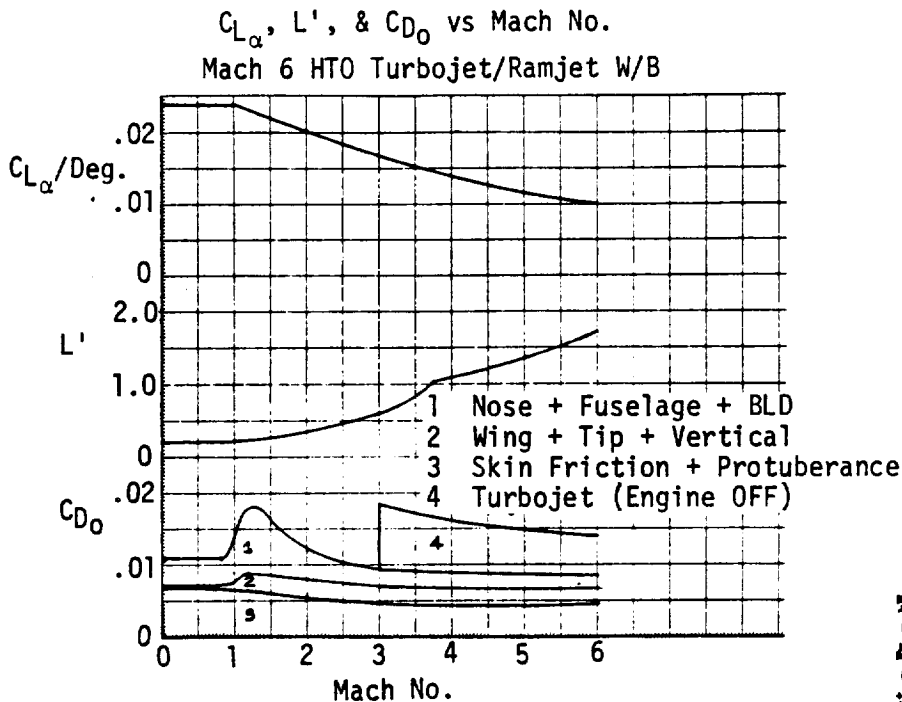
(C) FIGURE 4-54 B212A CHARACTERISTICS



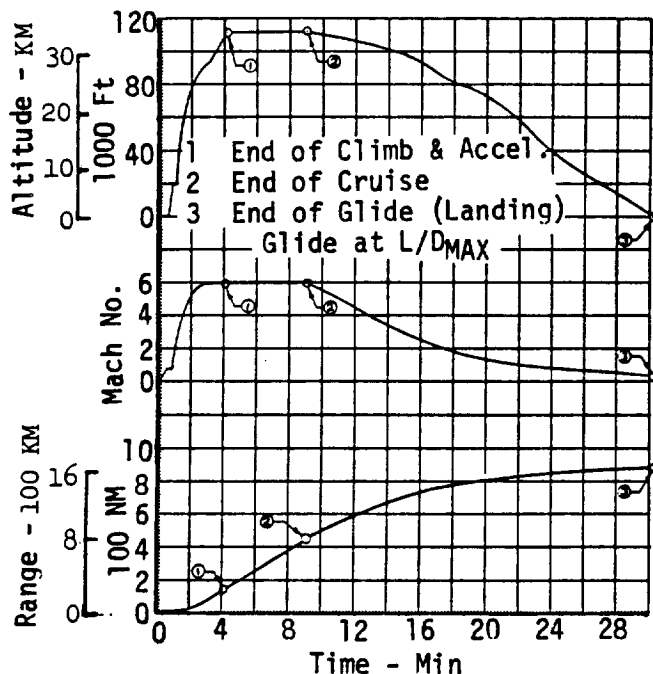
<u>PARAMETERS</u>		<u>PHYSICAL CHARACTERISTICS</u>			
Mach No.	6	SIZE:	ft	(m)	
		Overall Length	67.6	(20.60)	
Thrust/Weight*	1.3	Total Span	38.7	(11.79)	
		Height	15.5	(4.72)	
Test Time	5 min	AREAS:	ft ²	(m ²)	
Payload	1000 lb (453.6 kg)	Planform	1004	(93.27)	
Load Factor	5.0 (Struct)	Wing	641	(59.54)	
	3.5 (TPS)	Tips, Total	98	(9.10)	
Basing	Dual	Vert Tails, Total	172	(15.97)	
Thermal Prot. Sys	Passive	VOLUMES:	ft ³	(m ³)	
		Total	3048	(86.26)	
Tankage	Integral	Propellant	1005	(28.44)	
Fuel Type	Subcooled LH ₂	PROPULSION:	ACCEL.	CRUISE	
		Engine	(2) F100 GE-100	Ramjet	
Wing Loading	37.1 lb/ft ² (181.14 kg/m ²)	Thrust (SLS, Uninst) lb	23,380 (ea)	-	
			(N) (103,994) (ea)	-	
		Thrust (Mach 6 @ lb	-	9,750	
		Cruise Alt) (N)	-	(43,368)	
LE Sweep	75°	A _{capt} ft ²	12.0 (ea)	21.45	
		(m ²)	(1.11)	(1.99)	

*Uninstalled SLS thrust/takeoff gross weight.

(C) FIGURE 4-55
B212A CAPABILITIES



Altitude, Mach No. & Max Range vs Time
Configuration B212A



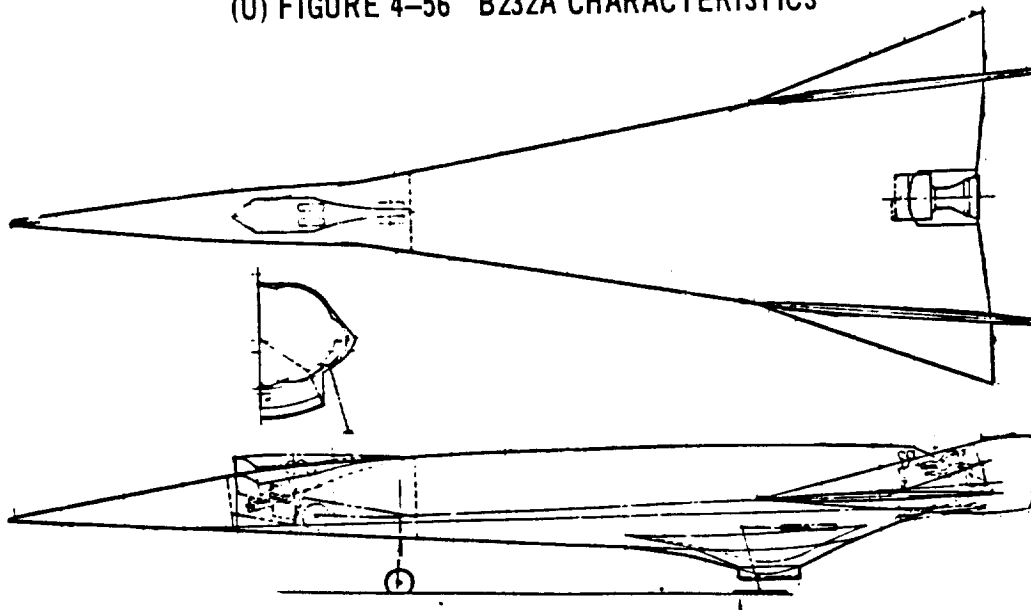
B212A WEIGHT

	lbm.	kg
STRUCTURE	14747	6689.1
EQUIPMENT	4929	2235.3
PROPULSION	12864	5835.0
PROPELLANT	4683	2124.2
OWE	32540	14759.9
TOGW	37223	16884.1

COST (3 VEHICLES)
(MILLIONS OF 1970 DOLLARS)

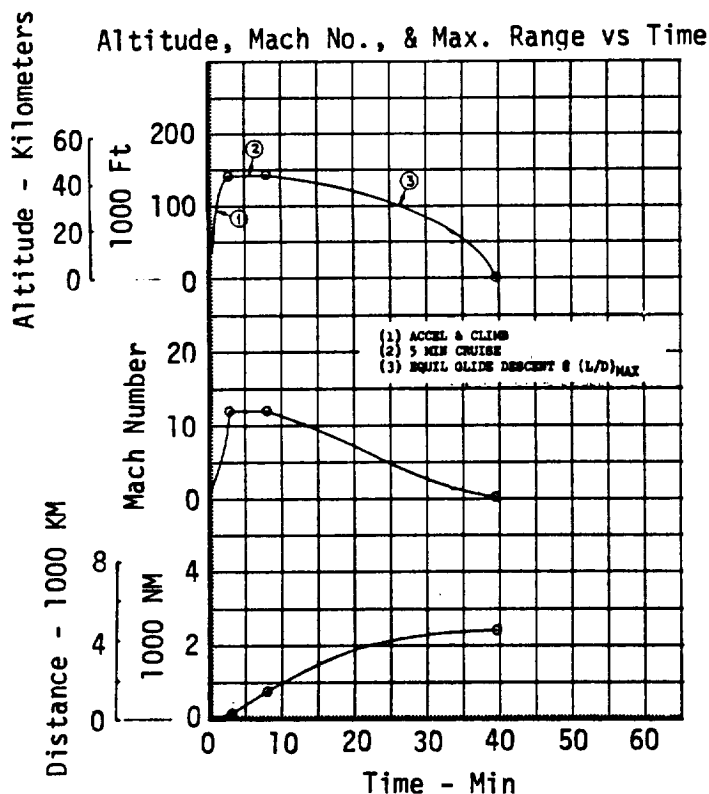
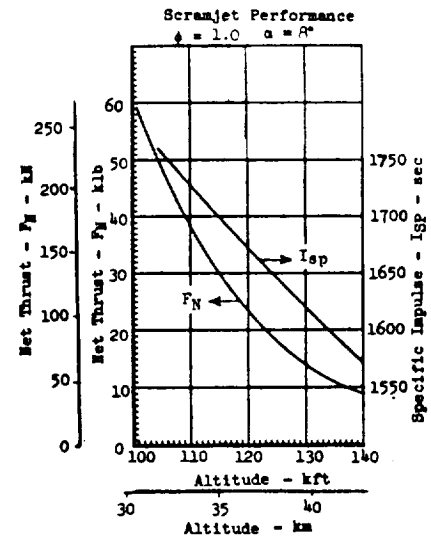
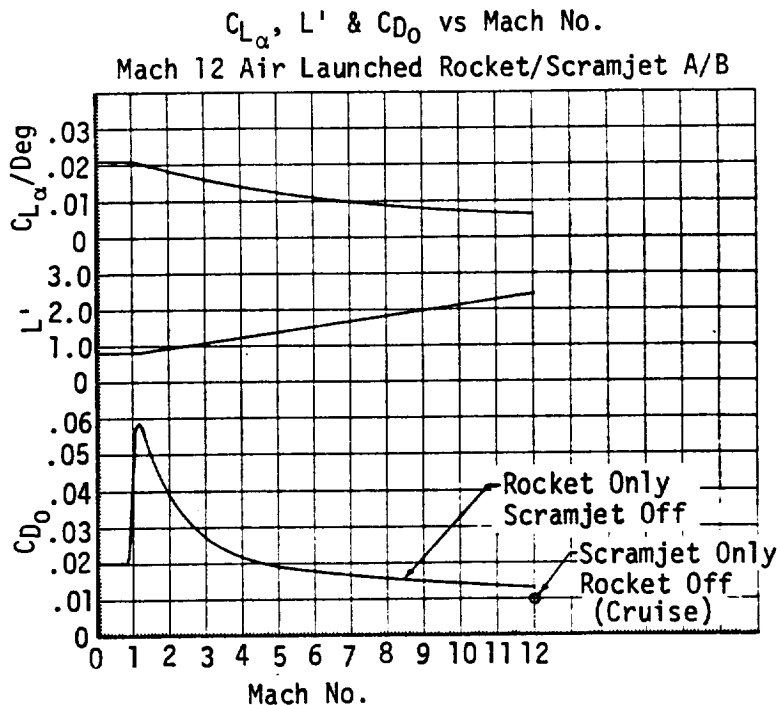
RDT&E	304
INVESTMENT	149
OPERATING	212
TOTAL	665

(U) FIGURE 4-56 B232A CHARACTERISTICS



<u>PARAMETERS</u>		<u>PHYSICAL CHARACTERISTICS</u>			
Mach No.	12	SIZE:	ft	(m)	
TVAC/TOGW	1.5	Overall Length	75.9	(23.13)	
Test Time	5 min	Total Span	31.1	(9.48)	
Payload	1000 lb (453.6 kg)	Height	12.9	(3.93)	
Load Factor	5.0 (Struct) 3.5 (TIPS)	AREAS:	ft ²	(m ²)	
Basing	Dual	Planform	947	(87.98)	
Thermal Prot.Sys	Passive	Wing	794	(73.76)	
Tankage	Integral	Tips, Total	107	(9.94)	
Fuel Type	Subcooled LH ₂	Vert.Tails, Total	170	(15.79)	
Wing Loading	82.4 psf (402.31 kg/m ²)	VOLUMES:	ft ³	(m ³)	
LE Sweep	80°	Total	4260	(120.56)	
		Propellant	2418	(68.43)	
		PROPULSION:	ACCEL	CRUISE	
		Engine	LR 129 RKT	SJ	
		Thrust (VAC) lb	117,015	-	
		(N)	(520,483)	-	
		Thrust(Mach 12 @ lb	-	7,300	
		Cruise Alt) (N)	-	(32,470)	
		Acapt ft ²	0	42.6	
		(m ²)	0	(3.96)	

(C) FIGURE 4-57 B232A CAPABILITIES



B232A WEIGHT

	lbm.	kg
STRUCTURE	15859	7193.5
EQUIPMENT	4805	2179.5
PROPULSION	6066	2751.5
PROPELLANT	51280	23260.2
OWE	26730	12124.5
TOGW	78010	35334.7

B232A

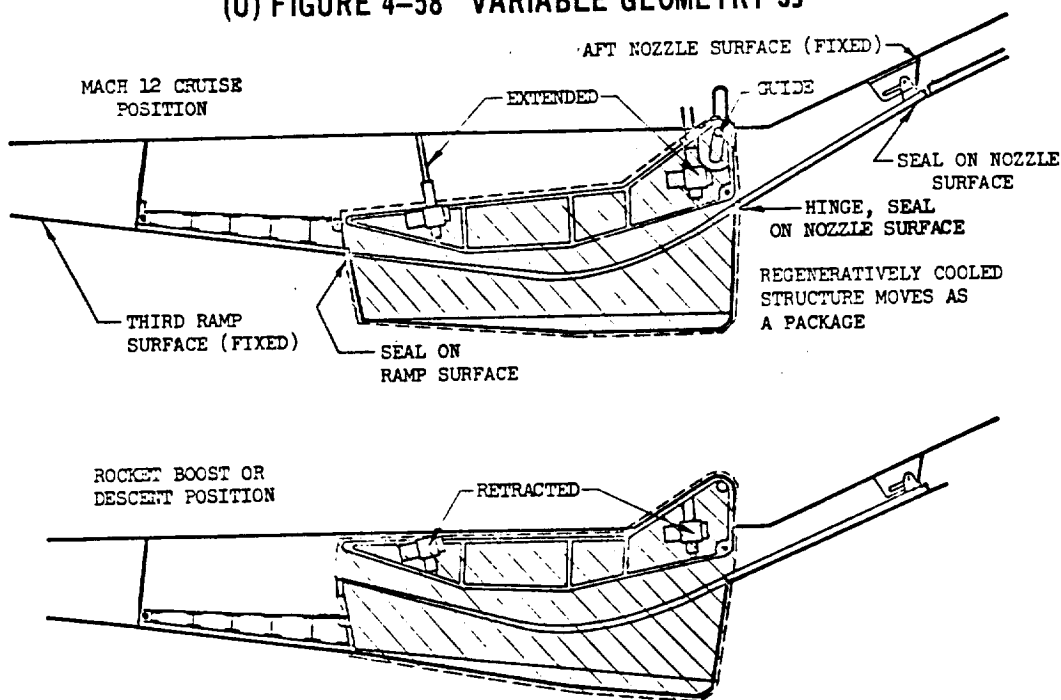
COST (3 VEHICLES)
(MILLIONS OF 1970 DOLLARS)

RDT&E	555
INVESTMENT	139
OPERATING	184
TOTAL	878

regeneratively and transpiration cooled with hydrogen fuel. At full power, the rocket operates at 3000 psi (2060 N/cm²) chamber pressure at a mixture ratio of 6.0. The engine operates at full power from launch to the start of cruise and is shut down during cruise. The engine is scaled to produce 117,000 lb (519,000 N) vacuum thrust, which results in a vehicle TVAC/TOGW of 1.5. The vacuum propellant specific impulse is 451 sec.

(U) During the cruise, the B232 aircraft uses a scramjet engine. The scramjet inlet, Figure 4-58, is based upon the analytical development described in Reference (1). The fixed geometry inlet employs an initial 4 degree aircraft forebody half angle compatible with an all-body design, followed by two successive two-dimensional external compression ramps of 4 and 5 degrees respectively, to achieve an external geometric contraction ratio of about 5.6:1. An overall geometric contraction ratio of 11.3 is achieved by means of a single shock turning the internal flow through a 13° angle. The inlet is designed to operate with shock-on-lip at 9° angle of attack at Mach 12. At that condition, the increased capture area achieved with angle of attack results in an overall contraction ratio equal to about 22:1, a combustor inlet Mach number equal to 3.9, and near optimum cruise performance.

(U) FIGURE 4-58 VARIABLE GEOMETRY SJ



(U) The scramjet engine consists of a series of regeneratively cooled modules, Figure 4-58. Each module has an internal inlet contraction section, a burner, and an internal nozzle expansion section. The aft surfaces of the lower fuselage are also used for nozzle expansion area. Each internal inlet section has a contraction ratio of 2:1. Each burner section is a divergent duct 30 in (76 cm) in length. The burners have an expansion ratio of 1.20 (outlet area to inlet area) with fuel injectors located at the entrance to

each burner. The scramjet burners are sized for Mach 12 cruise and operate stoichiometrically during cruise. For rocket-powered acceleration to Mach 12 and for unpowered descent and deceleration, the engine module (including the internal inlet contraction section and the internal nozzle expansion section) is retracted into the vehicle lower fuselage.

(U) The performance characteristics used for this engine were taken from cycle analyses described in Reference (1) and these data are presented in Figure 4-57. Expanding the scramjet exhaust gases onto the vehicle aft surfaces resulted in an incremental lift which adds a vehicle $\Delta L/D$ of 1.5 at Mach 12.

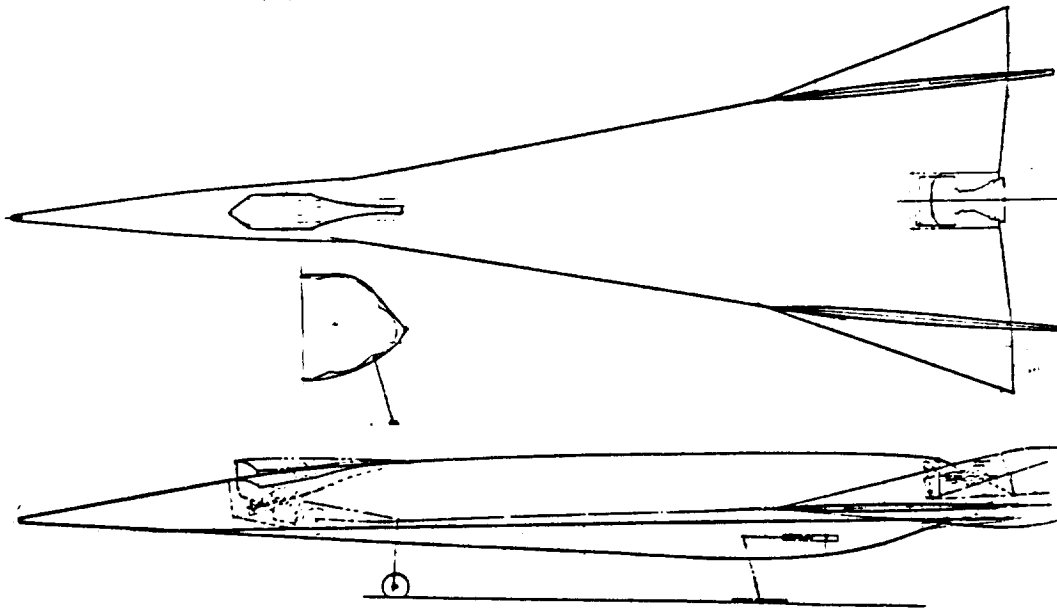
4.3.4 (U) ROCKET CONFIGURATION (CLASS B233) - Vehicle B233A shown in Figure 4-59, is the baseline Mach 12, manned, air launched vehicle of an all-body shape. Acceleration, climb, and cruise flight are powered by a rocket engine providing a maximum range and total flight time of 2700 nmi (5000 km) and 42 min respectively. Performance, cost, and weight data for this configuration are shown in Figure 4-60. Cruise altitude for this vehicle is near 150,000 ft (45,750 m) and incorporates a slight climb due to decreasing propellant weight.

(C) The propulsion system for the B233A consists of a rocket engine used for acceleration and Mach 12 cruise. A scaled Pratt and Whitney Aircraft LR-129 high chamber pressure rocket is the selected system engine. The scaled version of the PWA LR-129 rocket engine is 84.5 in (214.4 cm) long, and has a conventional bell nozzle with an expansion ratio of 75 and an exit diameter of 43.5 in (110.3 cm). The engine uses LO_2 and LH_2 propellants, incorporates both a pre-burner and a main burner in the basic cycle, and is regeneratively and transpiration cooled with hydrogen fuel. At full power, the rocket operates at 3000 psi (2060 N/cm²) chamber pressure at a mixture ratio of 6.0. The engine is operated at full power from launch to the start of cruise; it is throttled back to approximately 6% of full power during cruise to maintain stable flight. The engine is scaled to produce 116,900 lb (519,000 N) vacuum thrust which results in a vehicle $TVAC/TOGW$ of 1.5. The vacuum propellant specific impulse is 451 sec. The performance of the engine, when throttled for cruise operation, is presented in Figure 4-60.

4.3.5 (U) TURBOFAN - CONVERTIBLE SCRAMJET CONFIGURATION (CLASS B257) - The B257F design, shown in Figure 4-61, represents the baseline approach selected for a Mach 12, manned, blended wing-body concept capable of horizontal takeoff. The vehicle is powered during acceleration and climb by four fuselage mounted turbofan engines to Mach 3, and by a convertible scramjet engine from Mach 1 to Mach 12. The scramjet also powers the cruise portion of flight. Performance, weight, and cost data developed for this configuration are shown in Figure 4-62. Mach 12 cruise is at an altitude in excess of 140,000 ft (42,700 m) with a resultant maximum range and total flight time of 3100 nmi (5740 km) and 44 min.

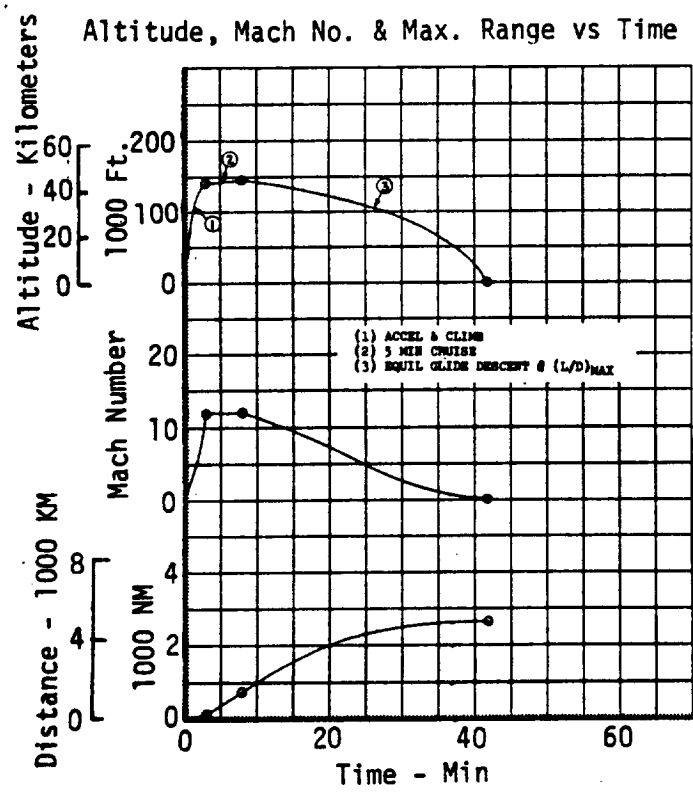
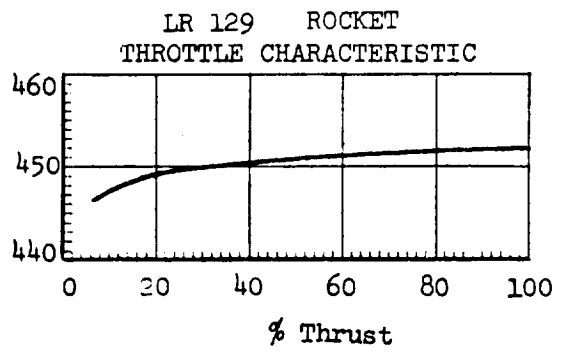
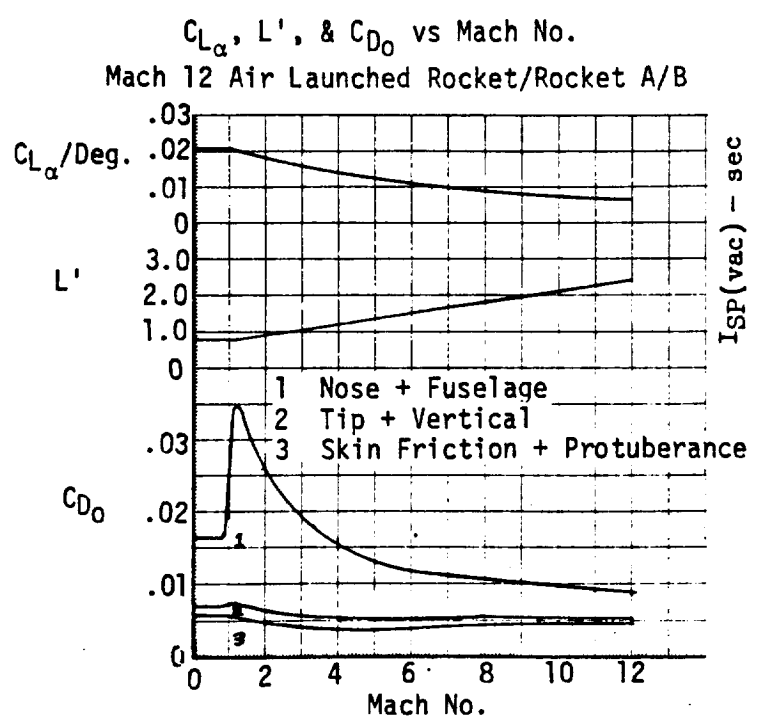
(C) The General Electric F100-GE-100 engine employed for acceleration to Mach 3 is a dual rotor, mixed flow afterburning turbofan having a 0.89 bypass ratio and an overall pressure ratio of 23.8. The GE TFCD-10C translating flap convergent-divergent nozzle is incorporated and the engines have been converted to use hydrogen fuel. The maximum internal temperature limits used in the JP-fueled engine design were maintained. The engines operate at maximum afterburner power setting for acceleration to Mach 3.0 above which they are shutdown. The performance used for these engines is presented in Figure 4-62.

(U) FIGURE 4-59 B233A CHARACTERISTICS



PARAMETERS		PHYSICAL CHARACTERISTICS			
Mach No.	12	SIZE:	ft	(m)	
		Overall Length	83.3	(25.39)	
TVAC/TOGW	1.5	Total Span	30.7	(9.36)	
		Height	12.8	(3.90)	
Test Time	5 min	AREAS:	ft ²	(m ²)	
Payload	1000 lb (453.6 kg)	Planform	929	(86.30)	
		Wing	778	(72.28)	
Load Factor	5.0 (Struct)	Tips, Total	105	(9.75)	
	3.5 (TPS)	Vert Tails, Total	170	(15.79)	
Basing	Dual				
Thermal Prot.Sys	Passive	VOLUMES:	ft ³	(m ³)	
		Total	3840	(108.67)	
Tankage	Integral	Propellant	2276	(64.41)	
Fuel Type	Subcooled LH ₂	PROPULSION:	ACCEL	CRUISE	
		Engine	(1)LR 129 RKT	(1)LR129 RKT	
Wing Loading	84.0 lb/ft ² (410.12 kg/m ²)	Thrust (VAC) lb (N)	116,925 (520,082)	(throttled)	
LE Sweep	80°			-	

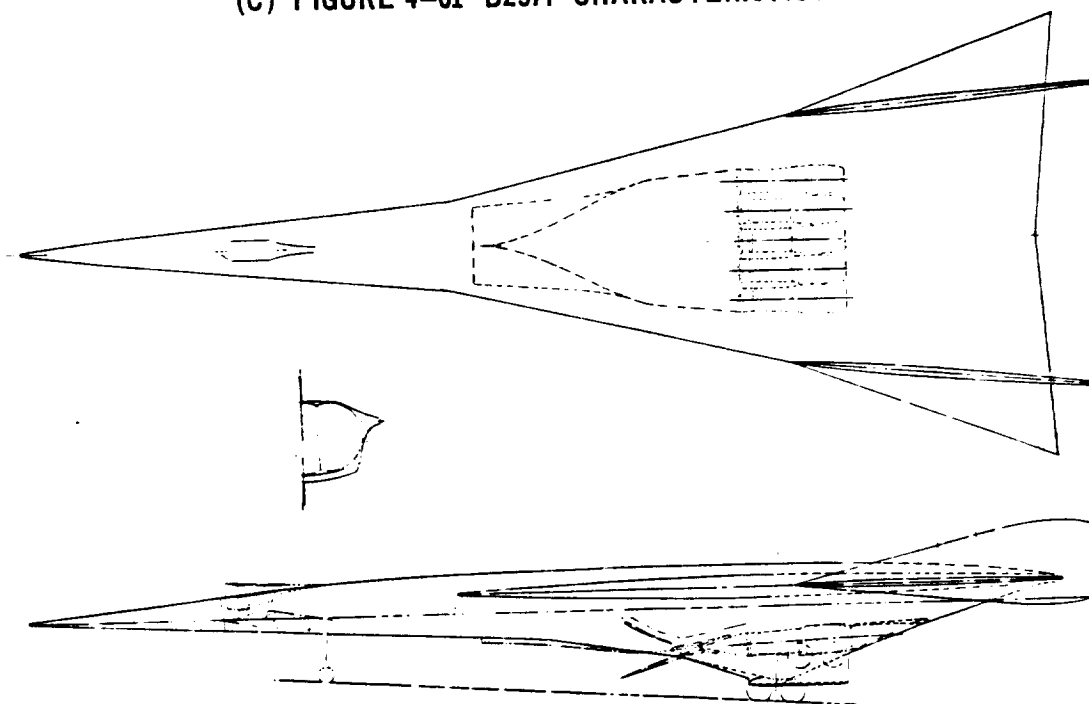
(C) FIGURE 4-60
B233A CAPABILITIES



B233A WEIGHT		
	lbm.	kg
STRUCTURE	16771	7607.2
EQUIPMENT	4827	2189.5
PROPULSION	3002	1361.7
PROPELLANT	53350	24192.2
OWE	24600	11153.4
TOGW	77950	35357.5

B233A COST (3 VEHICLES) (MILLIONS OF 1970 DOLLARS)	
RDT&E	243
INVESTMENT	124
OPERATING	178
TOTAL	544

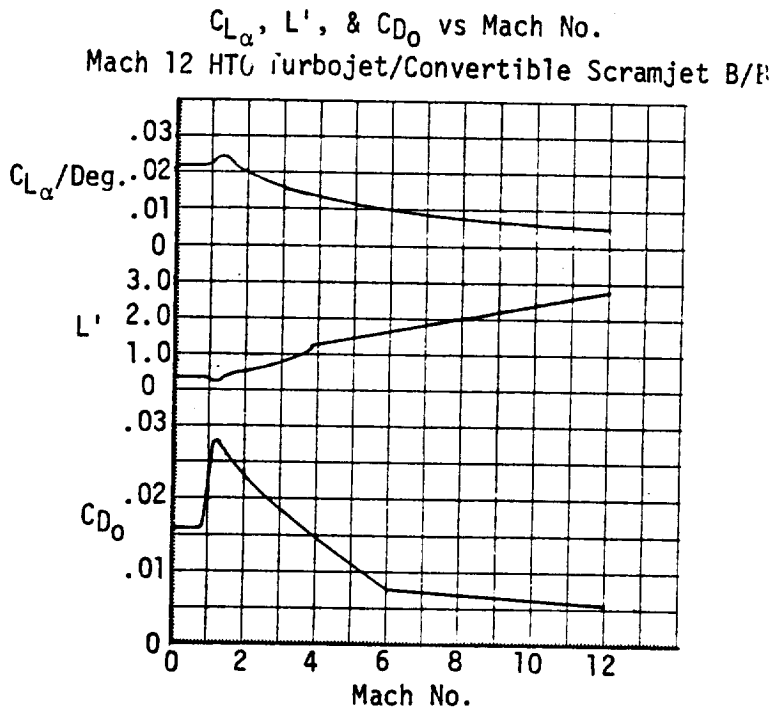
(C) FIGURE 4-61 B257F CHARACTERISTICS



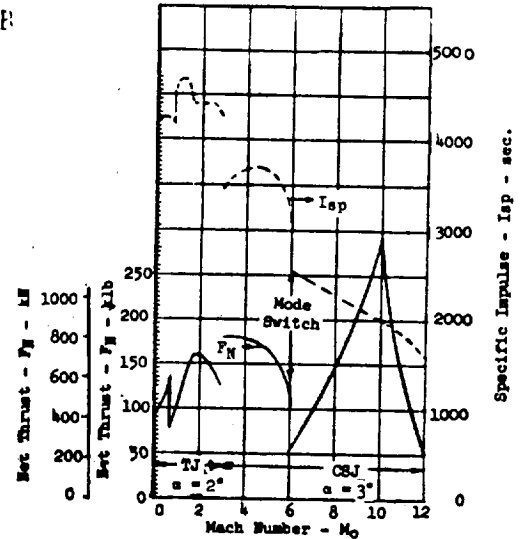
<u>PARAMETERS</u>		<u>PHYSICAL CHARACTERISTICS</u>			
Mach No.	12	SIZE:	ft	(m)	
		Overall Length	143.17	(43.64)	
Thrust/Weight*	1.1	Total Span	57.83	(17.62)	
		Height	25.77	(7.85)	
Test Time	5 min	AREAS:	ft ²	(m ²)	
Payload	1000 lb (453.6 kg)	Planform	2860	(265.69)	
Load Factor	5.0 (Struct) 3.5 (TPS)	Wing	2320	(215.53)	
Basing	Dual	Tips, Total	335	(31.12)	
		Vert. Tails, Total	550	(51.10)	
Thermal Prot. Sys.	Passive	VOLUMES:	ft ³	(m ³)	
		Total	17,400	(492.42)	
Tankage	Integral	Propellant	7,560	(213.95)	
Fuel Type	Subcooled LH ₂	PROPULSION:	ACCEL	CRUISE	
Wing Loading	41.2 lb/ft ² (201.15 kg/m ²)	Engine	(5)F-100-GE-100	CSJ	
		Thrust (SLS, Uninst) lb	23,380 (ea)	-	
		(N)	(103,994) (ea)	-	
		Thrust (Mach 12@	lb	16,280	
		Cruise Alt) (N)	-	(72,413)	
LE Sweep	77°	A _{capt} ft ²	11.95 (ea)	128.8	
		(m ²)	(1.11)	(11.97)	

* Uninstalled SLS thrust/takeoff gross weight.

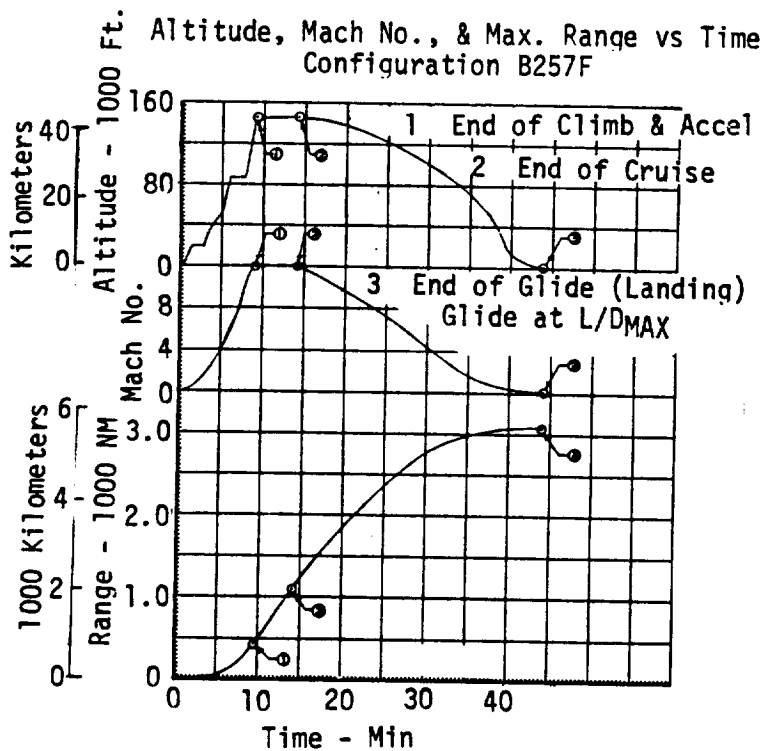
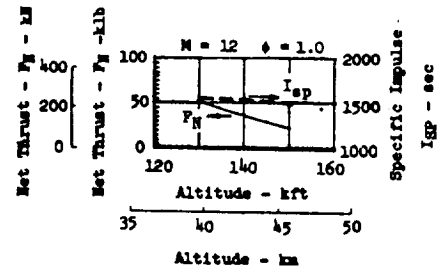
(C) FIGURE 4-62
B257F CAPABILITIES



Convertible Scramjet
Climb Performance



B257F Convertible Scramjet
Cruise Performance



B257F WEIGHT

	lbm.	kg
STRUCTURE	41350	18756.0
EQUIPMENT	6710	3043.6
PROPULSION	34540	15667.1
PROPELLANT	35230	15980.1
OWE	82600	37466.7
TOGW	117830	53446.8

B257F	
COST (3 VEHICLES)	
(MILLIONS OF 1970 DOLLARS)	
RDT&E	602
INVESTMENT	281
OPERATING	383
TOTAL	1266

(U) The inlets for the turbofan are fuselage mounted and located in the vehicle compression field. The fuselage forebody acts as a single external compression ramp. The capture area is varied by means of a movable cowl which retracts into the vehicle forebody when the turbojets are inoperative. The inlet is essentially an internal compression design. Throat area variation is provided by hinged upper and lower ramps in the inlet and diffuser.

(U) A convertible scramjet engine is used to accelerate this vehicle from Mach 3 to Mach 12 and for Mach 12 cruise operation. In the range Mach 1-3 this engine is operated to reduce base drag. The convertible scramjet inlet is based upon the analytical development of Reference (1). The inlet has a variable capture area and employs an initial 3° aircraft forebody half angle compatible with a wing-body design, followed by two successive fixed two dimensional external compression ramps of 6° each. The engine consists of a series of regeneratively cooled modules, Figure 4-63. Each module has an internal inlet contraction section, a burner, and an internal nozzle expansion section. The aft surfaces of the lower fuselage are also used for nozzle expansion area. Each internal inlet section has a contraction ratio of 2:1. Each burner section is a divergent duct 30 in (76 cm) in length. The burners have an expansion ratio of 1.20 with fuel injectors located at five axial stations along the burner. Staged fuel injectors are incorporated for efficient operation from Mach 1.0 to 12.0.

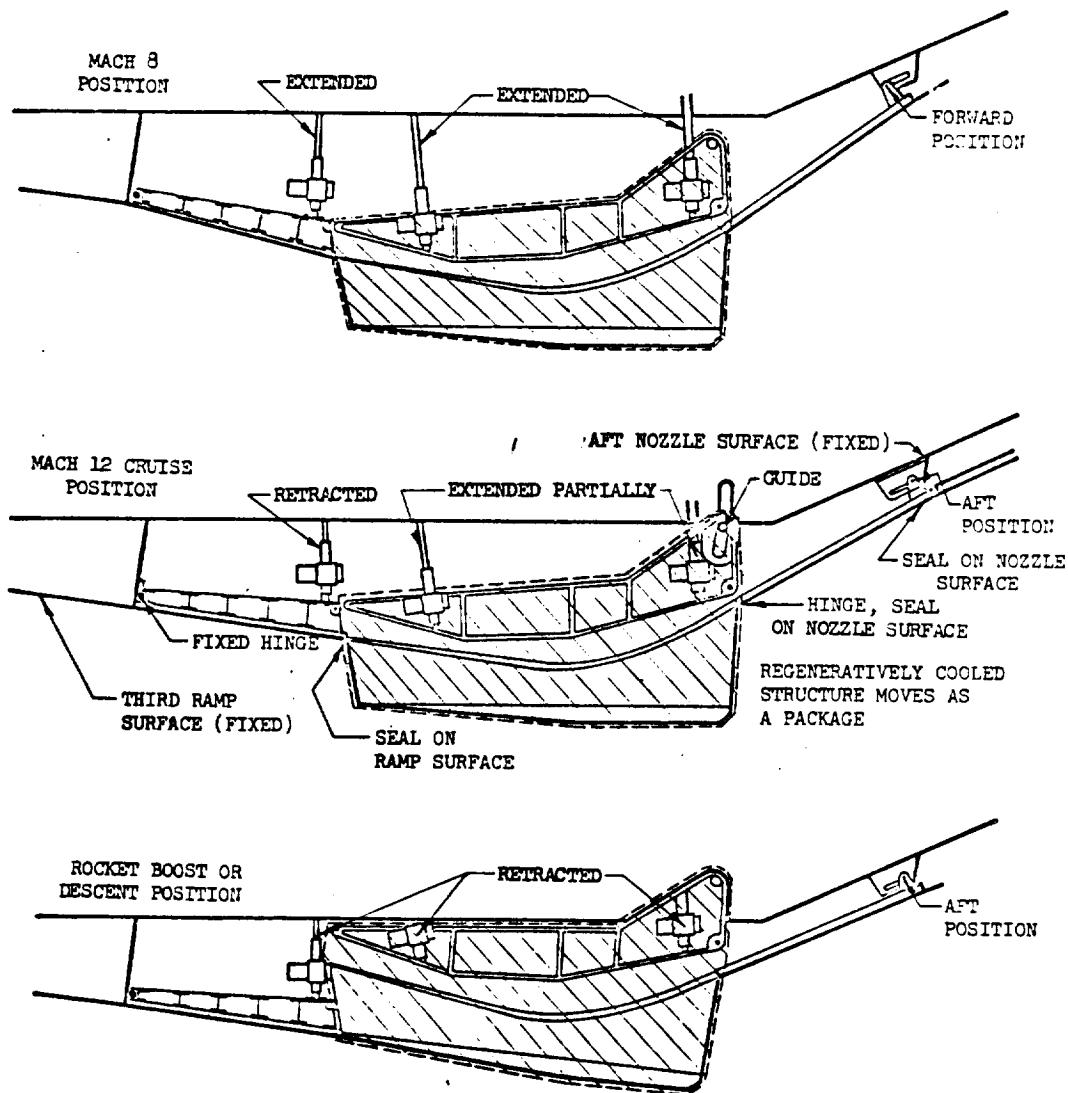
(U) The capture area is varied through use of hinged external ramps in conjunction with translating engine module (including the internal inlet contraction and the internal nozzle expansion sections), illustrated in Figure 4-62. This varies the inlet turning on the aft portion of the third ramp from 5° at the Mach 8 operating condition to 0° at Mach 12, Figures 4-63 and 4-64. The geometric contraction ratio of the inlet varies from about 11.9 at Mach 8 to about 9.1 at Mach 12, including 2:1 internal contraction. The overall contraction ratio and ramp angle schedules are illustrated in Figure 4-64.

(U) The engine operates in subsonic combustion mode from Mach 1 to 6, and in supersonic combustion mode from Mach 6 to 12, including Mach 12 cruise. The engine module is retracted into the vehicle lower fuselage during acceleration to Mach 1, and during unpowered descent and deceleration.

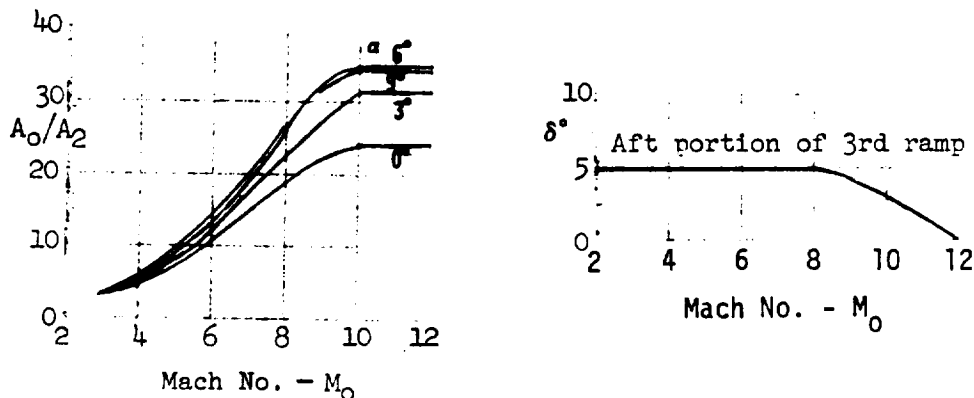
(U) The engine burners were sized for Mach 12.0 cruise. They operate at the maximum fuel flow below the thermal choke limit over the Mach range from 1 to 8, (where this limit is below the stoichiometric fuel flow) and operate stoichiometrically from Mach 8 to 12. The subsonic combustion mode performance characteristics were based on the data presented in Reference (16). The supersonic combustion mode performance characteristics used for this engine were taken from the cycle analyses of Reference (1) and are presented in Figure 4-62.

4.3.6 (U) ROCKET CONFIGURATION (CLASS B260) - Design B260A, Figure 4-65, is a configuration representing a Mach 12, manned, all-body vehicle that can be configured for either vertical or horizontal takeoff. The powered portions of flight rely on a rocket engine. Performance, cost, and weight data for this vehicle are presented in Figure 4-66. Cruise altitudes are near 150,000 ft (45,750 m). The maximum range and total flight time resulting are 2700 nmi (5000 km) and 43 min respectively.

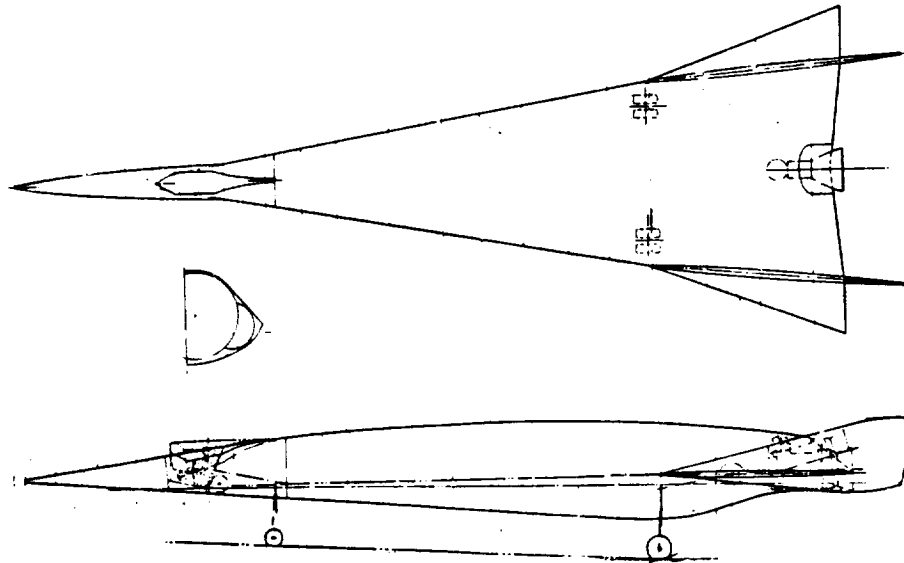
(U) FIGURE 4-63 VARIABLE GEOMETRY CSJ



(C) FIGURE 4-64 CSJ INLET CONTRACTION RATIO, MOVABLE RAMP ANGLE, AND BURNER MACH NO.



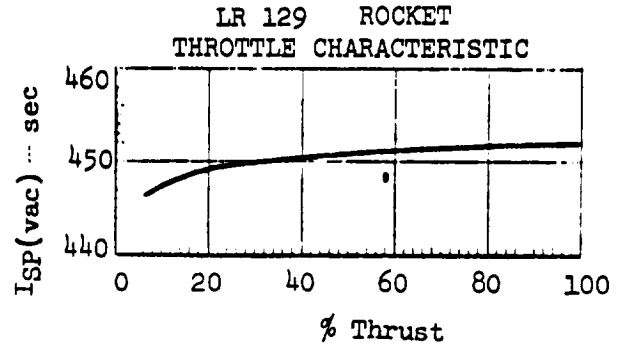
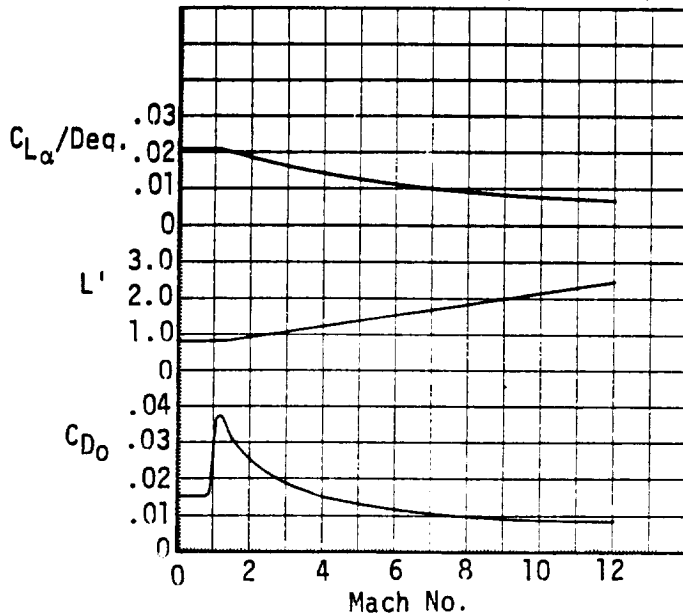
(U) FIGURE 4-65 B260A CHARACTERISTICS



<u>PARAMETERS</u>		<u>PHYSICAL CHARACTERISTICS</u>			
Mach No.	12	SIZE:	ft	(m)	
TVAC/TOGW	1.5	Overall Length	103.7	(31.61)	
Test Time	5 min	Total Span	37.7	(11.49)	
Payload	1000 lb (453.6 kg)	Height	16.1	(4.91)	
Load Factor	5.0 (Struct) 3.5 (TPS)	AREAS:	ft ²	(m ²)	
Basing	Dual	Planform	1405	(130.52)	
Thermal Prot.Sys.	Passive	Wing	1215	(112.87)	
Tankage	Integral	Tips, Total	144	(13.38)	
Fuel Type	Subcooled LH ₂	Vert.Tails, Total	286	(26.57)	
Wing Loading	105.0 lb/ft ² (512.65 kg/m ²)	VOLUMES:	ft ³	(m ³)	
LE Sweep	80°	Total	6990	(197.82)	
		PROPELLANT	4655	(131.70)	
		PROPULSION:	ACCEL	CRUISE	
		Engine	(1) LR-129 RKT	(1) LR-129 RKT	
		Thrust (VAC) lb (N)	220,875 (982,453)	-(Throttled)	

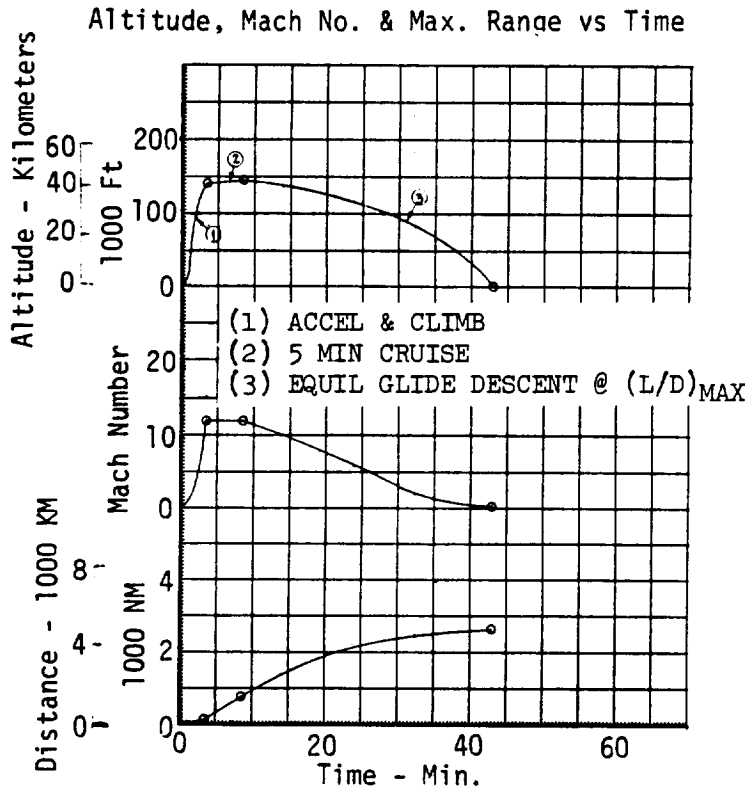
(C) FIGURE 4-66
B260A CAPABILITIES

$C_{L\alpha}$, L' , & C_{D0} vs Mach No.
Mach 12 HTO or VTO Rocket/Rocket A/B



B260A WEIGHT

	lbm.	kg
STRUCTURE	27777	12599.4
EQUIPMENT	5335	2419.9
PROPULSION	4788	2171.8
PROPELLANT	109350	49600.3
OWE	37900	17191.2
TOGW	147250	66791.5



B260A
COST (3 VEHICLES)
(MILLIONS OF 1970 DOLLARS)

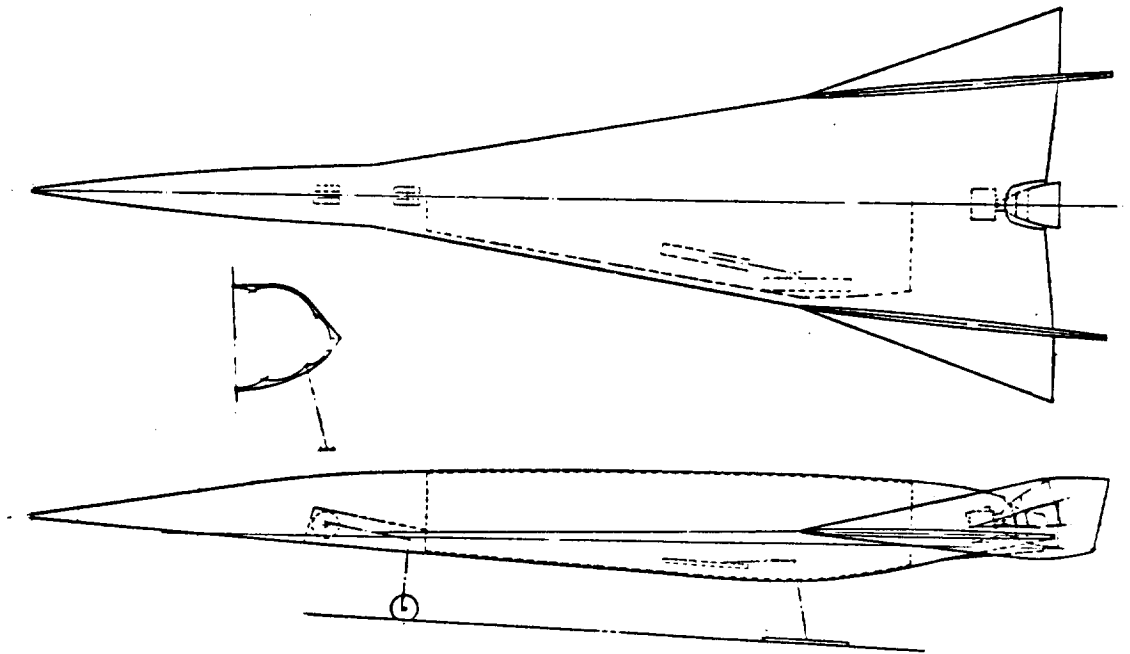
RDT&E	344
INVESTMENT	160
OPERATING	238
TOTAL	742

(C) The scaled version of the Pratt and Whitney Aircraft LR-129 rocket engine employed is 113 in (288 cm) long, and has a conventional bell nozzle. The nozzle has an expansion ratio of 75 and an exit diameter of 60 in (152 cm). The engine uses LO_2 and LH_2 propellants, incorporates both a pre-burner and a main burner in the basic cycle, and is regeneratively and transpiration cooled with hydrogen fuel. At full power, the rocket operates at 3000 psi (2060 N/cm²) chamber pressure at a mixture ratio of 6.0. The engine is operated at full power from launch to the start of cruise; it is throttled back to approximately 5% of full power during cruise to maintain stable flight. The engine is scaled to produce 220,000 lb (561,000 N) vacuum thrust which results in a vehicle TVAC/TOGW of 1.5. The vacuum propellant specific impulse is 451 sec. The performance of the engine when throttled for cruise operation is presented in Figure 4-66.

4.3.7 (U) UNMANNED ROCKET CONFIGURATION (CLASS B284) - Design B284A, Figure 4-67, is the baseline configuration representing an unmanned, Mach 12, air launched, all-body vehicle. The performance, weight, and cost data developed for this configuration are shown in Figure 4-68. The range, flight time and cruise altitude are all quite similar to the manned B260A vehicle.

(C) The scaled version of the PWA LR-129 rocket engine employed is 68 in (173 cm) long, and has a conventional bell nozzle with an expansion ratio of 75 and an exit diameter of 34.5 in (87.5 cm). The engine uses LO_2 and LH_2 propellants, incorporates both a pre-burner and a main burner in the basic cycle, and is regeneratively and transpiration cooled with hydrogen fuel. At full power, the rocket operates at 3000 psi (2060 N/cm²) chamber pressure at a mixture ratio of 6.0. The engine is operated at full power from launch to the start of cruise; it is throttled back (to approximately 6% of full power) during cruise to maintain stable flight. The engine is scaled to produce 111,900 lb (496,000 N) vacuum thrust which results in a vehicle TVAC/TOGW of 1.5. The vacuum propellant specific impulse is 451 sec. The performance of the engine when throttled for cruise operation is presented in Figure 4-68.

(U) FIGURE 4-67 B284A CHARACTERISTICS



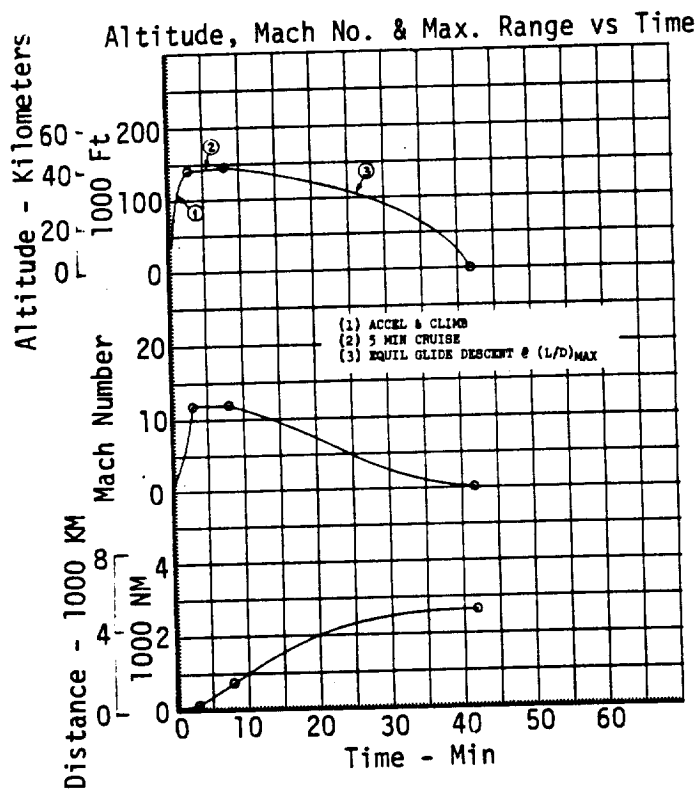
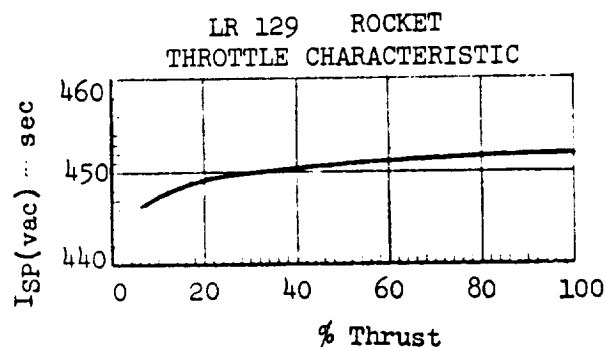
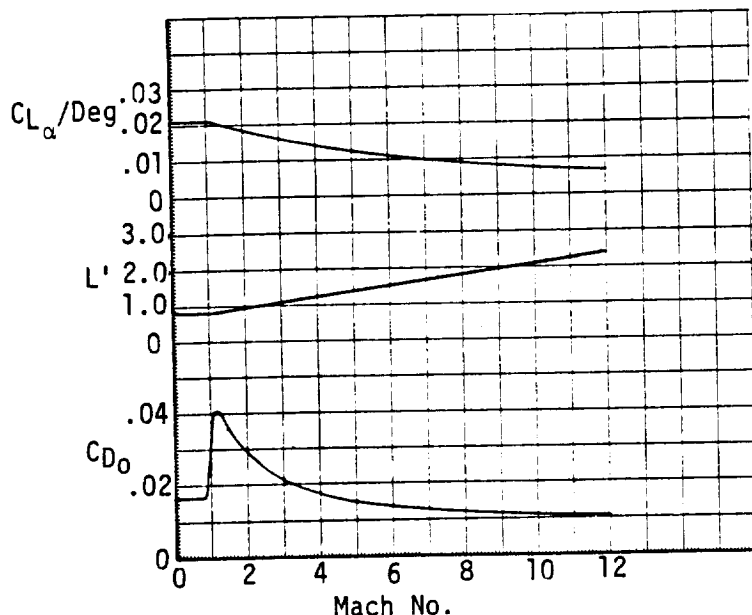
<u>PARAMETERS</u>		<u>PHYSICAL CHARACTERISTICS</u>			
Mach No.	12	SIZE:	ft	(m)	
TVAC/TOGW	1.5	Overall Length	84.0	(28.60)	
Test Time	5 min	Total Span	30.9	(9.42)	
		Height	14.3	(4.36)	
Payload	1000 lb (453.6 kg)	AREAS:	ft ²	(m ²)	
Load Factor	5.0 (Struct)	Planform	881.	(81.84)	
	3.5 (TPS)	Wing	727.	(67.54)	
Basing	Dual	Tips, Total	109.	(10.13)	
		Vert.Tails, Total	178.	(16.54)	
Thermal Prot.Sys.	Passive	VOLUMES:	ft ³	(m ³)	
Tankage	Integral	Total	3810	(107.82)	
		Propellant	2190	(61.98)	
Fuel Type	Subcooled LH ₂	PROPULSION:	ACCEL	CRUISE	
Wing Loading	84.7 lb/ft ² (413.54 kg/m ²)	Engine	(1) LR-129 RKT	(1) LR-129 RKT	
LE Sweep	80°	Thrust (VAC) lb	111,885	-(Throttled)	
		(N)	(497,664)	-	

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(C) FIGURE 4-68
B284A CAPABILITIES

CL_α , L' , & CD_0 vs Mach No.
Mach 12 Air Launched Rocket/Rocket A/B (Unmanned)



B284A WEIGHT		
	lbm.	kg
STRUCTURE	15425	6996.7
EQUIPMENT	4785	2170.4
PROPULSION	2980	1351.7
PROPELLANT	51400	23314.6
OWE	23190	10518.8
TOGW	74590	33833.5

B284A COST (5 VEHICLES) (MILLIONS OF 1970 DOLLARS)	
RDT&E	234
INVESTMENT	176
OPERATING	188
TOTAL	598

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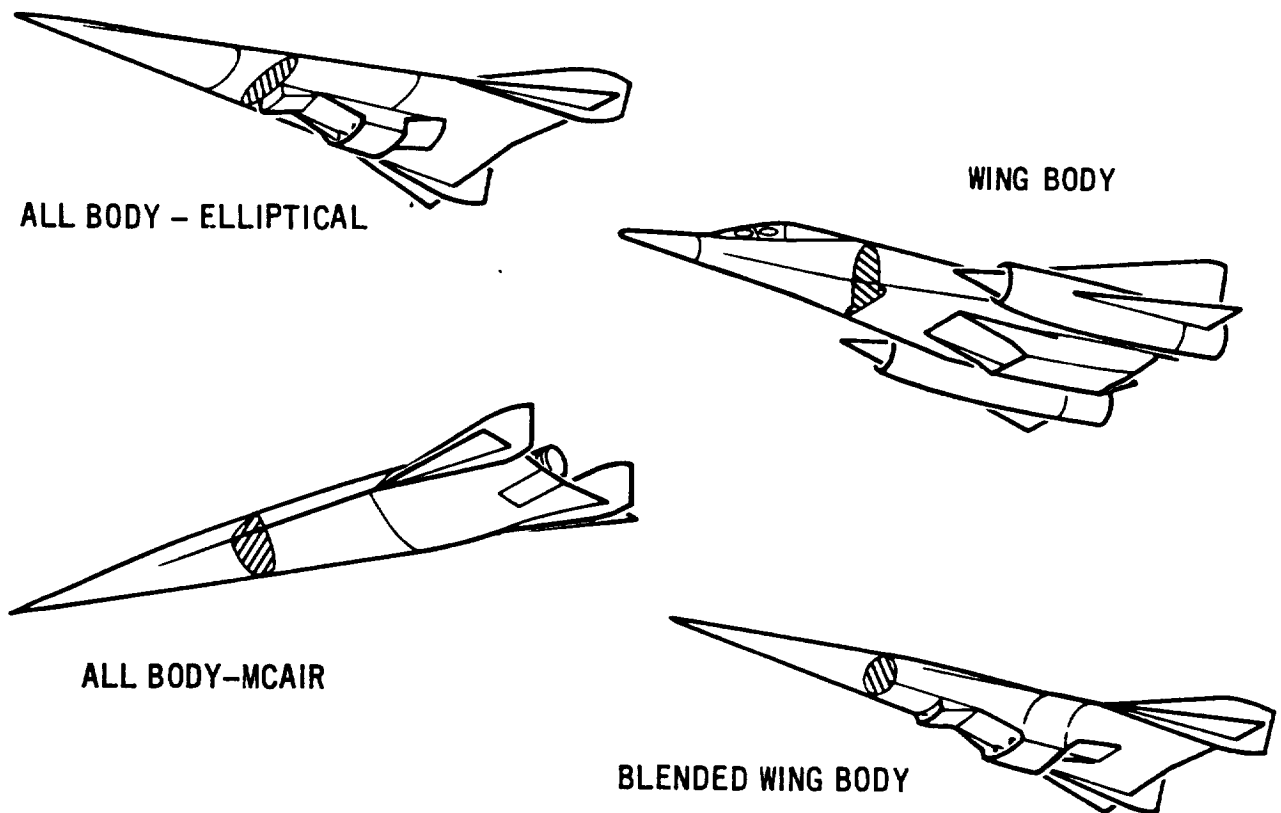
4.4 PARAMETRIC STUDIES

(U) The purpose of the Phase II parametric studies was to refine the Phase I selected vehicles in order to determine the "near optimum" research vehicles that will satisfy research requirements for the class of operational vehicles defined in Volume II. To do this it was necessary to determine the sensitivities of cost and the research value of the flight test vehicles to the various vehicle shape, propulsion, and mission requirements.

(U) The seven baseline vehicles are divided into four basic configurations, (vehicle shape), illustrated in Figure 4-69. Configuration B257 was studied in both the elliptic all-body shape and the blended wing body shape. Configurations B207, B232, B233, B260 and B284 were studied in the MCAIR all-body shape and B212 was studied in the wing-body shape. Parametric studies were conducted using the MCAIR all-body shape. The results of these specific vehicles were extrapolated to apply to all vehicles with the same shape.

(U) With each variation in parameters, the representative flight research vehicle was resized to maintain the specified mission, as described in Section 4.2.7. The operational weight empty is a strong measure of the cost of the vehicle. The planform area and the takeoff gross weight were limiting considerations for vehicles to be air launched from a C-5A aircraft. The low speed flight characteristics were used to determine the landing characteristics of the flight test vehicle while the range and flight time determined the operational qualities of the vehicle.

(U) **FIGURE 4-69 VEHICLE CONFIGURATIONS**



MCDONNELL AIRCRAFT

4.4.1 (U) ROCKET ENGINE - The RDT&E cost of the scaled (rubberized) LR-129 rocket engines used in Phase I was a major contributor to program cost. In Phase I a tradeoff had been conducted for one vehicle, which indicated that program cost reductions might be achieved by using off-the-shelf rocket engines. In Phase II, a more detailed study was conducted to establish program cost reductions which would result from the use of off-the-shelf rocket engines in the rocket powered research vehicles. In the study, the vehicle characteristics and differences in total program cost were compared.

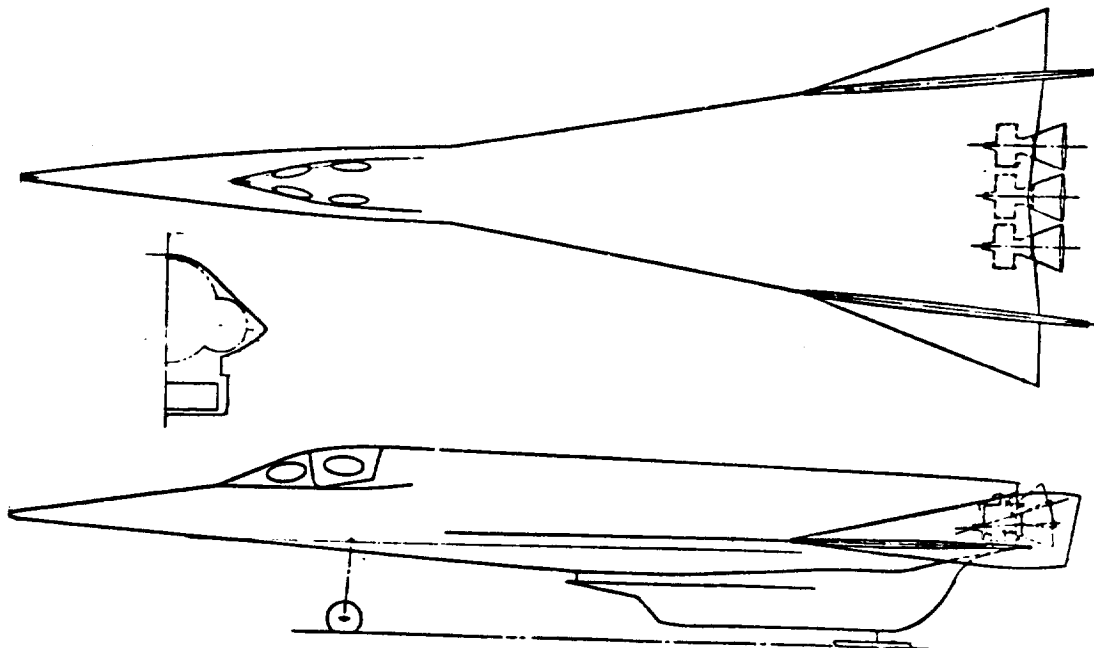
(U) This tradeoff study was conducted in conjunction with the overall vehicle studies so that mission performance, design arrangements, etc., were consistent for the various parametric vehicle designs, and thus, the study provides valid comparisons. The two elements of this tradeoff were (1) a comparison of rubberized and off-the-shelf engines, and (2) a comparison of different quantities of off-the-shelf engines.

(a) (U) Study Process - To conduct the tradeoff, baseline vehicles were designed with a scaled (rubberized) LR-129 rocket engine, and parametric vehicles were designed with an integer number of off-the-shelf rocket engines, while maintaining constant design and mission capability, and approximately the same vacuum-thrust-to-TOGW (T/W) ratio. Figures 4-70 through 4-74 show the vehicles using off-the-shelf engines. Off-the-shelf engines are those which are currently scheduled to complete PFRT by 1975. Minor modification such as throttling, which has been demonstrated for these engines, does not change the off-shelf status. Performance of the off-the-shelf rocket engines meeting these criteria is presented in Figure 4-75.

(b) (U) Evaluation - The dominant factors affecting program cost were (1) engine RDT&E cost; (2) airframe OWE, since it directly impacted airframe costs; and (3) engine performance (I_{sp}), since it directly affected airframe OWE and thus affected airframe cost. The ground test requirements did not impact this trade because, for both types of engines, existing and under-construction facilities are adequate. Research value of the vehicle was not affected by rocket engine choice, so that this parameter did not impact this trade. In Figure 4-76 the cost factors are presented for a typical vehicle. These comparisons show that the major elements which drove the trade results were the much reduced RDT&E cost for the off-the-shelf engine powered vehicles, and the airframe cost increase resulting from lower engine performance. The increased operating and refurbishment cost of vehicles powered with off-the-shelf engines was a minor element. Figure 4-77 illustrates the effect of these elements.

(U) The net benefit of using off-the-shelf engines is shown in Figure 4-78 for four of the rocket powered vehicles. In every case, use of off-the-shelf engines permitted significant cost savings. For one vehicle, the B233 concept, an investigation was made on the number of off-the-shelf engines. In this case the T/W ratio was dependent on the number of engines used. The results showed that the five-engine version was more effective than the six-engine version; (Section 4.4.2.1 discusses why a T/W value less than 1.50 is more effective than the baseline value of 1.50).

(U) FIGURE 4-70 B207B CHARACTERISTICS



PHYSICAL CHARACTERISTICS

GEOMETRY

L.E. Sweep	80°
Overall Length ft (m)	63.3 (19.29)
Total Span ft (m)	23.3 (7.10)
Height ft (m)	11.5 (3.51)

AREAS:

	ft ²	m ²
Planform	520	(48.3)
Wing	413	(38.4)
Tips, Total	62	(5.8)
Vertical Tail, Total	101	(9.4)

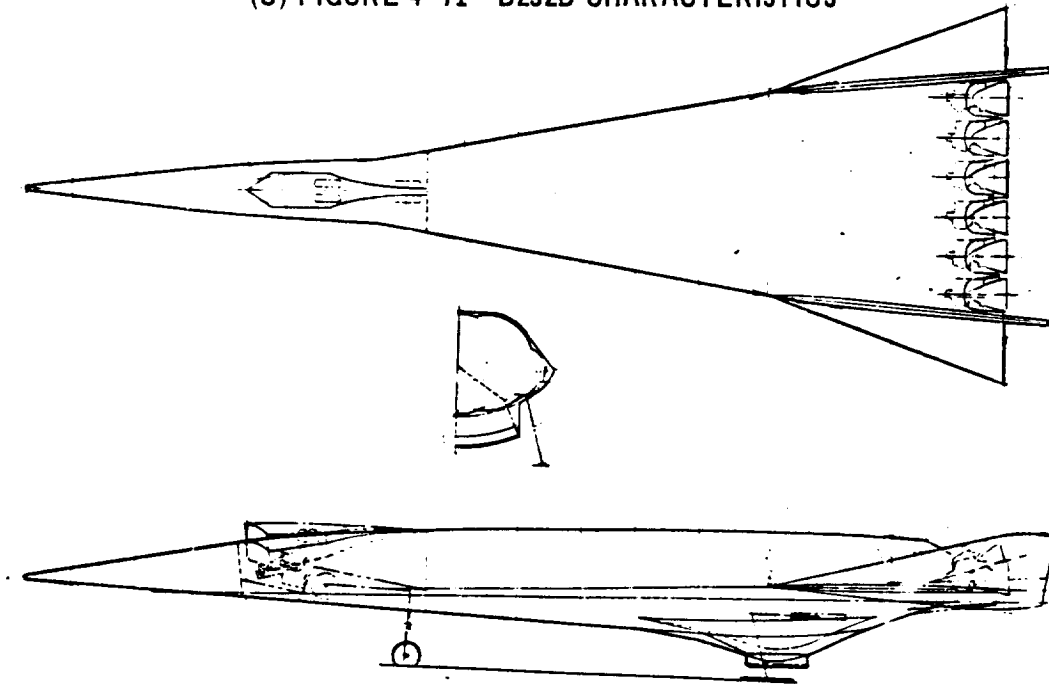
VOLUMES:

	ft ³	m ³
Total	2386	(67.5)
Propellant	1008	(28.5)

PROPULSION:

		ACCEL.	CRUISE
		(3) RL10 (Mod) Rkt	Ramjet
Engine			
Thrust (VAC)	lb	22,750 (ea)	-
	(N)	(101,192) (ea)	-
Thrust (Mach 6 @	lb	-	8740
Cruise Alt)(N)		-	(38,875)
A _{capt}	ft ²	0	16.1
	(m ²)	0	(1.50)

(U) FIGURE 4-71 B232B CHARACTERISTICS



PHYSICAL CHARACTERISTICS

GEOMETRY:

L.E. Sweep	80°
Overall Length ft (m)	83.6 (25.48)
Total Span ft (m)	35.0 (10.67)
Height ft (m)	13.7 (4.18)

AREAS:

	ft ²	m ²
Planform	1187	(110.3)
Wing	1006	(93.5)
Tips, Total	136	(12.6)
Vertical Tail, Total	216	(20.1)

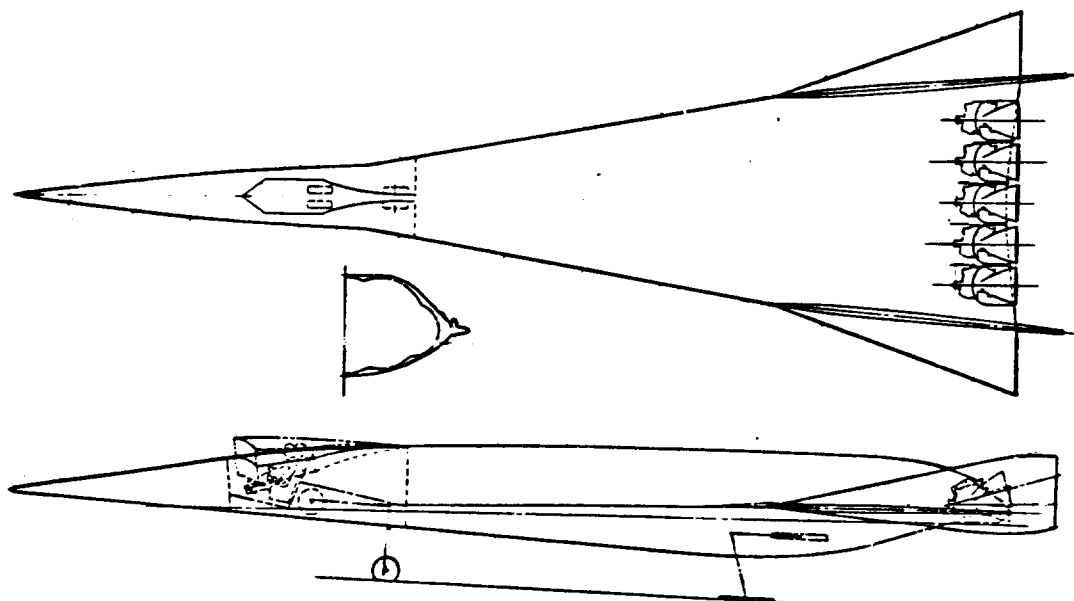
VOLUMES:

	ft ³	m ³
Total	6000	(169.9)
Propellant	3537	(100.2)

PROPULSION:

		ACCEL.	CRUISE
Engine	(6) RL10(Mod)	Rkt	SJ
Thrust (VAC)	1b	22,750 (ea)	-
	(N)	(101,192) (ea)	-
Thrust (Mach 12 @	1b	-	9140
Cruise Alt)	(N)	-	(40,655)
A _{capt}	ft ²	0	53.4
	(m ³)	0	(4.96)

(U) FIGURE 4-72 B233B CHARACTERISTICS



PHYSICAL CHARACTERISTICS

GEOMETRY:

L.E. Sweep	80°
Overall Length ft (m)	93.5 (28.50)
Total Span ft (m)	35.2 (10.73)
Height ft (m)	13.8 (4.21)

AREAS:

	ft ²	m ²
Planform	1202	(111.7)
Wing	1018	(94.6)
Tips, Total	138	(12.8)
Vertical Tail, Total	222	(20.6)

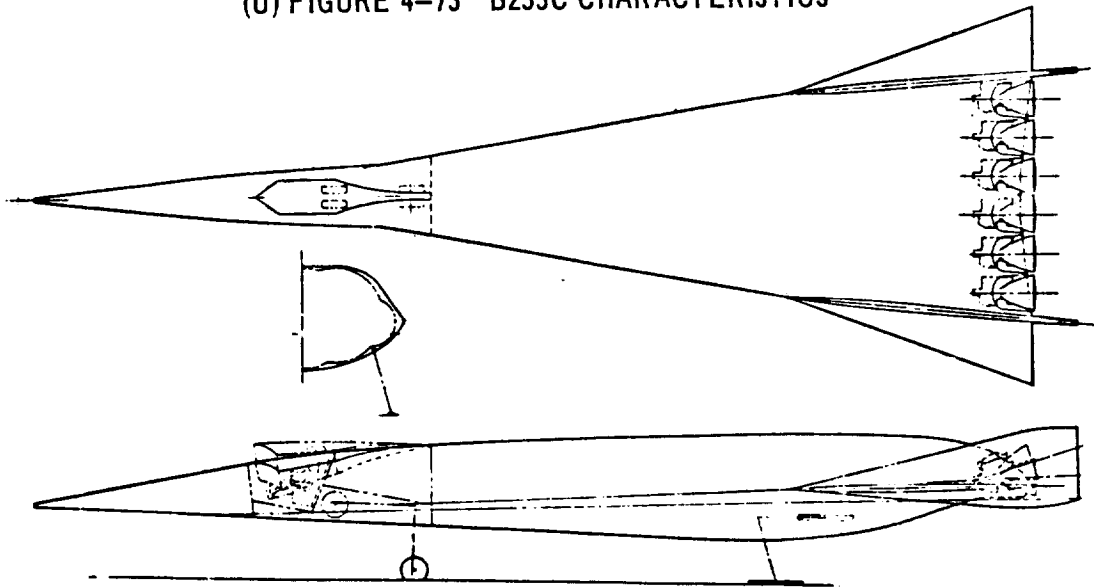
VOLUMES:

	ft ³	m ³
Total	5630	(159.4)
Propellant	3478	(98.5)

PROPULSION:

	ACCEL.	CRUISE
Engine	(5) RL10(Mod) Rkt	(1) RL(Mod) Rkt (throttled)
Thrust lb	22,750 (ea)	
(N)	(101,192)	

(U) FIGURE 4-73 B233C CHARACTERISTICS



PHYSICAL CHARACTERISTICS

GEOMETRY:

L.E. Sweep	80°
Overall Length ft (m)	92.4 (28.16)
Total Span ft (m)	34.8 (10.61)
Height ft (m)	13.7 (4.18)

AREAS:

	ft ²	m ²
Planform	1175	(109.2)
Wing	995	(92.4)
Tips, Total	135	(12.5)
Vertical Tail, Total	217	(20.2)

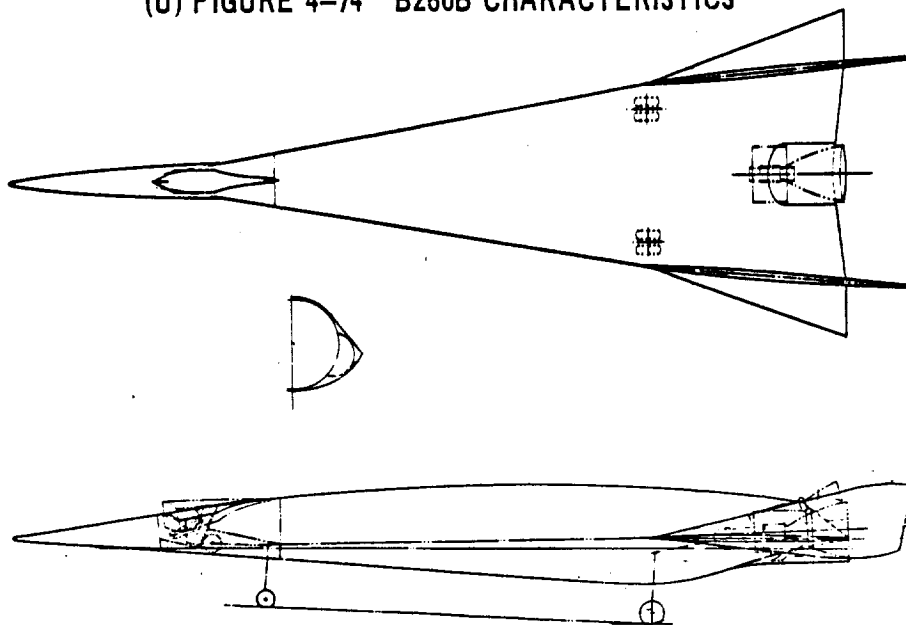
VOLUMES:

	ft ³	m ³
Total	5410	(153.2)
Propellant	3360	(95.2)

PROPULSION:

	ACCEL.	CRUISE
Engine	(6) RL10(Mod) Rkt	(1) RL10(Mod) Rkt (throttled)
Thrust lb	22,750 (ea)	
(N)	(101,192)	

(U) FIGURE 4-74 B260B CHARACTERISTICS

PHYSICAL CHARACTERISTICS

GEOMETRY:

L.E. Sweep	80°
Overall Length ft (m)	113.1 (34.47)
Total Span ft (m)	41.6 (12.68)
Height ft (m)	17.0 (5.18)

AREAS:

	ft ²	m ²
Planform	1708	(158.7)
Wing	1488	(138.2)
Tips, Total	177	(16.4)
Vertical Tail, Total	350	(32.5)

VOLUMES:

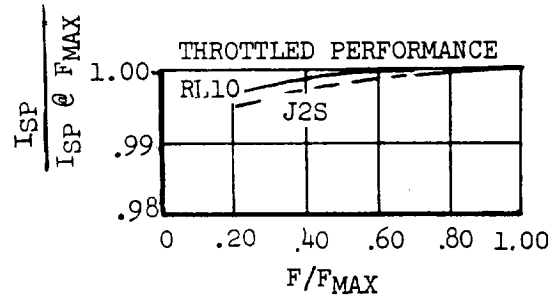
	ft ³	m ³
Total	9380	(265.6)
Propellant	6470	(183.2)

PROPULSION:

	ACCEL.	CRUISE
Engine	(1) J-2S	(1) J-2S (throttled)
Thrust (SLS, uninst)	1b 265,000	
	(N)(1,178,720)	

(C) FIGURE 4-75 OFF-THE-SHELF ROCKET ENGINES

	PWA RL10-A-3-8	Rocketdyne J2S
Vacuum Thrust	22,750 lb (101,000 N)	265,000 lb (1,180,000 N)
Specific Impulse (Vacuum)	430 sec	431 sec
Nozzle Expansion Ratio	32	40
Chamber Pressure	450 psia (310 N/cm ²)	1230 psia (848 N/cm ²)
O/F	5.0	5.5

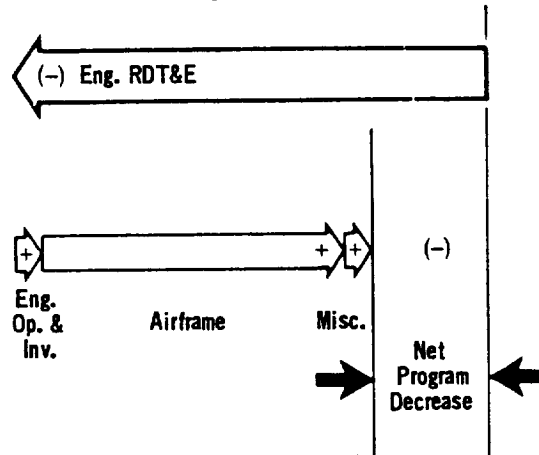
(C) FIGURE 4-76
TYPICAL COST COMPARISON

(M = 12, AIR-LAUNCHED, ALL-ROCKET VEHICLE)

	Specially- Developed Engine (1 x LR-129) T/W = 1.5	Off-the-Shelf Engines (5 x RL-10) T/W = 1.1
Engine Performance Airframe OWE	451 Sec. 24,600 Lb (109,300 N.)	431 Sec. 28,660 Lb (127,700 N.)
Engine RDT&E	119.0 M\$	24.0 M\$
Investment	3.8 M\$	6.9 M\$
Operations & Refurbishment	9.1 M\$	12.1 M\$
Airframe Total Cost (with Refurbishment)	334.4 M\$	391.4 M\$
Miscellaneous	78.5 M\$	83.6 M\$
Program Total Cost	544.8 M\$	518.0 M\$

TP8257-230

(U) FIGURE 4-77
HOW OFF-THE-SHELF ROCKETS
DECREASE COST



(U) FIGURE 4-78 IMPROVEMENT USING OFF-SHELF ENGINES

Vehicle Description	Min. - Cost LR-129 Powered Vehicle (Spec. Developed Engine)		Optional Vehicle (Off-the-Shelf Engine)		Net Program Cost Reduction Using Off-the-Shelf Rockets (M\$)
	T/W	Cost (M\$)	T/W	Cost (M\$)	
M = 6 AIR RKT + RJ (B207)	1.25	612	1.6	593 (3xRL10 Mod)	19
M = 12 AIR RKT + SJ (B232)	1.25	873	1.4	838 (6xRL10 Mod)	35
M = 12 AIR RKT (B233)	1.25	535	1.1 1.4	518 (5xRL10 Mod) 520 (6xRL10 Mod)	17 15
M = 12 HTO RKT (B260)	1.25	737	1.4	728 (1xJ2S)	9

All-Body Configurations Only

(c) (U) Conclusion - The result of this trade study shows that the use of off-the-shelf rocket engines produced the least costly rocket-powered research vehicles. The implications are that such a research vehicle program could be initiated now, without awaiting engine development. Essentially the entire flight envelope could be flown without engine development.

4.4.2 (U) CONFIGURATION DESIGN - Configuration design studies have been conducted to refine and improve the performance of the vehicles retained from Phase I. Two fundamental studies were conducted. The first considered vehicle shaping variations, and the second considered the vehicle thrust loadings. The nominal cross-sectional shape for all vehicles studied is presented in Figure 4-5. The selection of lower surface ramp angles was previously discussed in Section 4.3 and is summarized in Figure 4-79. The results of these studies are presented in the following sections.

(U) **FIGURE 4-79**
FOREBODY RAMP ANGLES

- o 4° For Rocket Acceleration
- o 3° For Turbojet Acceleration.
- o 4°-4°-5° For All Body Scramjet Cruise.
- o 4°-4°-6° For All Body Convertible Scramjet.
- o 3°-6°-6° For Wing Body Convertible Scramjet.

4.4.2.1 (U) Vehicle Shape Study - Vehicle shape studies were conducted on the MCAIR all-body (B233) and elliptic all-body shapes (B257). For the MCAIR all-body shape, the depth of the body at the aircraft centerline was held constant as the leading edge sweep was changed. As the leading edge sweep angle varied, so did the body width, resulting in both an aspect ratio change and a cross-sectional area change. For the vehicles having elliptical cross-sections, the ratio of major diameter to minor diameter was held constant for all leading edge sweep angles. Thus, the cross-sectional area varied, but the cross-sectional aspect ratio did not change. Each of these vehicle shapes was sized for five minutes cruise at Mach 12. The resultant vehicles are shown in Figures 4-80 through 4-83.

(U) For both the MCAIR all-body and for the elliptic all-body configurations, the fineness ratio was dependent on the leading edge sweep angle. The fineness ratio is defined as:

$$f = \frac{\ell}{d_{\text{eff}}}$$

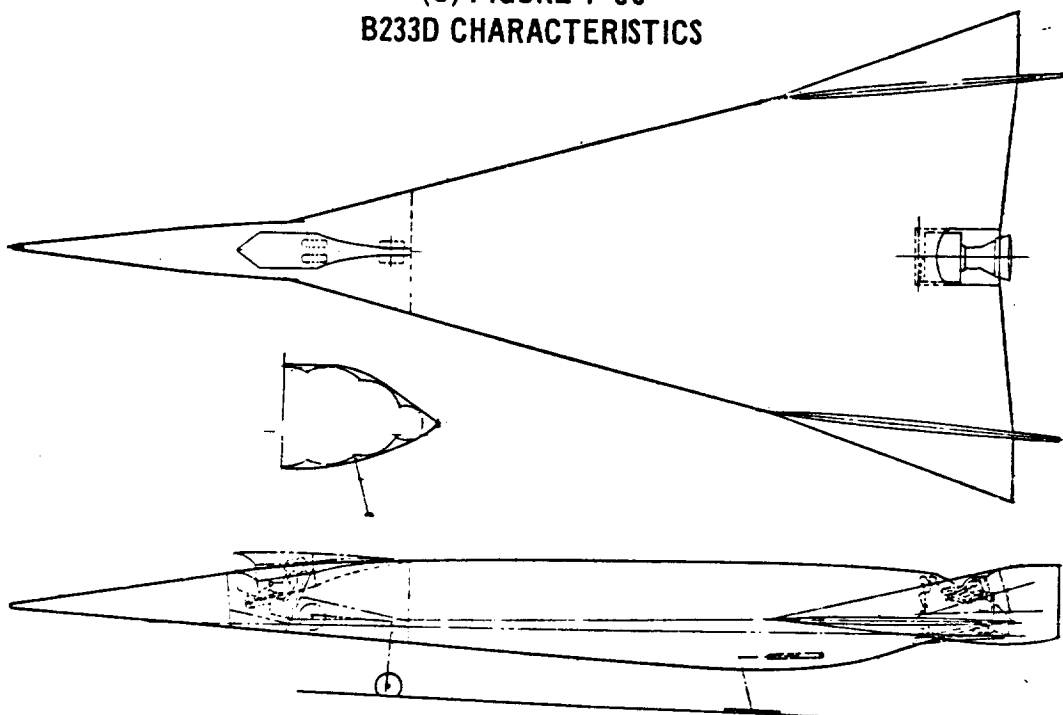
where: ℓ is the length of the body and d_{eff} is the effective diameter of the body. The effective diameter of the body is defined as:

$$d_{\text{eff}} = 2 \sqrt{\frac{S_c}{\pi}}$$

where: S_c is the maximum cross-section of the body.

(U) A correlation between fineness ratio and C_D was found during the study for the basic configurations. This relationship is presented in Figure 4-84 for the B233 vehicle.

(U) FIGURE 4-80
B233D CHARACTERISTICS



PHYSICAL CHARACTERISTICS

GEOMETRY:

L.E. Sweep	75°
Overall Length ft (m)	79.6 (24.26)
Total Span ft (m)	37.3 (11.37)
Height ft (m)	12.5 (3.81)

AREAS:

	ft ²	m ²
Planform	1110	(103.1)
Wing	979	(90.9)
Tips, Total	94	(8.7)
Vertical Tail, Total	151	(14.0)

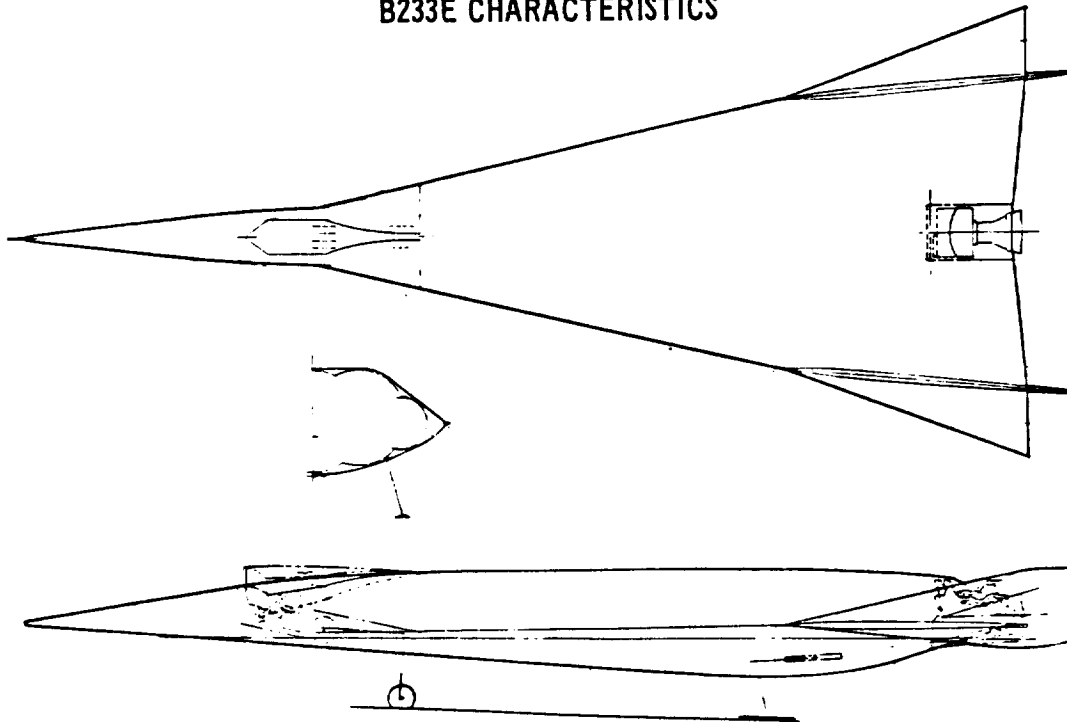
VOLUMES:

	ft ³	m ³
Total	4710	(133.4)
Propellant	2686	(76.1)

PROPULSION:

	ACCEL.	CRUISE
Engine	LR-129 Rkt	LR-129 Rkt
Thrust lb	112,000	
(N)	(499,000)	

(U) FIGURE 4-81
B233E CHARACTERISTICS



PHYSICAL CHARACTERISTICS

GEOMETRY:

L.E. Sweep	77°
Overall Length ft (m)	80.6 (24.57)
Total Span ft (m)	31.8 (9.69)
Height ft (m)	12.6 (3.84)

AREAS:

	ft ²	m ²
Planform	1005	(93.4)
Wing	866	(80.5)
Tips, Total	97	(9.0)
Vertical Tail, Total	156	(14.5)
Wetted, Total		

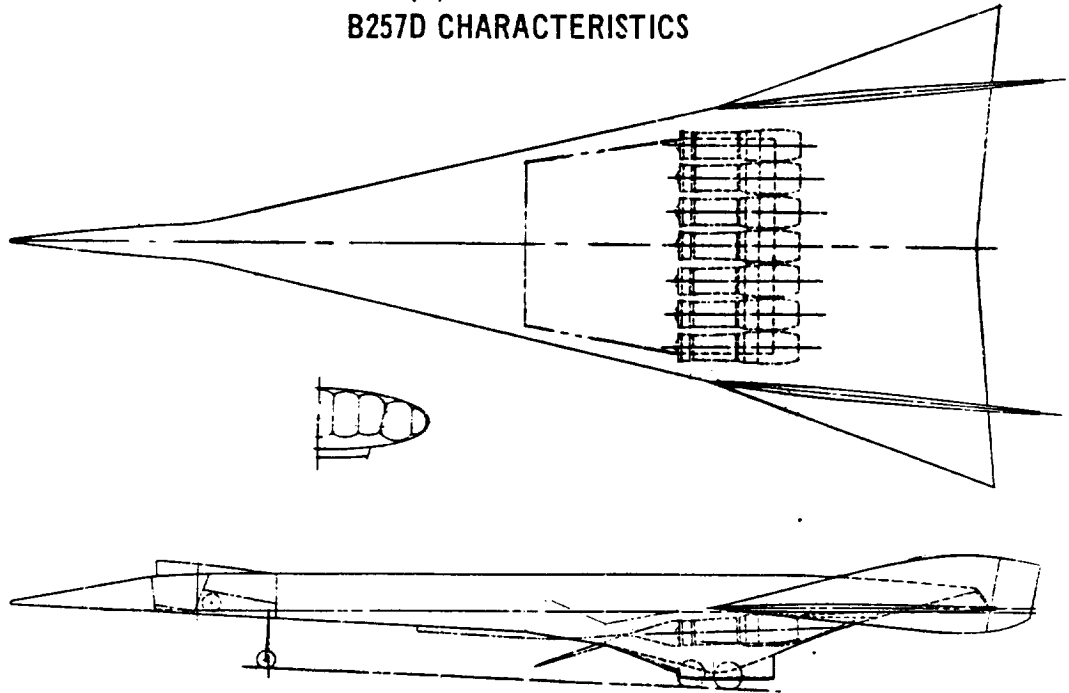
VOLUMES:

	ft ³	m ³
Total	4360	(123.5)
Propellant	2484	(70.3)

PROPULSION:

	ACCEL.	CRUISE
Engine	LR-129 Rkt	LR-129 Rkt
Thrust 1b	112,000	
(N)	(499,000)	

(C) FIGURE 4-82
B257D CHARACTERISTICS



PHYSICAL CHARACTERISTICS

GEOMETRY:

L.E. Sweep	77°
Overall Length ft (m)	148.3 (45.20)
Total Span ft (m)	70.0 (21.34)
Height ft (m)	19.3 (5.88)

AREAS:

	ft ²	m ²
Planform	3900	(362.3)
Wing	3400	(315.9)
Tips, Total	464	(43.1)
Vertical Tail, Total	688	(63.9)

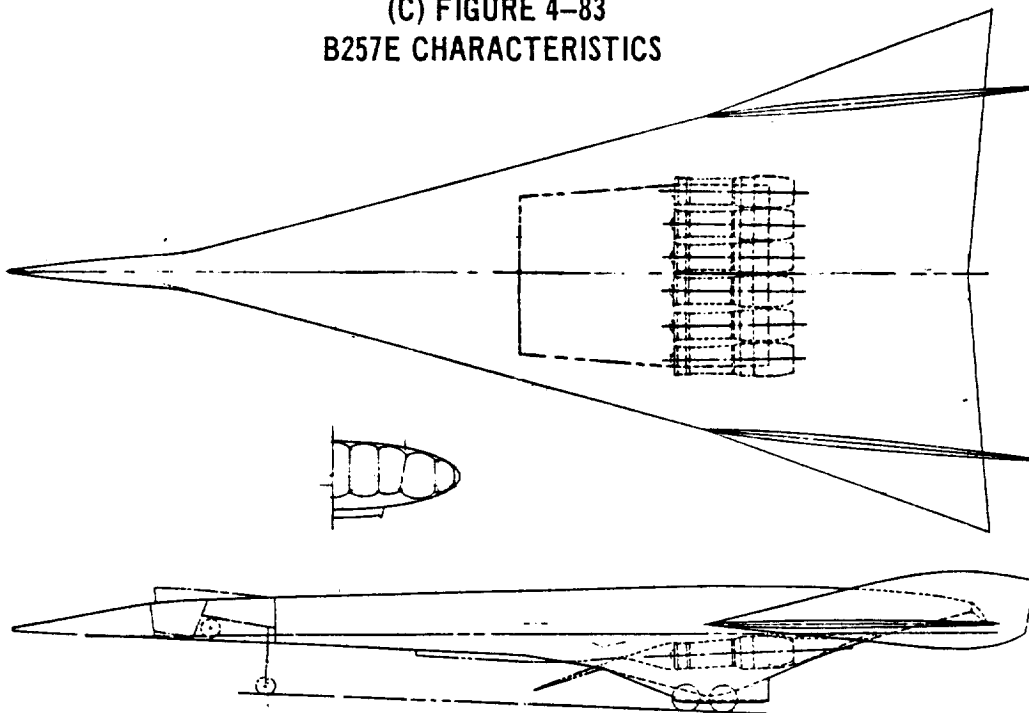
VOLUMES:

	ft ³	m ³
Total	16,150	(457.4)
Propellant	7,540	(213.5)

PROPULSION:

	ACCEL.	CRUISE
Engine	(7) F-100-GE-100	CSJ
Thrust (SLS, UNINST) lb	23,380 (ea)	-
(N)	(103,994) (ea)	-
Thrust (Mach 12 @ 1b	-	22,200
Cruise Alt) (N)	-	(98,746)
A _{capt} ft ²	11.95 (ea)	175.7
(m ²)	(1.11) (ea)	(16.32)

(C) FIGURE 4-83
B257E CHARACTERISTICS



PHYSICAL CHARACTERISTICS

GEOMETRY:

L.E. Sweep	75°
Overall Length ft (m)	125.0 (38.10)
Total Span ft (m)	62.9 (19.17)
Height ft (m)	18.0 (5.49)

AREAS:

	ft ²	m ²
Planform	3040	(282.4)
Wing	2690	(249.9)
Tips, Total	317	(29.4)
Vertical Tail, Total	470	(43.7)

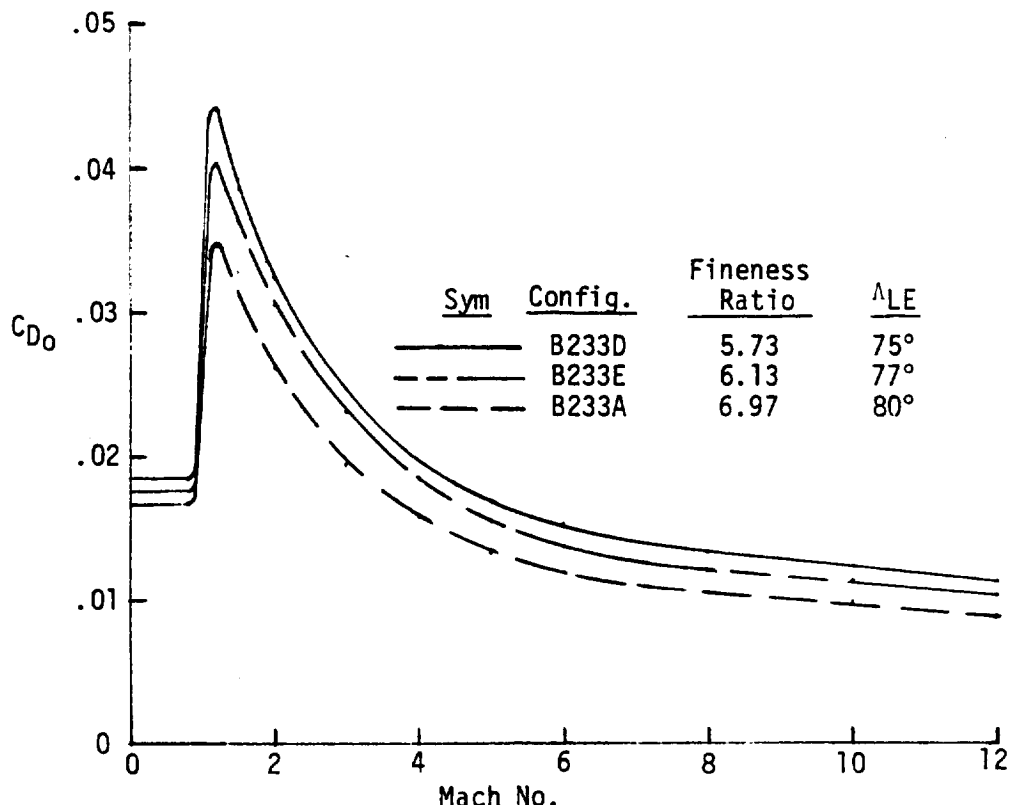
VOLUMES:

	ft ³	m ³
Total	16,150	(457.4)
Propellant	7,540	(213.5)

PROPULSION:

	ACCEL.	CRUISE
Engine	(6) F-100-GE-100	CSJ
Thrust (SLS, UNINST) lb	23,380 (ea)	-
(N)	(103,994) (ea)	-
Thrust (Mach 12 @ lb	-	17,300
Cruise Alt.) (N)	-	(76,950)
A _{capt} ft ²	11.95 (ea)	136.8
(m ²)	(1.11) (ea)	(12.71)

(U) FIGURE 4-84
 C_{D0} VARIATION WITH FINENESS RATIO
(Power On)



(U) Figures 4-85 through 4-87 present the results of the vehicle shape parametric study for the MCAIR all-body rocket configurations. These results apply to configurations B207, B232, B233, B260 and B284.

(U) Figure 4-85 indicates that as the leading edge sweep is increased, the operational weight empty, and consequently the cost of the airplane, is decreased. However, the low speed handling qualities of the airplane are degraded. A measure of this degradation is the low speed $(L/D)_{max}$ which is shown in Figure 4-85.

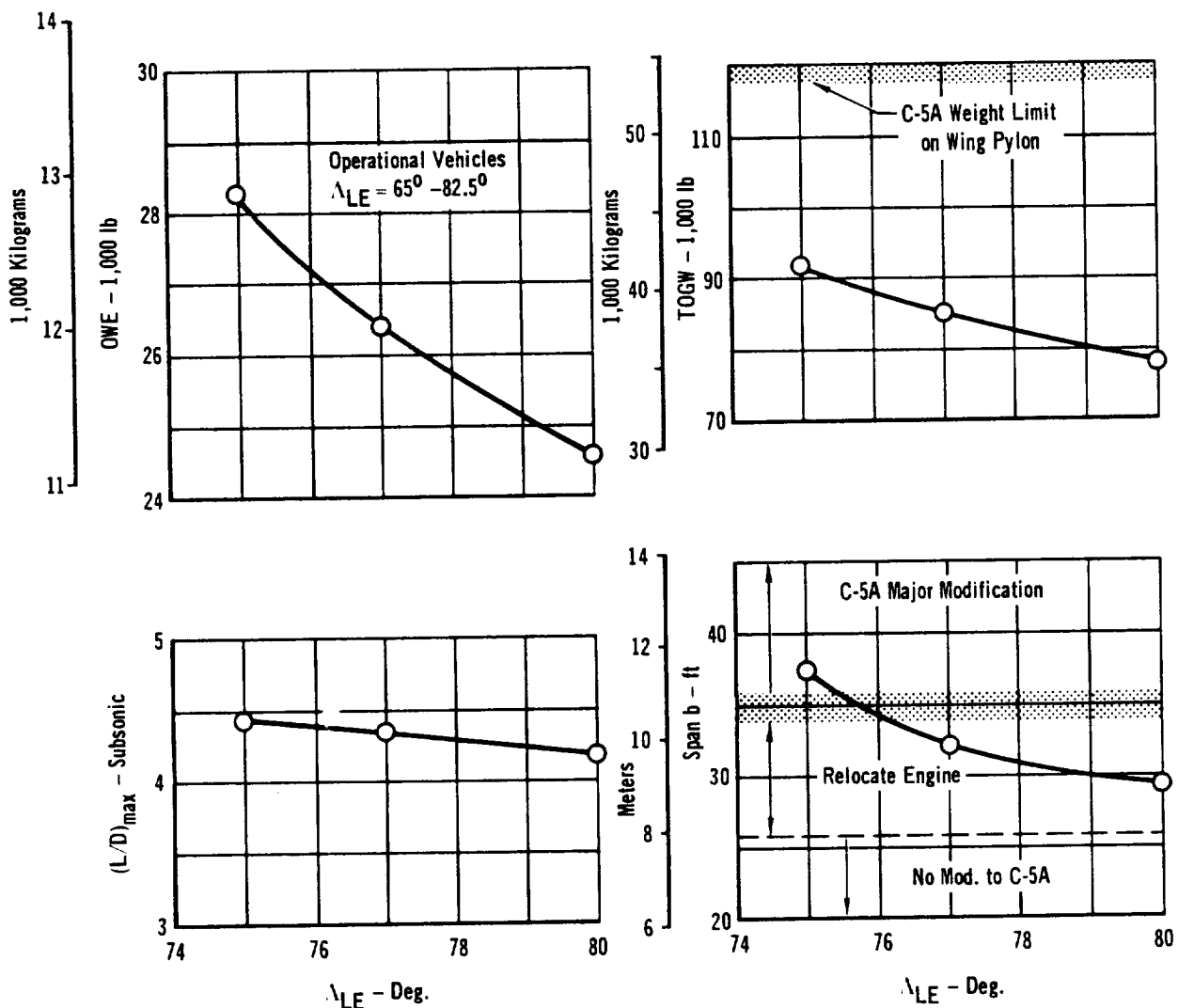
(U) Figure 4-86 shows that as the leading edge sweep is increased, the wing loading at start of cruise increases, the vehicle fineness ratio increases, and the cross-sectional aspect ratio decreases. It was also determined that the weight fractions of the vehicle did not change significantly with leading edge sweep and for all practical purposes were the same. These fractions are presented in Figure 4-87.

(U) Figure 4-85 also shows the various limitations on size and weight the air launched vehicles must satisfy for launching from a C-5A aircraft. The rationale and the modifications to the C-5A that establish these limits were presented in Section 4.1. For the particular vehicle studied (B233), the relationship between leading edge sweep and span indicated that for leading edge sweeps greater than 76°, the

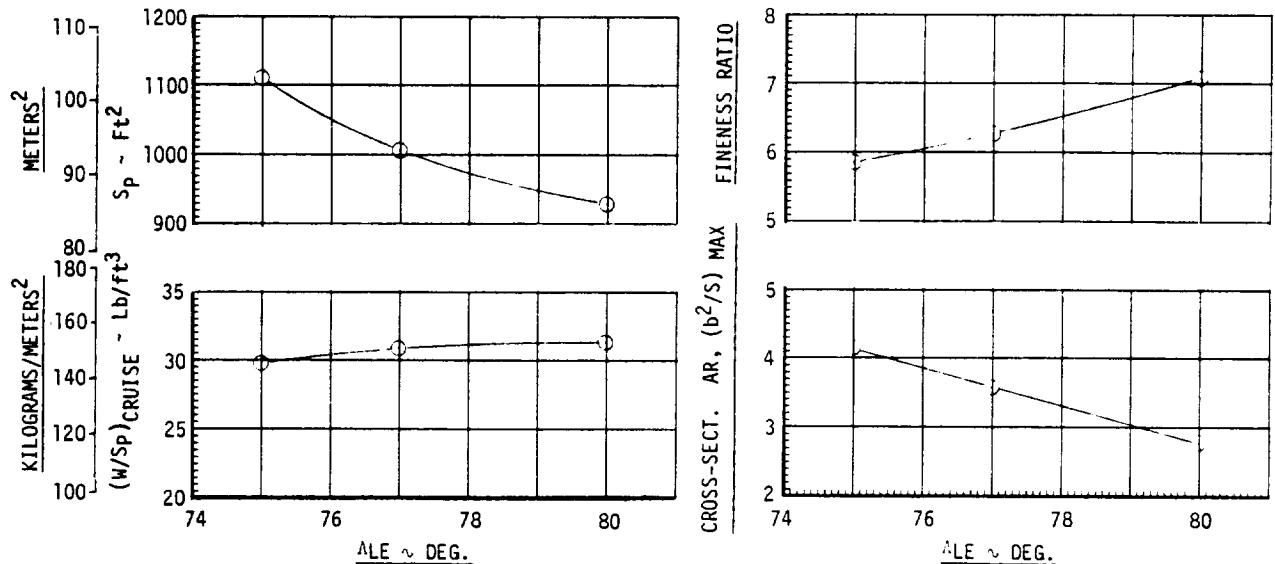
particular vehicle can be launched from the C-5A aircraft with relocation of the right inboard engine further outboard. For angles less than approximately 76° , a relocation of the C-5A inboard engine is required together with a major modification of the C-5A fuselage and landing gear. Figure 4-85 shows that the weight of this MCAIR all-body configuration is well within the capability of the C-5A aircraft.

(U) For the other MCAIR air launched all body configurations (namely configurations B207, B232, and B284), it was necessary to determine the relation between leading edge sweep and span, and in this way determine the extent of modifications necessary to the C-5A aircraft.

(U) FIGURE 4-85
LEADING EDGE SWEEP EFFECTS - MCAIR ALL BODY/ROCKET
 $M_D = 12.0$, Air Launch, Test Time = 5 Minutes



(U) FIGURE 4-86
LEADING EDGE SWEEP EFFECTS - MCAIR ALL BODY/ROCKET
MD = 12.0, Air Launch, Test Time = 5 Minutes



(U) FIGURE 4-87
WEIGHT FRACTIONS
MCAIR ALL BODY/ROCKET

Air Launch Weight Fraction	1.0
Accel. & Climb Fuel Fraction	.63
Cruise Fuel Fraction	.06
OWE Fraction	.31

(U) In analyzing the results of this study, the leading edge sweep for the MCAIR all-bodies was selected as 80°. This was a compromise between cost, as indicated by the operational weight empty, and low speed landing characteristics. With leading edge sweep angles greater than 80°, $(L/D)_{max}$ and lateral control at landing would become critical. Also, only engine relocation is required to launch this vehicle from the C-5A aircraft.

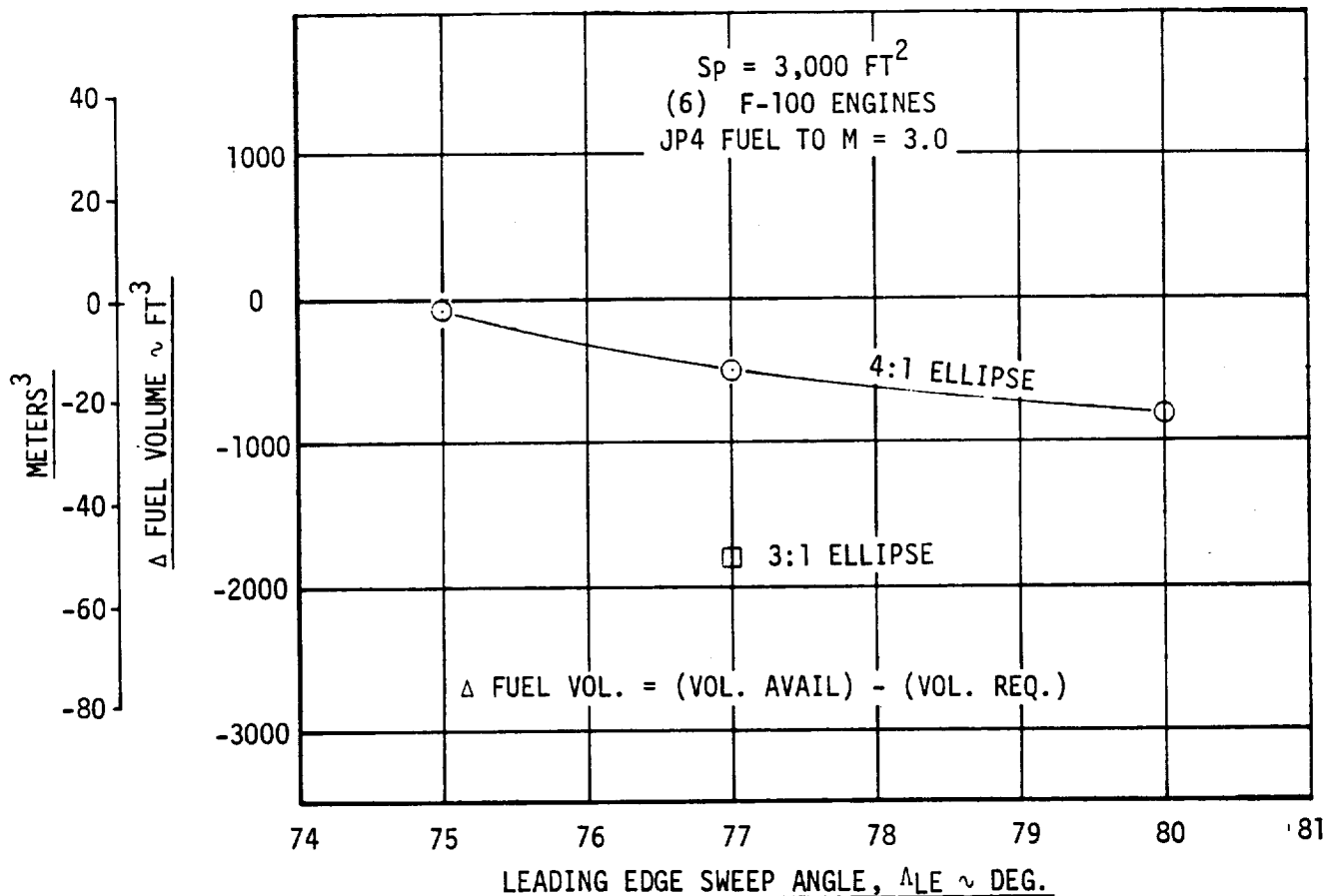
(U) Leading edge sweep studies were also conducted for the elliptic all-body airbreather configuration (B257B). This configuration is ground launched, cruises at Mach 12, and has five minutes test time. It incorporates a group of F-100 airbreather engines (using JP-4 fuel) for climb and acceleration to Mach 3. A hydrogen fueled convertible scramjet is used for climb and acceleration from Mach 3 to 12, and for the cruise phase of flight. Selection of JP in lieu of LH₂ for the turbojet was based on the results of a comparative study described in Section 4.4.3.1.

(U) As the leading edge sweep is increased on the elliptic cross-section vehicle, the drag, and consequently the fuel required, is decreased. However, the fuel available is also decreased, and at a much faster rate. This is illustrated in Figure 4-88 for a fixed size 4:1 ellipse ratio vehicle with six F-100 engines. As indicated in this plot, the fuel volume available minus the fuel volume required became more negative with increasing leading edge sweep.

(U) Also plotted in Figure 4-88 is a point obtained for a 3:1 ellipse ratio vehicle. This vehicle has more fuel volume, but it also has higher drag. The drag effect dominates as indicated by the fact that the fuel shortage increases as the ellipse ratio is decreased.

(U) For a leading edge sweep of 75° , it was possible to obtain design convergence for the vehicle using six F-100 engines. However, to obtain design convergence for a leading edge sweep of 77° required the use of seven engines. For a leading edge sweep of 80° , it was not possible to obtain design convergence to complete the mission requirements. This was due to the limited fuel volume available at high sweep angles.

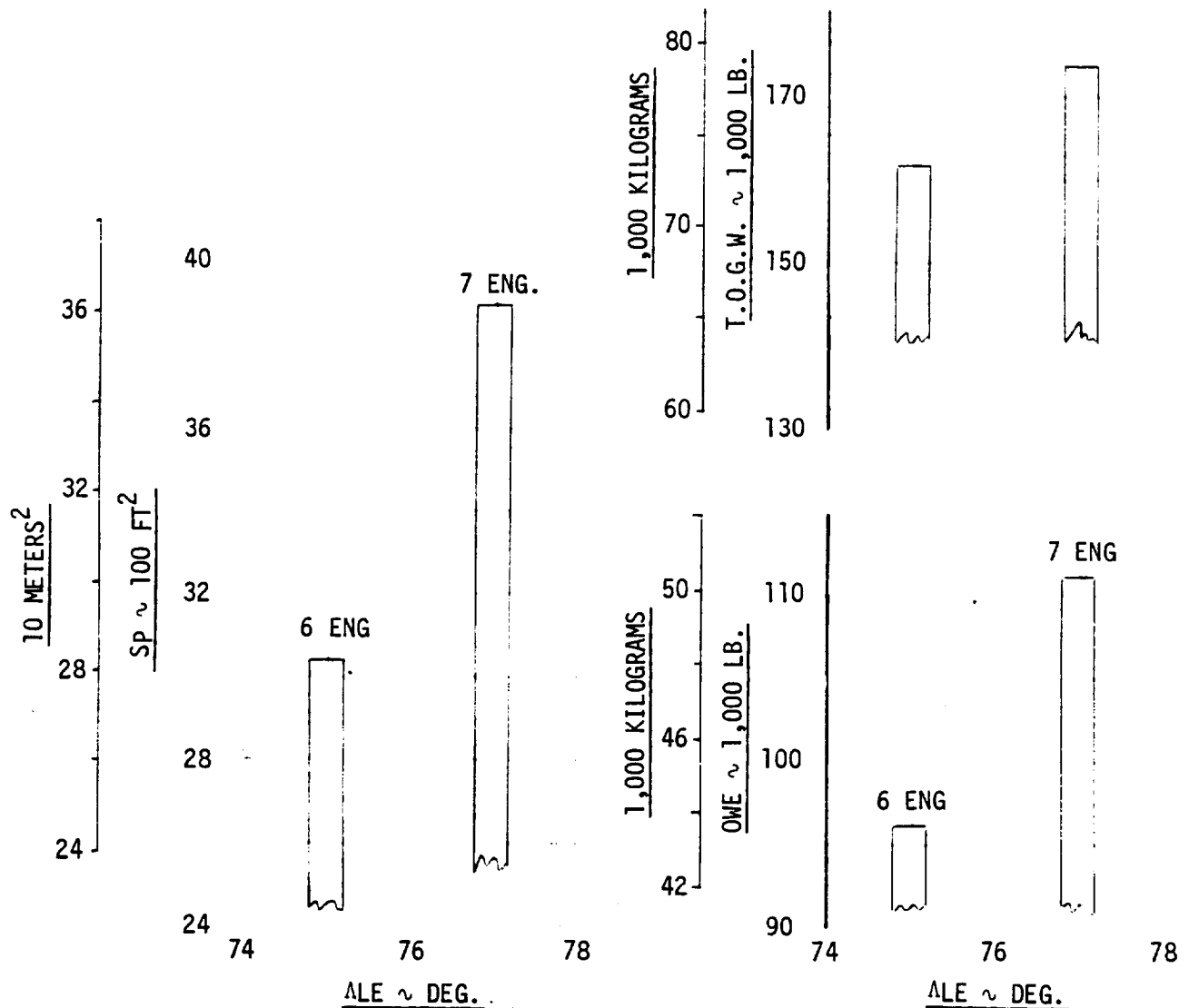
(U) FIGURE 4-88
VOLUME DEFICIENCY OF ELLIPTIC ALL BODY
Airbreather Accel, $M_0 = 12.0$, HTO, Test Time = 5 Min.



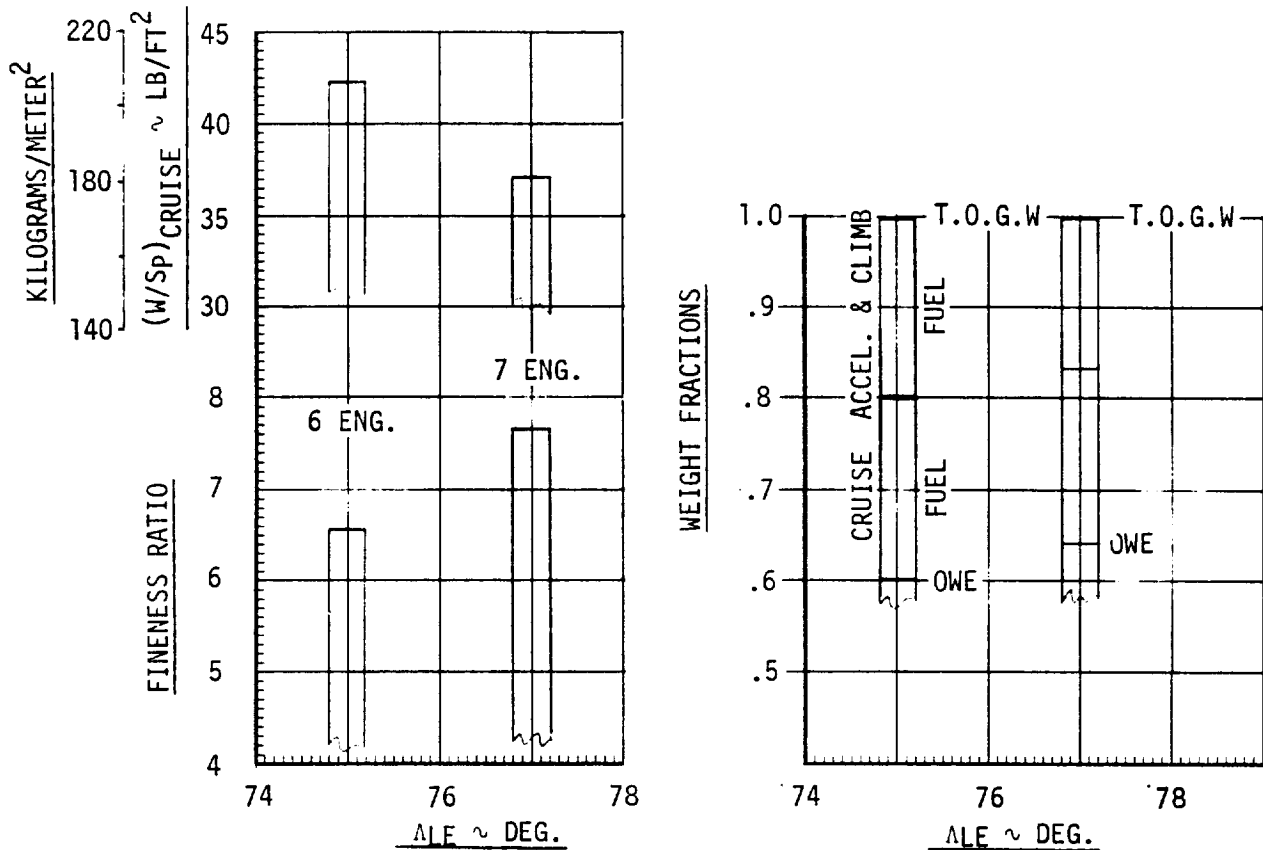
(U) The results of the leading edge sweep studies for the sized 4:1 elliptic cross-section vehicles are presented in Figures 4-89 and 4-90. As indicated by these plots, the resulting airplanes are large, both in planform area and in gross weight. This implies that the cost of these vehicles is high.

(U) The above results indicate that to reduce the size of the vehicle it was necessary to reduce the transonic drag and to increase the volumetric efficiency. With respect to leading edge sweep change, these are opposing effects for a given cross-section shape. Therefore, to improve this configuration it was necessary to change the cross-section shape. This led to the development of the blended wing-body configuration (B257F) shown in Figure 4-61.

(U) FIGURE 4-89
LEADING EDGE SWEEP EFFECTS - ELLIPTIC ALL BODY/AIRBREATHING
Turbojet Accelerator, $M_0 = 12.0$, Horizontal Takeoff, Test Time = 5 Min.



(U) FIGURE 4-90
LEADING EDGE SWEEP EFFECTS - ELLIPTIC ALL BODY/AIRBREATHER
Turbojet Accelerator, $M_D = 12.0$, Horizontal Takeoff, Test Time = 5 Min.



(U) A comparison of the characteristics of the blended wing body with the elliptic body is presented in Figure 4-91. As indicated in this figure, the size of the airplane is considerably reduced. The operational weight empty is reduced about 27,800 lb (12,600 kg) effecting a considerable reduction in the cost of the vehicle. A significant portion of this weight reduction resulted from the number of engines required, four for the blended wing-body compared to six for the elliptic vehicle.

(U) The conclusions obtained from the vehicle shape study are:

(1) For the MCAIR all-body rocket configuration, the leading edge sweep should be 80°.

(2) For the airbreather elliptic all-body shape, the leading sweep of 75° produced the smallest sized airplane. However, this shape was rejected and a blended wing-body was developed to further reduce the size of the airplane.

(U) FIGURE 4-91
COMPARISON OF HORIZONTAL TAKEOFF CONFIGURATIONS

Airbreather With Convertible Scramjet Engines

M = 12 5 Minute Test Time

	Blended Wing Body	4:1 Elliptic All Body
Fineness Ratio	7.53	5.4
Leading Edge Sweep	77°	75°
No. F-100 Turbojets	4	6
Operational Wt. Empty	68,400 lb (31,000 kg)	96,200 lb (43,600 kg)
Planform Area	1,980 ft ² (184 m ²)	3,040 ft ² (282 m ²)
Takeoff Gross Wt.	105,310 lb (47,800 kg)	161,821 lb (73,400 kg)

4.4.2.2 (U) Thrust to Weight Ratio - Thrust to weight ratio parametric studies were conducted for the MCAIR all-body rocket configurations (B207 and B233, Figures 4-51, 4-59, 4-72, and 4-73) and for the blended wing-body airbreather configuration (B257F, Figure 4-61). The purpose of this study was to determine the thrust to weight ratio that would give the minimum operational weight empty (OWE).

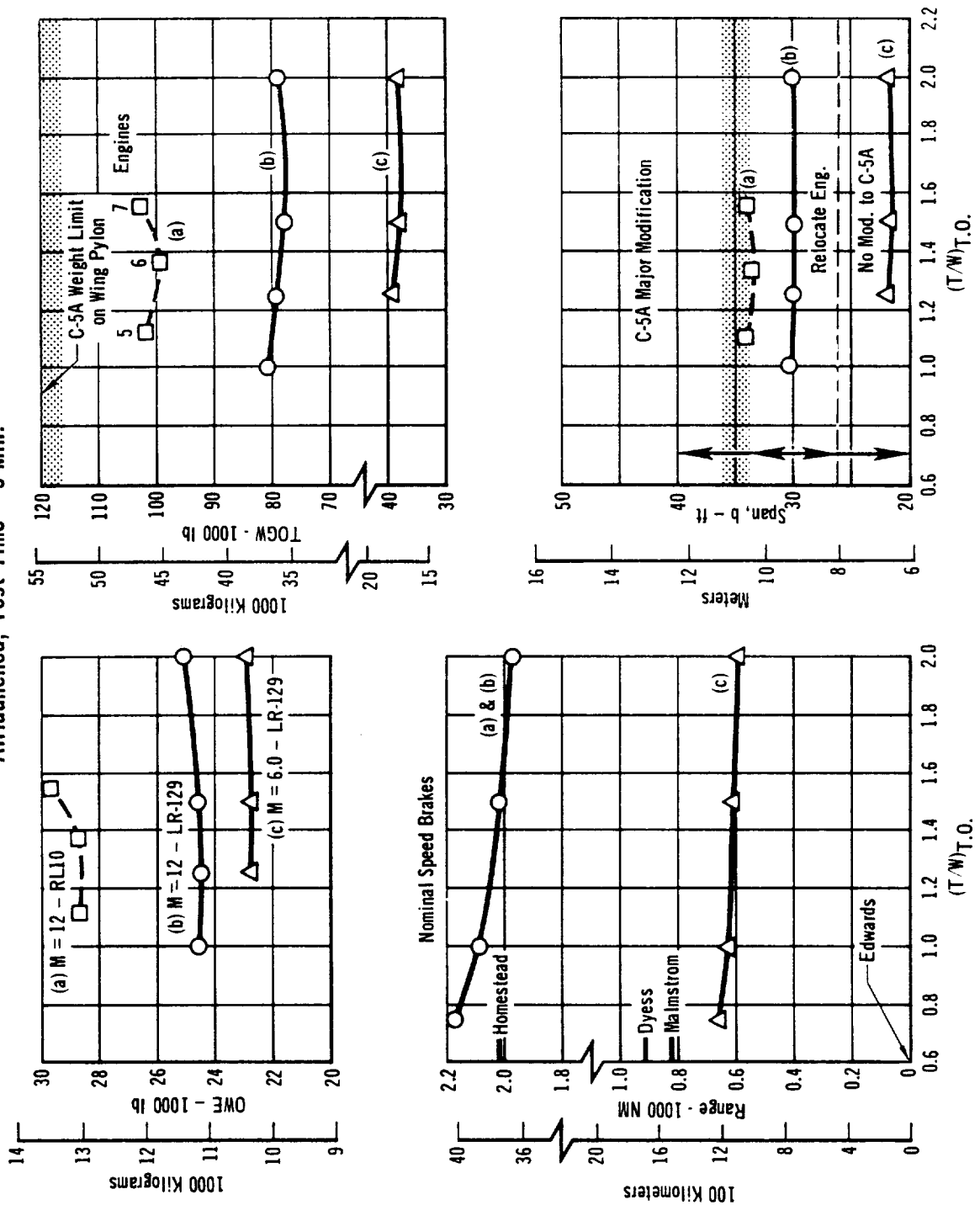
(U) The thrust to weight ratio studies were applied to three different rocket configurations. These configurations were:

- o The M = 12 all-body configuration with the LR129 scaled rocket.
- o The M = 6 configuration with the LR129 scaled rocket.
- o The M = 12 configuration with five, six and seven RL10 engines.

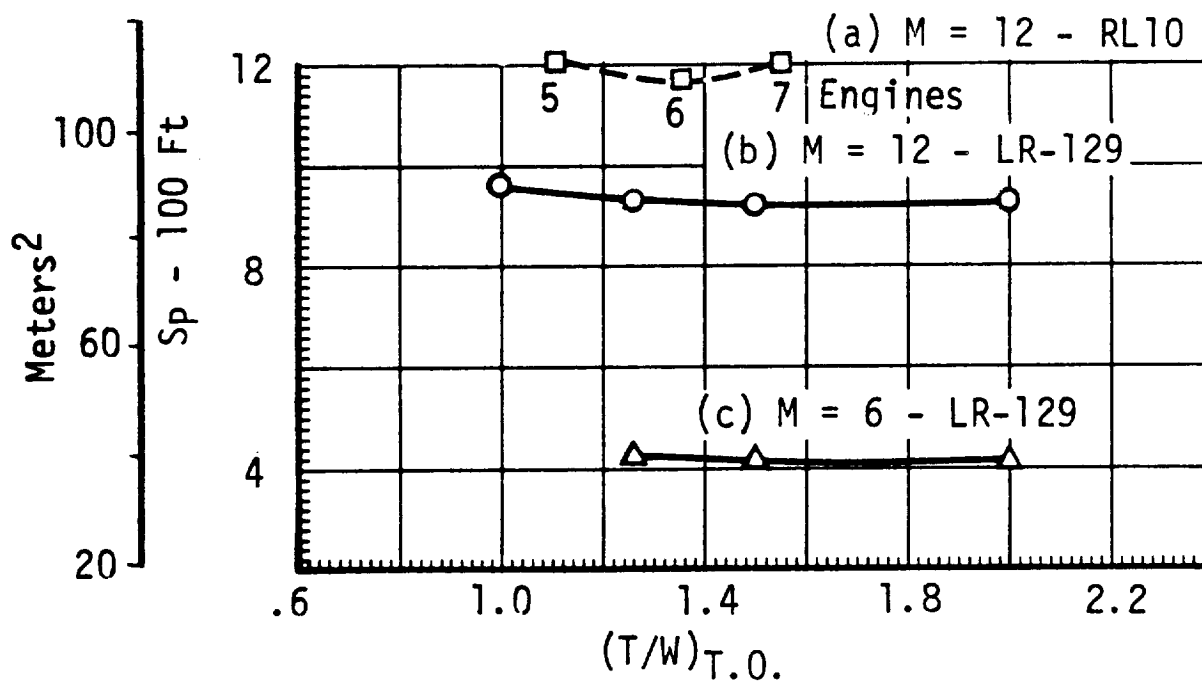
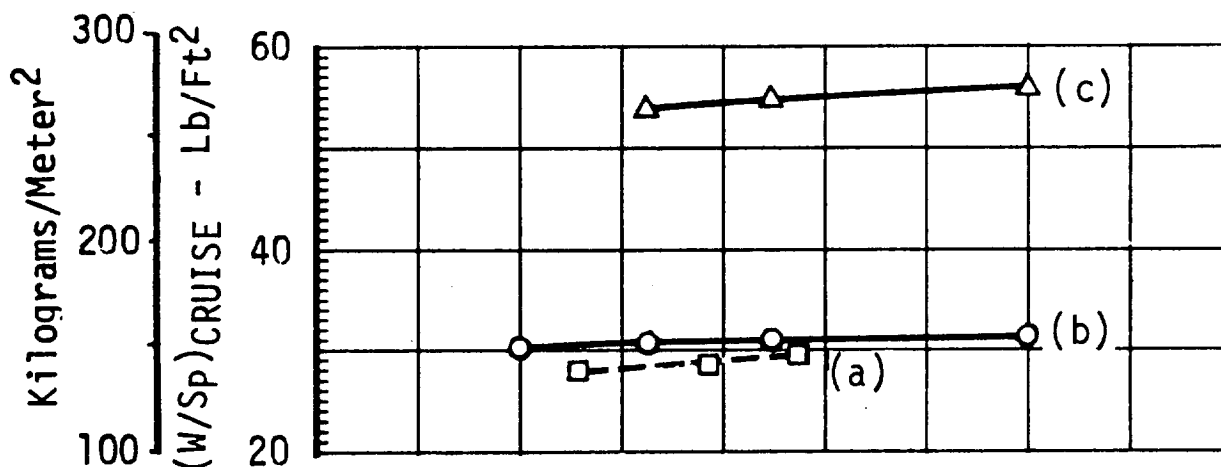
For all cases, the vehicle was air launched and the test time was fixed at five minutes. In this case, the thrust to weight ratio is the ratio of vacuum thrust to the take-off gross weight. The results of these studies were applicable to configurations B207, B232, B260 and B284.

(U) Figures 4-92 and 4-93 present the results of this study. Shown on Figure 4-92 are the C-5A weight limitations and modification constraints previously discussed in Section 4.1. From the standpoint of weight, all these vehicles can be launched from the C-5A airplane. However, with respect to size, Figure 4-92 shows that only the M = 6 configuration can be transported and launched with no modification to the C-5A airplane. Both the M = 12 configurations appear to only require relocation of the right inboard engine of the C-5A further from the fuselage.

(U) FIGURE 4-92
THRUST TO WEIGHT EFFECTS - MCAIR ALL BODY/ROCKET
Airlaunched, Test Time = 5 Min.



(U) FIGURE 4-93
THRUST TO WEIGHT EFFECTS - MCAIR ALL BODY/ROCKET
Airlaunched, Test Time = 5 Min.



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(U) The plot of OWE shows that for the configurations with the scaled LR129 rockets, the thrust to weight ratio (T/W) has a very small effect. As a matter of fact, between 1 and 1.5 T/W, the plot shows an almost constant OWE. For the case of the fixed RL10 engines, the five and the six engine configurations have almost the same OWE.

(U) Also presented in Figure 4-92 are the total ranges, as a function of T/W, for the Mach 6 and the Mach 12 configurations with the scaled rocket engines. This total range is the straight line range from the launch point to the landing point, i.e. the sum of the distance traveled during acceleration, climb and cruise for the powered portion of the flight, and during the unpowered glide along a nominal flight path. This nominal flight path is described in Section 4.4.6.

(U) Figure 4-93 presents the planform area and wing loading for the three configurations.

(U) Thrust to weight ratio studies were conducted on the blended wing-body airbreather (configuration B257) with the F-100 turbojet and convertible scramjet propulsion package. For this study, T/W was defined as the uninstalled sea level static thrust of the F-100 engines divided by the take-off gross weight. Design convergence was obtained using four, five, and six F-100 engines with JP-4 fuel. Design convergence was not obtained for a three F-100 engine configuration since three engines did not produce enough thrust to accelerate through the transonic region.

(U) Figures 4-94 and 4-95 present the results of this study. As can be seen from these plots, the configuration size and weight increases as the number of F-100 engines increases. This means that the minimum number of engines required to perform the mission resulted in a configuration of lowest cost.

(U) Also shown in Figure 4-94 are the straight line ranges from the take-off point to the landing point. The range is the sum of the distance traveled during climb, acceleration, and cruise for the powered portion of the flight and during the unpowered phase of the flight following the nominal glide path to the landing point. This nominal unpowered flight path is described in Section 4.4.6.

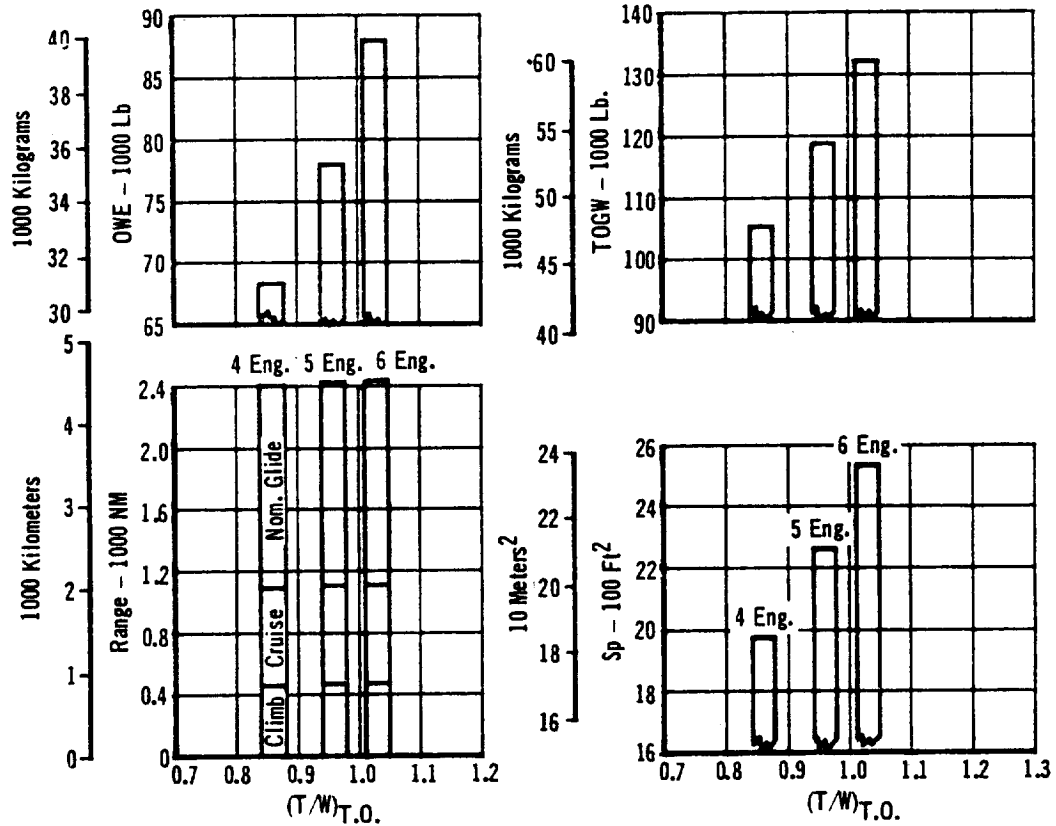
(U) Figure 4-96 presents the time, distance, and acceleration rates for the four engine and the six engine airbreather/convertible scramjet configurations.

(U) The results of the thrust to weight ratio studies are:

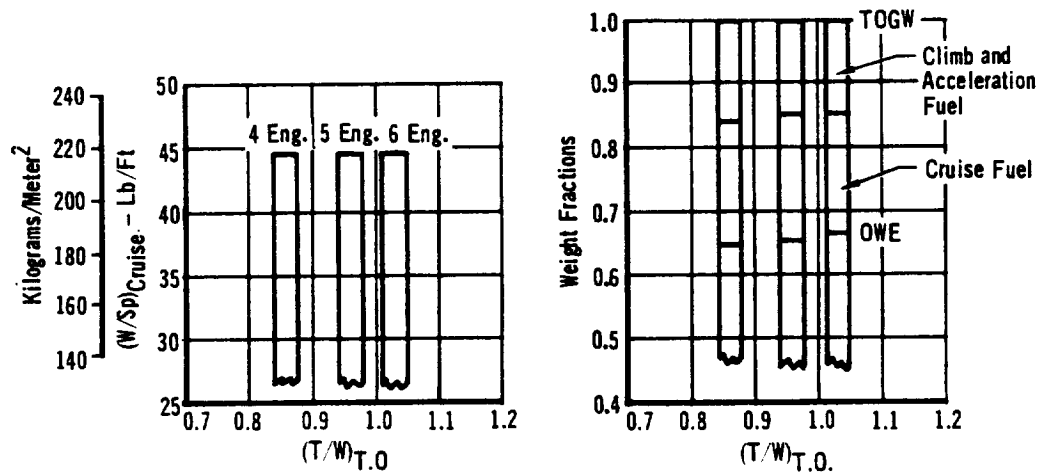
(1) For a scaled rocket engine, the thrust to weight ratio should be between 1.0 and 1.5. A thrust to weight ratio in this range minimizes both vehicle OWE and program cost as indicated in Figures 4-92 and 4-175 respectively.

(2) For off-the-shelf turbojet engines, the minimum number of engines needed to complete the mission gave the lowest program costs.

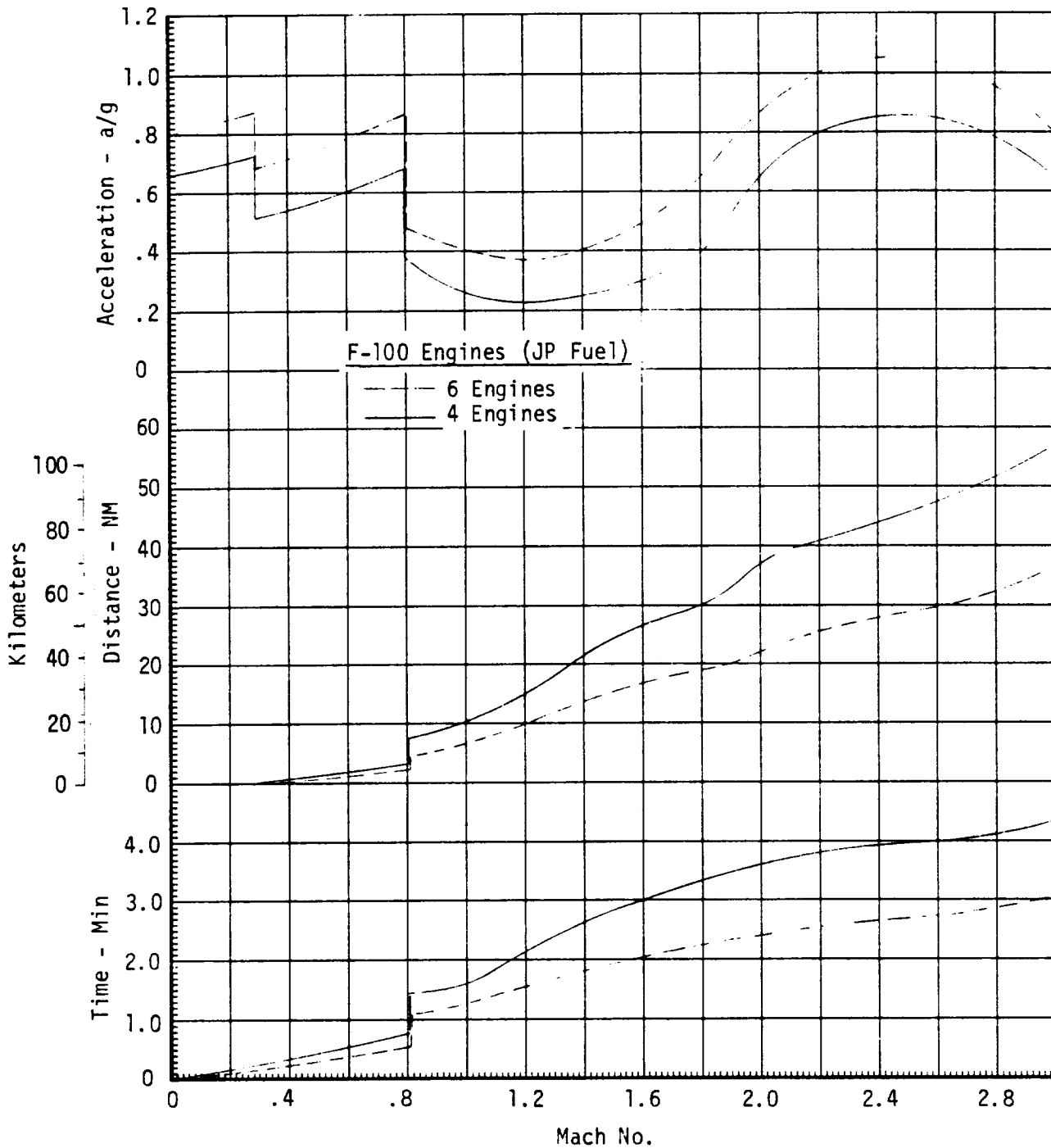
(U) FIGURE 4-94
THRUST TO WEIGHT EFFECTS - BLENDED WING BODY/AIRBREATHER
JP Turbojet/LH₂ Convertible Scramjet, $M_D = 12.0$, Test Time = 5 Minutes



(U) FIGURE 4-95
THRUST TO WEIGHT EFFECTS - BLENDED WING BODY/AIRBREATHER
JP Turbojet/LH₂ Convertible Scramjet, $M_D = 12.0$, Test Time = 5 Minutes



(U) FIGURE 4-96
TRANSONIC ACCELERATION - BLENDED WING BODY/AIRBREATHER
JP Turbojet/LH₂ Convertible Scramjet, $M_D = 12.0$, Test Time = 5 Minutes



4.4.3 (U) PROPELLANTS - Tradeoff studies were conducted to determine the impact of propellant selection on the performance of flight research vehicles and ground support systems requirements. These studies can be classified into two categories: storable vs cryogenic fuels, and subcooled vs normal boiling point (NBP) liquid hydrogen. The results were interpreted in terms of vehicle operating weight empty (OWE), takeoff gross weight (TOGW), and program costs. The conclusions reached were in favor of JP for the initial airbreathing acceleration phase and for the use of subcooled rather than NBP hydrogen.

4.4.3.1 (C) Storable Hydrocarbon vs Cryogenic Hydrogen - One of the principal studies was an evaluation based on climb performance using either conventional JP fuel or subcooled hydrogen in the turbojet mode from takeoff through Mach 3. The vehicle selected for the tradeoff study was a horizontal takeoff, all airbreathing, Mach 12 five minute cruise vehicle (Configuration B257). This vehicle uses the F-100 advanced turbofan engine for acceleration to Mach 3 and a hydrogen fueled dual-mode, convertible scramjet (CSJ) from Mach 1 to Mach 12.

(C) Results of the climb tradeoff are summarized in Figure 4-97. Use of JP for takeoff and initial acceleration results in a decrease in OWE of 17.2%, which scales to a 10.6% reduction in TOGW. The wide difference in OWE reflects the effects of increased tank volume (liquid hydrogen density is .085 that of JP) and the requirement to add an additional engine due to the size and weight increase in the airplane. In order to obtain a converged design with the all hydrogen airplane, five engines are required for the acceleration phase to overcome the additional drag and weight penalty associated with the increased full volume. The use of JP fuel for the climb and acceleration phase enables attainment of a converged design with four engines. A five engine JP climb vehicle was sized (see Figure 4-142) but is not included in this trade-off as the four engine configuration provided a lighter configuration with lower RDT&E and investment costs. The energy content, or heating value, for hydrogen is nearly three times that for JP on a gravimetric basis, so that the effects on TOGW are not quite as pronounced. The differences are even more striking when compared on the basis of flight research program cost. From Figure 4-97 again, it can be seen that the use of JP reduces airplane size to the extent that RDT&E and investment costs are 8.94% lower and operational costs 14.6% lower than for the all-hydrogen airplane. The total reduction in program cost is nearly \$135 million for the smaller aircraft, with no change in either test time or cruise Mach number.

(C) FIGURE 4-97
CLIMB TRADE-OFF RESULTS
(HTO Airbreather: 5 Minutes at Mach 12)

	JP Climb to M3 <u>4 engines</u>	H ₂ Climb to M3 <u>5 engines</u>
OWE	68,400 lb (31,050 kg)	82,600 lb (37,500 kg)
TOGW	105,310 lb (47,800 kg)	119,847 lb (54,600 kg)
RDT&E & INVEST.	\$804 M	\$883 M
OPERATIONAL	\$327 M	\$383 M

(U) In addition, there are several subjective elements which should be summarized. Although the JP fueled aircraft has a higher volumetric efficiency, the hydrogen fueled vehicle's specific impulse performance (higher energy/lb) is somewhat better, as indicated by its low fuel/TOGW fraction (.31 and .35 for the LH₂ and JP fueled vehicles respectively). The use of JP for the initial acceleration and the landing modes results in a slightly lower thermal protection requirement (in the area of the JP tank) than would be necessary for hydrogen. Another advantage of this choice is that unmodified turbojet engines will be available for operation at program inception. However, this choice results in the requirement for "dual mode" operation and separate airborne fuel systems.

(U) Conclusion - Use of JP fuel for the initial acceleration of the flight vehicle rather than hydrogen, reduced the vehicle size and cost.

4.4.3.2 (U) Subcooled Fuel Performance - Past MCAIR hypersonic aircraft studies have shown that significant performance advantages can be realized for operational vehicles by utilizing subcooled rather than normal boiling point (NBP) hydrogen. This tradeoff was performed in Phase II for the flight research vehicles to quantify the performance improvement for the Mach 12 air launched rocket vehicle and the HTO airbreather.

(U) Generally, subcooling provides:

- o Decreased tank ullage
- o Lower tank operating pressures
- o Increased fuel density
- o Unattended hold potential

These advantages are obtained at only a modest increase in propellant cost, which is a minor cost element. No increase in airborne equipment is required to handle the subcooled hydrogen.

(U) For the tradeoff, the subcooled hydrogen was considered at a maximum temperature of 30°R (16.7°K) at lift off; NBP temperature is 36.48°R (20.3°K). Initial fuel load was at a minimum temperature of 25°R (13.9°K) and a slush quality (mass solid/mass solid + liquid) of 50%, currently within the demonstrated state-of-the-art for manufacture and pumping. Unattended ground hold data was developed by calculating the available heat sink from the 50% slush point to 30°R (16.7°K).

4.4.3.2.1 (U) Airborne Performance Improvement - One of the more obvious advantages in subcooling is the increase in fuel density resulting from a 7°R (3.9°K) temperature reduction. The corresponding densities are 4.66 lb/ft³ (.075 gm/cm³) and 4.42 lb/ft³ (.071 gm/cm³), a difference of about 5%. Data from the Saturn program (Vehicle AS-203) reveals that subcooling increases usable fuel volume an additional 1% due to the elimination of vapor entrainment within the liquid (unavoidable with saturated liquids).

(U) The fuel vapor pressure of hydrogen is extremely sensitive to small variations in temperature. This can become a dominant factor in tank operating pressures, and hence in structural design requirements and vehicle weight. Since the maximum liquid temperature occurs near the end of flight, that point determines the fuel vapor pressure/tank operating pressure requirements. Figure 4-98 compares the final

fuel temperatures and resulting tank pressures for subcooled and NBP hydrogen in a typical Mach 12 configuration. The noted tank pressure is essentially vapor pressure at the end of cruise plus a nominal 3 psi (2.07 N/cm²) to allow for net positive suction head (NPSH), pump inlet losses, and tank pressure operating margin at at the highest temperature (highest vapor pressure) condition. Impact of subcooled hydrogen usage on fuel feed and pressurization systems is minimal. The major requirement is active pressurization without venting during ground hold (as opposed to venting for NBP operation) to maintain positive tank pressures. This requirement will not penalize aircraft weight or performance. During all flight phases, present autogenous pressurization systems will provide the necessary pressurant.

(U) FIGURE 4-98
LIQUID HYDROGEN TEMPERATURE RISE - MACH 12 CONFIGURATION

	Initial Fuel Temperature		Final Fuel Temperature		Maximum Tank Pressure (at end of cruise)	
	°R	(°K)	°R	(°K)	psia	(Newtons/cm ²)
5 minutes test time (subcooled LH ₂)	30	(16.7)	32.6	(18.1)	10.5	(7.25)
10 minutes test time (subcooled LH ₂)	30	(16.7)	33.7	(18.7)	12.1	(8.35)
5 minutes test time (NBP H ₂)	36.5	(20.3)	38.6	(21.4)	23.6	(16.3)

(U) The effect of test time (steady state cruise time) is also indicated by comparing the 5 minute and 10 minute cases. The additional time corresponds to a fuel temperature increase of 1.1°R (.61°K), with an attendant vapor pressure increase of 1.6 psi (1.1 N/cm²). The NBP hydrogen fueled vehicle with only 5 minutes test time at Mach 12 requires a tank pressure of 23.6 psia (16.3 N/cm²) at the end of cruise, which is essentially gauge pressure at that altitude. All values are based on an average heat leak to the fuel of 100 Btu/ft²-hr (315 Watts/m²).

(U) The results of these incremental improvements are illustrated in Figure 4-99 for both the HTO airbreathing all LH₂ system and air launched rocket Mach 12 LO₂/LH₂ systems. The effect of introducing JP fuel into the vehicle for improved volumetric performance in the turbojet mode is presented in the previous section. The thrust requirements for the fixed engine size on the TJ/CSJ combination imposes a requirement for six turbofan engines to achieve a design solution for the NBP fueled airplane. Two subcooled aircraft are shown, one with six engines and another with five engines, to illustrate the increments in achieving the 8.2% reduction in OWE with subcooling. As a point of reference, an airplane utilizing 50% slush hydrogen can provide a 17.5% reduction in the NBP aircraft OWE. However, this should be taken as a limit for performance potential, since it provides for no unattended hold time and requires improvements in slush manufacture, storage, and pumping technology. (Availability of 50% slush at launch requires manufacture of higher quality slush to overcome heat losses.)

(C) FIGURE 4-99
EFFECT OF H₂ SUBCOOLING ON VEHICLE SIZE
(Mach 12 Flight Vehicles: 5 Minutes Test Time)

	Fuel State	OWE	No. of Engines
HTO Airbreather TJ to M3 + M1 ≤ CSJ ≤ M12 (LH ₂ fuel TJ and CSJ)	NBP	90,000 lb (40,800 kg)	6
	Subcooled 30°R (16.7°K)	87,600 lb (39,800 kg)	6
	Subcooled 30°R (16.7°K)	82,600 lb (37,500 kg)	5
	50% Slush	74,200 lb (33,700 kg)	5
Air Launch Rocket (LO ₂ /LH ₂)	NBP	25,500 lb (11,580 kg)	1
	Subcooled 30°R (16.7°K)	24,600 lb (11,190 kg)	1
	50% Slush	23,850 lb (10,820 kg)	1

(U) The effects on the air launched rocket vehicle are not quite so pronounced. The fuel density improvement is damped out by the 6:1 O/F ratio and the relatively large quantity of oxidizer on board.

4.4.3.2.2 (U) Ground Performance Improvement - In addition to airborne performance gains obtained through the use of subcooled fuel, advantages are realized by virtue of increased ground hold potential and reduced vapor loss during transport and storage, and elimination of vapor venting near the aircraft. It is considered that all support hardware is within the present technical capability, with the possible requirement for a slush hydrogen pilot plant to demonstrate bulk slush generation techniques on a continuous basis. The impact of subcooled fuel on aircraft ground handling is generalized in Figure 4-100.

(U) FIGURE 4-100
IMPACT OF SUBCOOLING ON GROUND HANDLING

OPERATION

GROUND HOLD
CHILLDOWN/FILL
STORAGE
SPECIAL BASE HANDLING
(Subcool on site)
LOGISTICS/SUPPLY

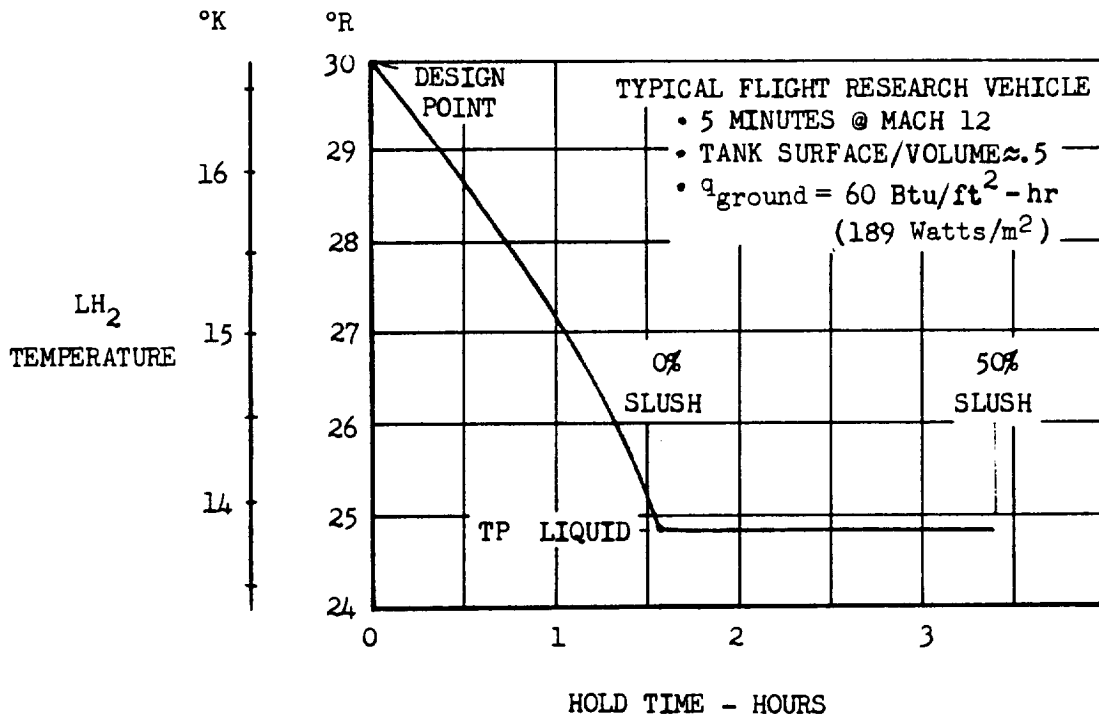
IMPACT

UNATTENDED CAPABILITY
POTENTIAL TIME REDUCTION
ACTIVE PRESSURIZATION REQUIREMENT
ADDITION OF VACUUM PUMPS

NONE

(a) (U) Aircraft Ground Hold - The most pronounced ground performance gain for subcooled hydrogen is in the aircraft ground hold operation. For NBP, constant LH₂ venting is required during ground hold, with liquid replenishment (top-off) to offset boil-off losses. With subcooled fuel, the tank can be isolated from ground support and venting hardware until the maximum allowable flight LH₂ temperature is reached. One minor requirement is the necessity for active (regulated) pressurization to offset tank ullage pressure reduction while the ullage cools during ground hold. The available unattended hold time for the typical flight research vehicle studied in Phase II is about 3-1/2 hours, as depicted in Figure 4-101. The flight vehicle was assumed to be loaded with 50% slush hydrogen, which melts during ground hold at constant temperature to the Triple Point (0% slush) and is then allowed to warm to the design point value of 30°R (16.7°K). Ground hold (and airborne hold for the air launch concepts) mean heat leak is 60 Btu/ft²-hr (189 W/m²). This mean value was determined from consideration of the chilldown heat load, three to four hours ground hold time, and an ultimate steady state ground hold heat leak of 20 Btu/ft²-hr (63 W/m²). It should be noted that this available hold time also corresponds well with pre-launch requirements for the air launch vehicle, providing unattended hold shortens the duration of the pilot's pre-launch time in the pressure suit.

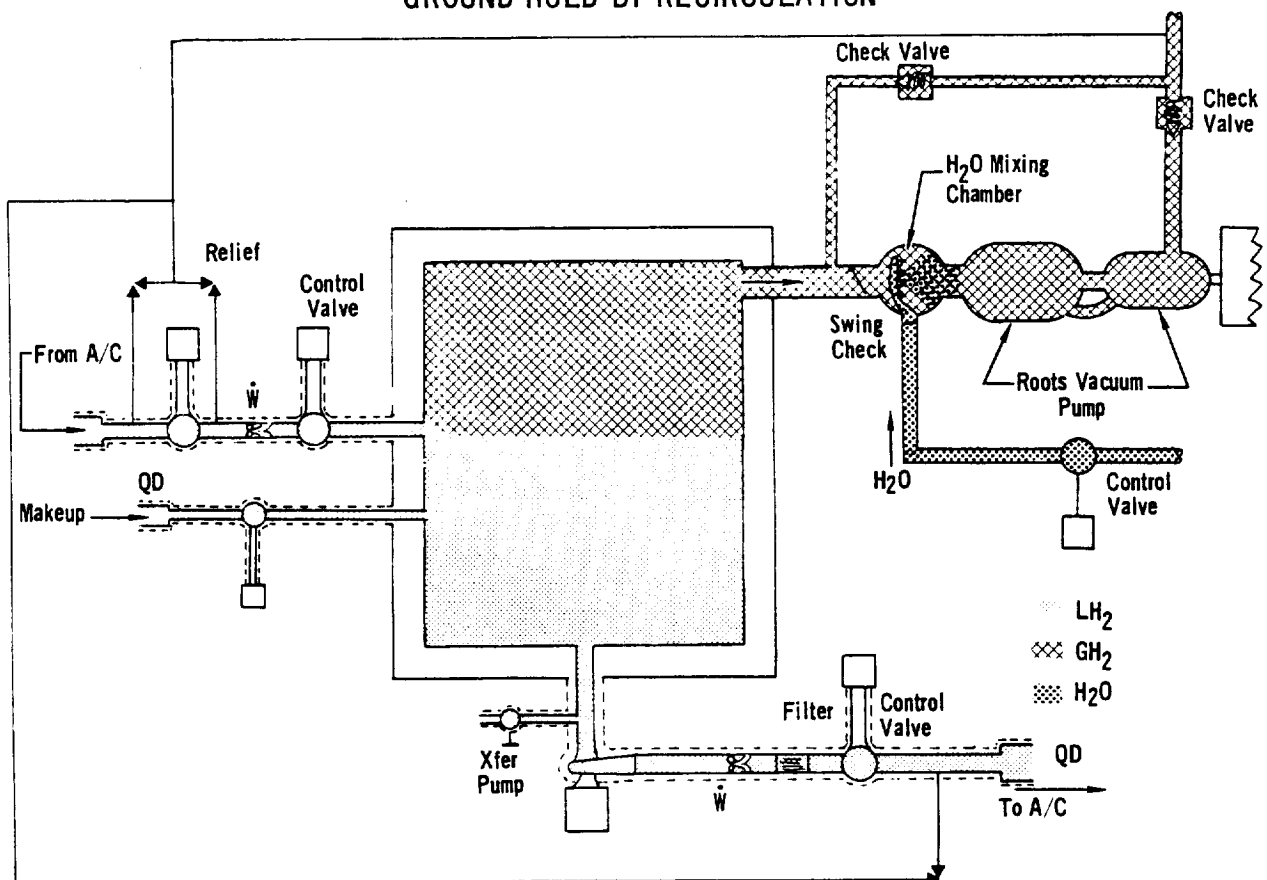
(C) FIGURE 4-101
UNATTENDED GROUND HOLD



(U) If additional ground hold is desired (beyond 3-1/2 hours) due to difficulties in pre-flight preparation, it is possible to provide an attended hold capability for an indefinite time. This is accomplished by off-loading fuel and recirculating it through a fuel ground conditioning unit to maintain the desired subcooled state. Other techniques were investigated, such as helium injection or total aircraft recycle, but these were found to be costly and operationally unattractive. Recirculation through ground support equipment proved to provide the best all around extended ground hold capability. Not only is indefinite hold time available, but the aircraft is immediately ready for operation upon disconnect of servicing equipment.

(U) A schematic of a recirculation ground support trailer is shown in Figure 4-102. The basic principle of operation is low pressure boil-off, or vacuum pumping, in which the temperature of the fluid is lowered by drawing a vacuum in the vapor space, which in turn reduces the equilibrium boiling temperature. For example, to maintain the aircraft fuel load at the design point, the liquid hydrogen is withdrawn from the aircraft at a temperature of 30°R (16.7°K) at a flow rate of 2.26 lb/sec (1.03 kg/sec). The flow rate is controlled by the control valve downstream of the flow meter, which also reduces the fluid pressure

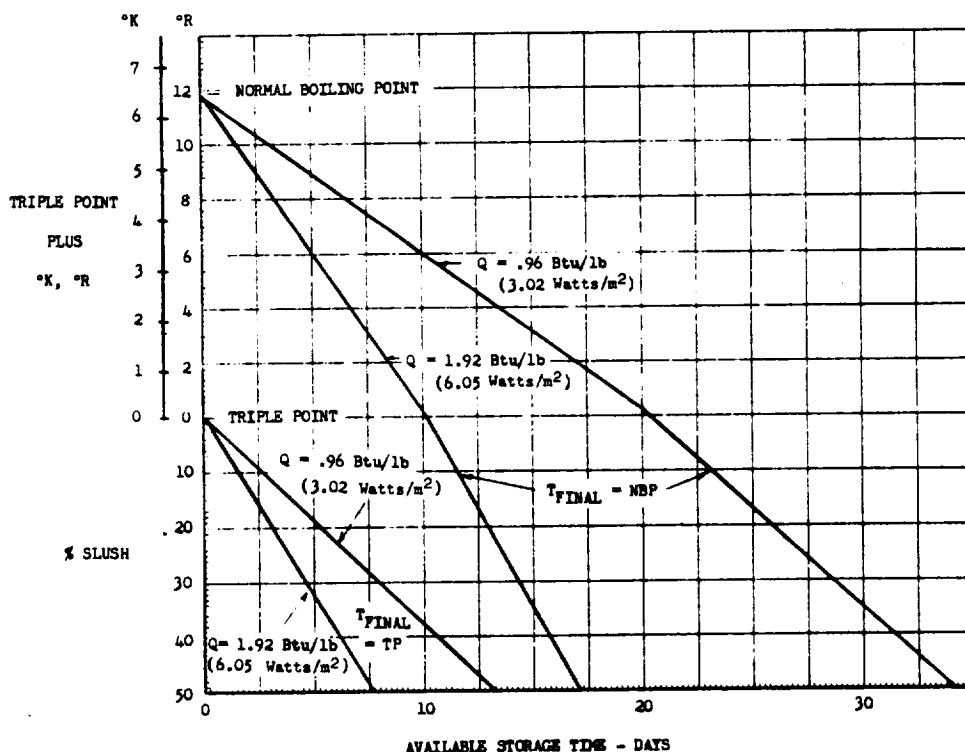
(U) FIGURE 4-102
GROUND HOLD BY RECIRCULATION



to approximately 2.5 psia (1.73 N/cm²). The vacuum pump removes gaseous hydrogen from the fluid conditioning tank at a rate of .09 lb/sec (.041 kg/sec), which is sufficient to negate both the vehicle heat leak and the energy rise from the ground support hardware. Water injection is utilized in the suction end of the vacuum pump to raise the temperature of the gaseous hydrogen to 40°F (277.7°K), while also providing a water seal inside the vacuum pump. This water seal is necessary to maintain rated suction pressures. LH₂ make-up is provided at a rate of .09 lb/sec (.041 kg/sec) to offset the boil-off rate. The cooled LH₂ is returned to the vehicle by the transfer pump, at a rate equal to the withdrawal rate. Estimated cost for the complete cooling unit is \$175,000.00.

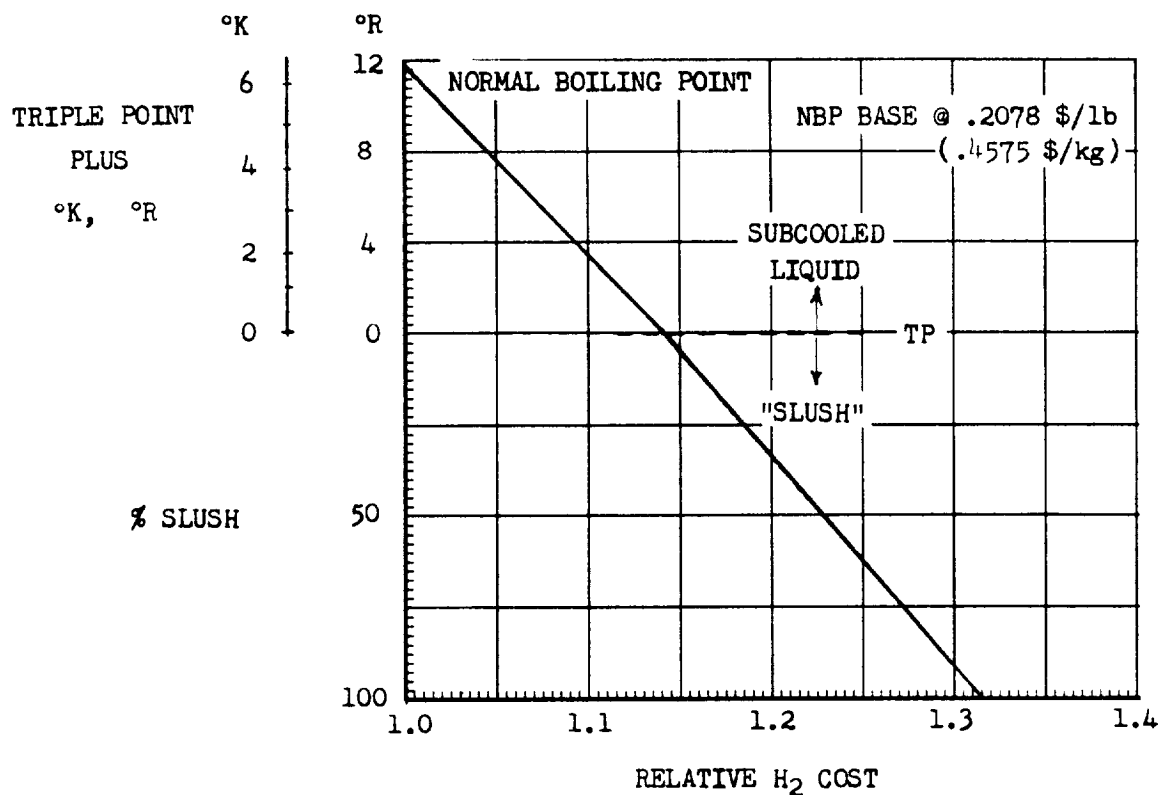
(b) (U) Logistics and Supply - Low pressure boil-off can also be used to subcool the hydrogen in the ground storage tankage at the using facility. The pumping capacity of the vacuum pump used in the ground hold unit above will subcool one load of LH₂ for the air launched rocket aircraft from NBP to 30°R (16.7°K) in 1.7 hours. Increasing the subcooled quantity will result in a proportionate increase in subcooling time. These times are not operationally limiting, however, since the subcooled fuel can be stored in separate tankage and used as required. Typical storage times are shown in Figure 4-103 for current state of the art ground storage systems. Values are presented for two different heat fluxes (which scale to a loss of 1/2 to 1% per day) and for the times to reach Triple Point (TP) and Normal Boiling Point (NBP).

(U) FIGURE 4-103
EFFECT OF HEAT LEAK ON SUBCOOLED H₂ STORAGE



(c) (U) Cost - The impact of subcooling on the cost of acquisition of a hydrogen system can be related directly to the amount of gas evolved during the low pressure boil-off or vacuum pumping technique. Relative costs are shown in Figure 4-104 for various degrees of subcooling and percentages of slush hydrogen. NBP liquid is used as the base value. If subcooled/slush hydrogen is produced in conjunction with the manufacture of liquid, only minor modifications to the LH₂ production plant are required. Two methods have been investigated which can supply 50% slush hydrogen in quantities of 20 tons/day. (18.2 metric ton/day) Air Reduction Company (AIRCO) has synthesized a unique direct refrigeration method in which the hydrogen is subcooled at atmospheric pressure, utilizing helium refrigeration. Estimated cost per pound of 50% slush is 4.7¢ (10.35¢/kg), in addition to the base cost of the LH₂. Details of this system are the proprietary information of AIRCO.

(U) FIGURE 4-104
COST OF SUBCOOLING



(U) An alternate method for producing slush hydrogen is by direct vacuum pumping, as noted above. If a vacuum pumping plant were installed in conjunction with a liquid hydrogen plant, it would be relatively easy to reclaim the boil-off hydrogen gas. The estimated cost increment for 50% slush [for a 20 ton day plant (18.2 metric ton/day)] is 4.5¢/lb (9.91¢/kg). The latter includes a gaseous hydrogen reclamation cost of 8.15¢/lb reclaimed (17.93¢/kg). Cost summaries are presented in Figure 4-105 and include 5 year depreciation of capital investment and annual operating costs for a 20 ton/day plant.

(U) FIGURE 4-105
SLUSH HYDROGEN PRODUCTION COSTS
(20 Ton/Day (18.2 Metric Ton/Day) - 50% Quality)

	AIRCO	VACUUM PUMPING
CAPITAL	\$1.730 M	\$1.464 M
ANNUAL OPERATING		
HELIUM	\$.010 M	-
POWER	\$.330 M	\$.090 M
GH ₂ RECLAIM		
TOTAL	\$.340 M	\$.364 M
5 YEAR OPERATING + CAPITAL	\$3,430,000	\$3,284,000

4.4.3.2.3 (U) Comparative Performance Increment - The aircraft performance benefits of subcooling, reflected in the OWE reduction pointed out earlier, can be compared with the ground performance increment on the basis of cost. Figure 4-106 reflects the differences in aircraft costs (RDT&E plus investment and operating) and isolates the effected ground system (special AGE and propellant costs) for a horizontal take-off airbreathing (TJ/CSJ) Mach 12 flight research vehicle. On a program basis, for 200 flights, RDT&E and investment costs for the vehicle using subcooled fuel are \$20 M lower than for the vehicle using NBP fuel, while the overall operating costs are reduced by \$14 M. These are additive, resulting in an overall program savings of \$34 M, which includes an increase in AGE of \$1.46 M and an increase in propellant costs of only \$75,000 over the life of the program.

(U) FIGURE 4-106
SUBCOOLED FUEL PROGRAM COSTS
(HTO Airbreather: 5 Minute Cruise at Mach 12)

	NBP (37°R)	Subcooled (30°R)
RDT&E + Investment (Millions)		
Baseline [function of (OWE)]	\$903.	\$883.
Δ Special Handling AGE	-	+1.4
Operating (Millions)		
Baseline [function of (OWE)]	\$397.	\$383.
Δ Propellant Costs	-	+.075
NET RESULT		-\$34.

(U) From the foregoing discussion it is readily seen that the use of sub-cooled hydrogen can provide significant airborne performance gains. These translate to an equally significant reduction in flight research vehicle program costs. The use of subcooled hydrogen is operationally attractive because of its relatively lower tendency to boil than saturated liquid. Since subcooled and slush hydrogen has been produced in an inexpensive and easy manner (vacuum pumping) and pumping/flow transfer of slush qualities up to 50% have been demonstrated, the techniques discussed here are consistent with the present level of hydrogen technology.

(U) Conclusion - The use of subcooled hydrogen is feasible and economical. Subcooled hydrogen provided significant performance gains without an attendant penalty for either airborne or ground systems, and was chosen for design of all remaining HYFAC vehicles.

4.4.4 (U) AIRFRAME STRUCTURE - The selection of design load factors, fuel tankage concept, and thermal protection system has a major influence on the structure design, vehicle performance, weights and cost. The effects are quantified and discussed in the paragraphs which follow. For each variation studied, the mission performance of the flight vehicle under evaluation was maintained constant.

4.4.4.1 (U) Load Factor - Design load factor affects baseline airframe air loads and inertia loads, external surface equilibrium temperatures, and engine inlet pressures. The effect of load factor on external surface equilibrium temperatures and inlet pressures at Mach 6 and Mach 12 is illustrated on Figures 4-107 and 4-108, respectively. These temperatures and pressures have been calculated on the basis of a 20° angle of attack and the corresponding altitude at which a given load factor can be achieved. It is expected that a 20° angle of attack will provide the capability to perform all the required maneuvers without exceeding the aircraft stability limits.

(U) Parametric studies were performed on the B212A Mach 6 TJ/RJ powered aircraft, B233A Mach 12 rocket powered aircraft, and the B257F Mach 12 TJ/CSJ aircraft to determine the effect of load factor on the aircraft size, cost, and performance. Load factors ranging from 3.5 to 7.3 g were considered. The lower boundary (3.5 g) was selected as being representative of that considered appropriate for operational hypersonic aircraft and the upper boundary (7.3 g) is consistent with the initial X-15 design as well as current operational fighter aircraft designs. The major elements of the aircraft that are affected by load factor are inlet structure, airframe structure and thermal protection system (insulation and shingle). First, the effect of load factor on the individual elements is considered. Then, the combined effect on the total aircraft size, performance and cost is shown.

(U) The inlet structure is primarily a non-circular pressure vessel and, therefore, its weight is quite sensitive to pressure increases. The weight of the pressure vessel segment of the inlet varies with pressure as indicated by the following relationship which is based on statistical analysis:

$$\frac{W_1}{W_2} = \left(\frac{P_1}{P_2} \right)^{.6}$$

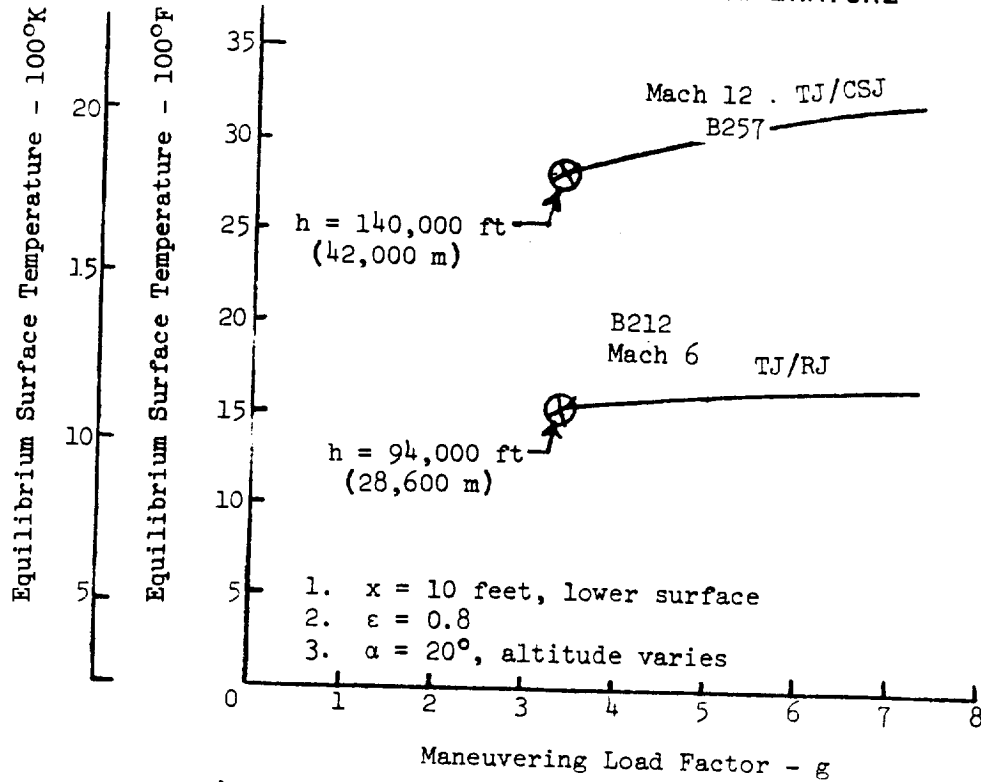
where W = weight

P = pressure

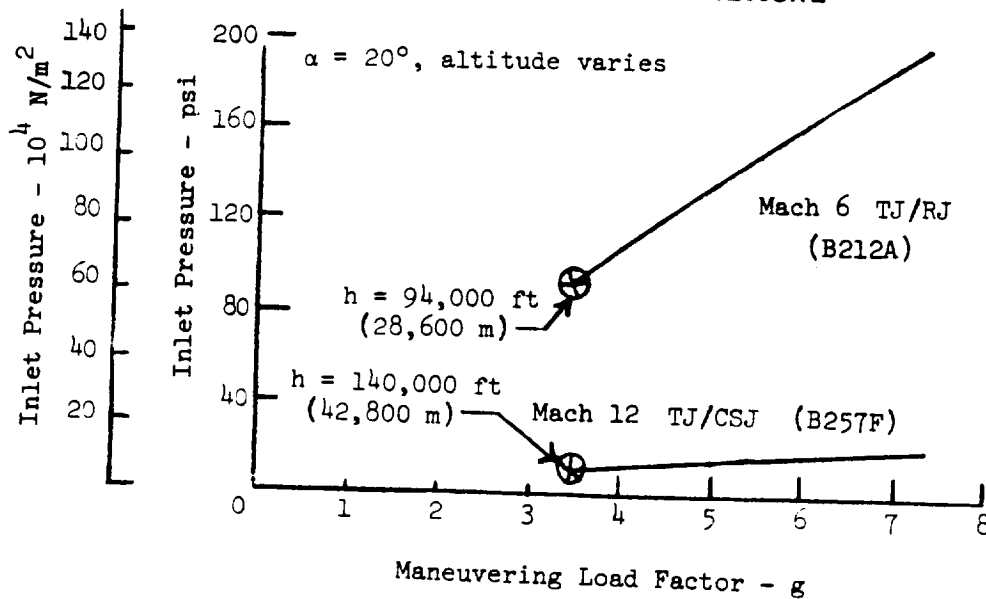
(U) Sensitivity of airframe weight to load factor is a function of aircraft configuration, among other things. Wing-body aircraft structure weight is more sensitive to load factor than that of the all-body configurations. Figure 4-109 illustrates the effect of increasing design load factor on the primary structure weight of B212, a Mach 6 wing-body aircraft. The indicated relative weight increase for higher load factors is due entirely to the associated increase in loads in the primary structure. Constant structural temperature was assumed in this comparison which would require an increase in the thermal protection system to maintain the temperature at a higher load factor.

(U) The increased surface temperature resulting from increased load factor requires that radiation shingles fabricated of materials with higher temperature capability be used over a significant area of the vehicle. This results in heavier and

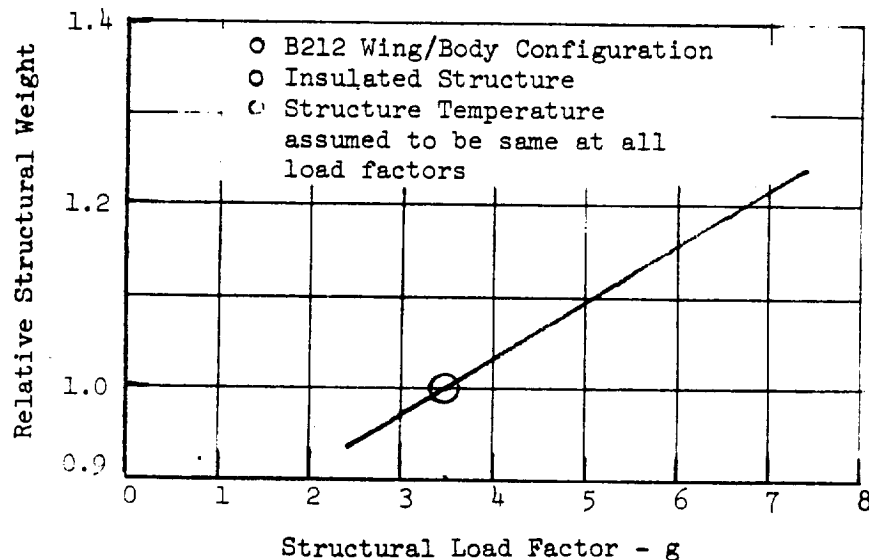
(U) FIGURE 4-107
LOAD FACTOR SENSITIVITY SURFACE TEMPERATURE



(U) FIGURE 4-108
LOAD FACTOR SENSITIVITY INLET PRESSURE



(U) FIGURE 4-109
LOAD FACTOR SENSITIVITY STRUCTURAL WEIGHT

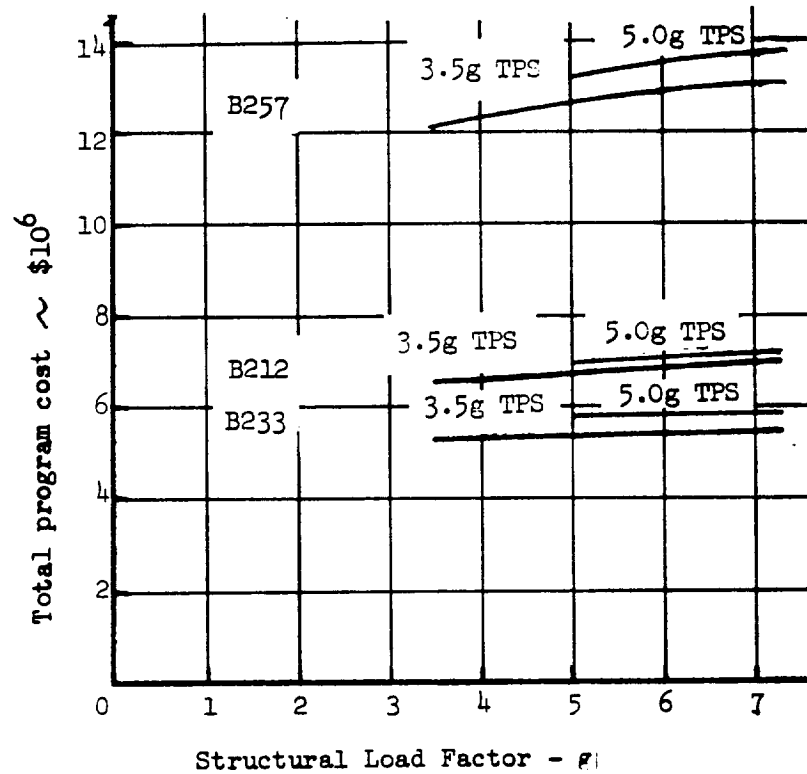
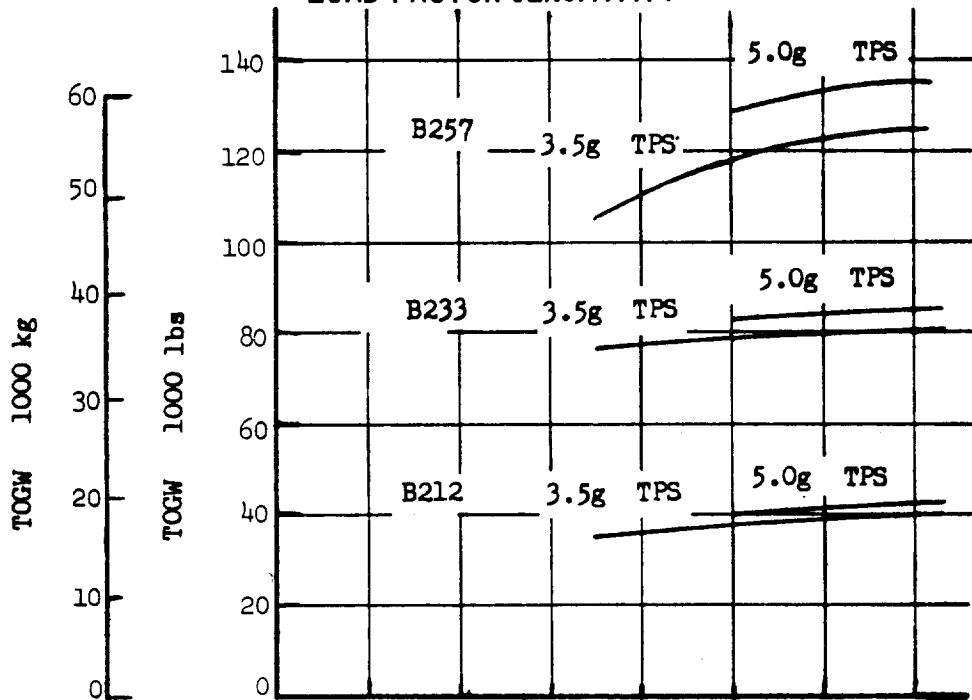


more expensive shingles in most cases. For example: the Mach 12 aircraft would require that about 10% of the surface be protected with coated tantalum alloy shingles if designed to 5.0 g maneuver, whereas a 3.5 g maneuver would require no tantalum. The insulation thickness itself is affected only slightly by the higher load factor maneuvers, since the time at temperatures is very short relative to the mission time; however, the selection of the insulation material would vary to be compatible with the operating temperature.

(U) When all of the effects of increased load factor are considered, the aircraft would have to be resized to retain constant performance (5 minute test time). Thus, increasing design load factor results in increased aircraft size, weight, and cost. Weight and cost variations of the specific aircraft selected for comparison are shown in Figure 4-110. Other factors such as growth, versatility, reliability, and research value must be considered before design load factor is established, however, the significance of these factors can only be judged on a qualitative basis. Consideration of cost and weight would suggest the selection of a low-load factor aircraft. However, consideration of versatility and growth capability points toward higher load factors. The research value, as associated with the operational aircraft, is essentially independent of design load factor due to the low design load factors (3.5 g) of the operational aircraft.

(U) A design load factor of 7.3 g's yield temperatures for the Mach 12 aircraft beyond the limits of current heat shield materials and one of 5.0 g's yields temperatures beyond the upper limit of coated columbium alloys on the forward section of the aircraft lower surface. Structure design to 7.3 g's is highly desirable

(U) FIGURE 4-110
LOAD FACTOR SENSITIVITY



from a structural redesign consideration; however, 5.0 g's is sufficient to provide the required research at a satisfactory safety level. A design providing 5.0 g's structural capability, in conjunction with the temperature restrictions associated with a 3.5 g maneuver at V_{max} , results in a favorable combination of minimum cost, weight and risk with maximum versatility growth potential.

4.4.4.2 (U) Tankage - The relative merits of integral and non-integral fuel tanks for hydrogen fueled aircraft have been the subject of many independent studies. The type of tankage affects the thermal protection systems, fuel volume fraction, structural weight and cost. A comparison has been made of integral and non-integral tankage for research vehicle B212A, a Mach 6 hydrogen fueled aircraft. The following combinations of tankage/primary structure concepts were considered as a basis for equal mission performance variations of the basic design:

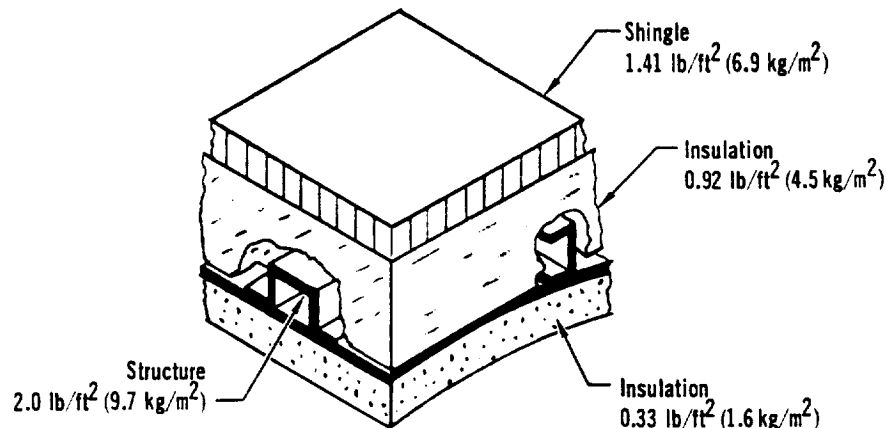
- o Integral tanks with insulated primary structure
- o Non-integral tanks with insulated primary structure
- o Non-integral tanks with hot primary structure.

(U) The integral tank system is employed on B212A and utilizes internal insulation to protect the primary structure from liquid hydrogen temperatures. This is consistent with minimum risk philosophy concerning structural design since the thermal gradient across the load bearing structure is minimized. Liquification of air on the outside surface of the tank (cryodepositing of air) can be eliminated by tailoring the internal insulation so that it will maintain the tank wall at a temperature higher than -320°F (77.4°K), the boiling point of air. Careful attention must be paid to openings in the tank wall so that heat shorts are eliminated. In addition, a barrier must be employed to prevent infiltration of the insulation by the cryogen as this would degrade the insulation. Techniques employed on the S4-B for internal cryogenic insulation would serve as examples for the solution of these problems.

(U) The non-integral tank system would not require internal insulation to retain the "minimum risk" status because the primary airframe loads are not carried in the -430°F (16.65°K) tank wall. Some of the tank volume lost to the required expansion gap between the tank and airframe structure can be recovered by placing the cryogenic insulation outside the tank in the expansion gap. Placing the cryogenic insulation on the outside of the non-integral tank eliminates the requirement of developing a vapor barrier to keep hydrogen out of the insulation. However, air must be prevented from leaking in through the insulation where it would become liquified. Once liquified and reduced in volume, a partial vacuum would exist and more air would be drawn in through the leak and become liquified (cryopumping of air).

(U) The results of the trade study show that aircraft size, weight, and cost are reduced by using an integral tank system. Preliminary analysis resulted in relative values of weight and thickness, as shown in Figure 4-111, which are reflected in the total aircraft weight and cost. The hot structure concept was rejected because of high weight and cost. Hot structure is also limited to speeds below Mach 6 at the design load factors, above this, the temperatures in the structure would exceed the design limit. Selection of internal insulation, though not required in the non-integral tank, improves the design in that cryopumping can be eliminated and heat shorts to the fuel minimized. On the basis of aircraft size, weight, and

(U) FIGURE 4-111
TANKAGE COMPARISON

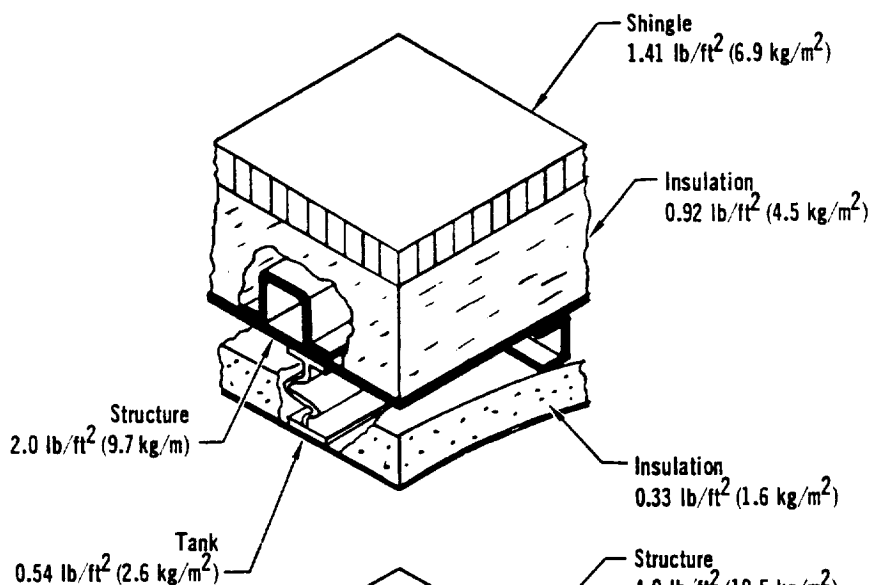


Wach 6
LH₂ Fuel
Surface Temp 1300°F
977°K
Heat Leak 100 btu/ft²hr
31.5 watts/m²

Baseline

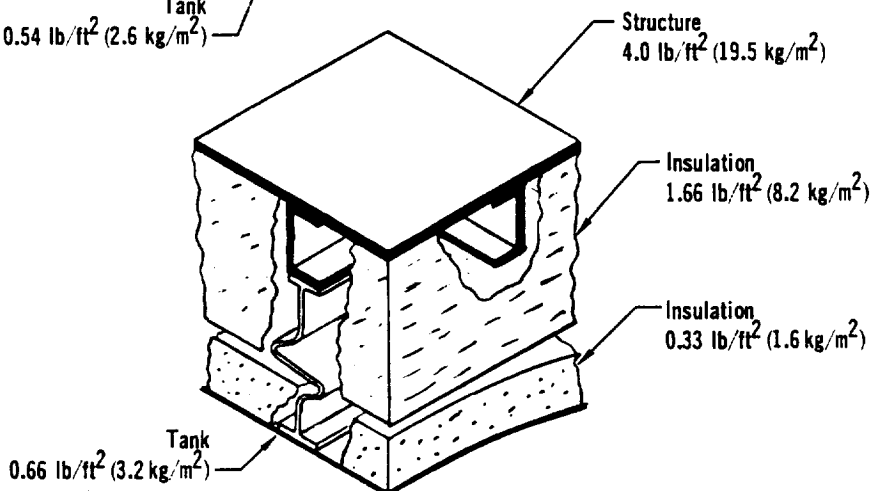
**Insulated Structure
Integral Tank**

Unit Weight 4.66 lb/ft² (22.7 kg/m²)
Thickness 1.9 inch (4.83 cm)
OWE 32,540 lb (14,750 kg)
TOGW 37,223 lb (16,900 kg)
Total Program Cost \$665 (10⁶)



**Insulated Structure
Non-Integral Tank**

Unit Weight 5.20 lb/ft² (25.3 kg/cm²)
Thickness 3.0 inch (7.62 cm)
OWE 33,900 lb (15,400 kg)
TOGW 38,910 lb (17,650 kg)
Total Program Cost \$676 (10⁶)



**Uninsulated Structure
Non-Integral Tank**

Unit Weight 6.65 lb/ft² (32.5 kg/m²)
Thickness 4.3 inch (10.9 cm)
OWE 38,650 lb (17,500 kg)
TOGW 43,543 lb (19,750 kg)
Total Program Cost \$712 (10⁶)

cost, the integral tank concept with insulated structure and internal tank insulation was found to be superior.

(U) Volume fraction (i.e., fuel volume/fuselage volume) is poorer for the non-integral tank concept than for the integral tank, since there must be a gap between the tank and the structure to allow for expansion due to differential temperatures between the warm structure and cold tank wall. The hot structure, non-integral tank has the poorest volume fraction (lowest) because it requires a greater expansion gap (due to the higher structural temperature) and also a greater insulation thickness (due to a less efficient heat sink arrangement).

(U) With an insulated structure approach, the total aircraft insulation weight is the same with either integral or non-integral tanks if internal tank insulation is employed. Without internal insulation, the integral tank concept would require a greater amount of insulation between the shingle and the tank structure to maintain the same heat input as the non-integral tank. This is due to the large number of heat shorts (shingle supports) as compared to the non-integral tank which has direct heat paths only at the widely spaced tank supports. Insulation requirements to protect the fuel and equipment bays are greater for the hot structure concept than for either type of insulated structure. This is because the mass (structure) used as a heat sink behind the insulation is greatly reduced when the structure is adjacent to the moldline surface.

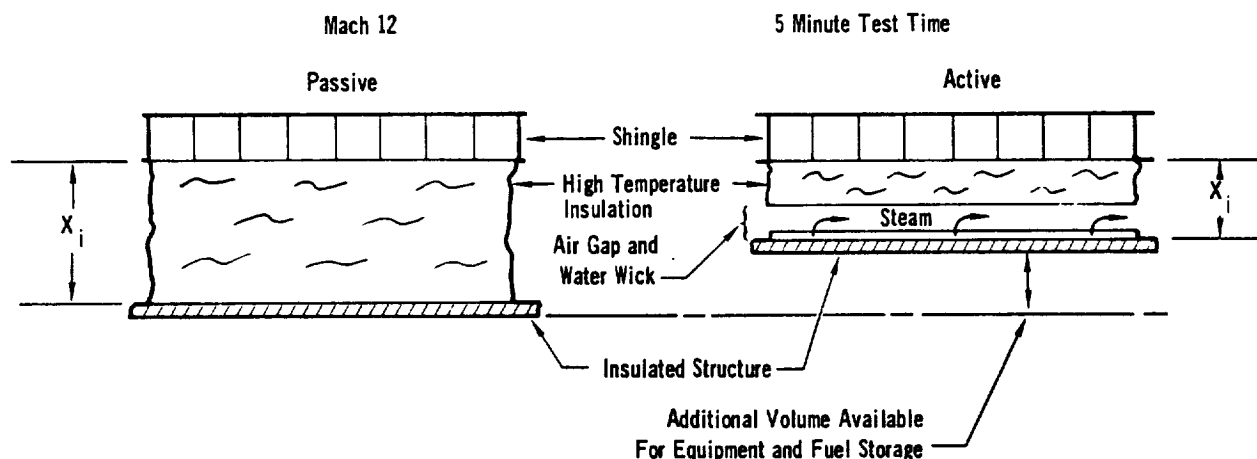
4.4.4.3 (U) Active Thermal Protection System - Past studies of operational, hydrogen fueled, cruise vehicles have shown that the reduction in weight and thickness that is possible with an active (water wick) thermal protection system has a significant influence on vehicle size and performance. For example, range could be improved as much as 10% by reconfiguring a hydrogen fueled, scramjet powered cruise vehicle with an active system.

(U) To establish what impact thermal protection has on the size, weight, and cost of a research aircraft, a tradeoff of active versus passive external (between shingle and structure) insulation concepts was performed. The B233A aircraft (Mach 12, all body, all rocket, air-launch, 5 minute test time) with an insulated load bearing structure/integral tankage arrangement was used as the basis for this study. However, the results are considered applicable to all test vehicles of this general Mach number and test time class. Shingle, structure/tankage, and internal cryogenic insulation requirements are nearly independent of the external insulation concept employed and hence were not included in the tradeoff.

(U) A comparison of weight, thickness, and volumetric efficiency of the active and passive systems is presented in Figure 4-112. As noted in this figure, unit weights and thicknesses are average values for the vehicle. That is, an average of actual external insulation (between shingle and structure) requirements at four representative vehicle locations (upper and lower surface for a typical fuel and non-fuel area) was determined. Tabulated values include the weight and/or thickness of high temperature insulation, air gap (0.25 inch, 0.635 cm) water, and wick. Since internal cryogenic foam insulation is required only in fuel tank areas, its weight is not included in the average unit weights and thicknesses presented in Figure 4-112. The size of the active system was based upon steady state conditions (neglects heat storage effects) and the equivalent step-input of surface temperature discussed previously in Section 4.2.3. Neglecting heat storage effects is conservative in determining active insulation requirements. In sizing the passive system, heat storage (transient) effects were accounted for per the procedure of Section 4.2.3.

MCDONNELL AIRCRAFT

(U) FIGURE 4-112
COMPARISON OF THERMAL PROTECTION CONCEPTS



Thermal Protection Concept	Unit (a) Weight		Unit (a) Thickness, X_i		Volumetric Efficiency (b)
	(PSF)	(Kg/m ²)	(In.)	(cm)	(%)
Passive	1.75	8.52	1.53	3.89	90.0
Active	0.55	2.69	0.78	1.98	92.6

(a) External Insulation (X_i) Requirements; Average Values for Mach 12 Aircraft.

(b) LH₂ Tankage Areas; Includes 0.6 Inch (1.52 cm) of Internal Cryogenic Foam Insulation.

(U) As shown in Figure 4-112, the active system reduces average unit external insulation weights by 1.2 psf (5.83 kg/m²) or 68%, and unit thicknesses by 0.75 in (1.91 cm), or 49%. The reduction in overall thickness requirements (total thickness between external moldline and fuel) results in a 2.6% increase in the volume available for LH₂ storage. A similar increase is likewise available for equipment storage in non-fuel areas.

(U) The effects of the thermal protection concept on vehicle size, weight, and cost are presented in Figure 4-113. Reconfiguring this vehicle with an active (water wick) system, reduced overall thermal protection weights by about 2800 lb (1268 kg). This large weight saving in conjunction with the increased volume available (2.6%) for equipment and fuel storage, resulted in a 9.5% decrease in vehicle planform area and a corresponding decrease in OWE (15%) and TOGW (13.8%). The 9.5% reduction in planform area is of particular importance for an airlaunch vehicle such as the B233, because of physical limitations associated with hanging it from a wing pylon on the C-5A. As noted in Figure 4-113, the overall reductions in vehicle size and weight translate directly into a program cost saving of over \$38 million or 7%.

(U) FIGURE 4-113
EFFECT OF THERMAL PROTECTION CONCEPT ON VEHICLE SIZE AND COST
(Mach 12; 5 Minute Test Time; Equal Research Value)

Parameter	Passive (B233A)	Active (B233A7-1)	Improvement With Active System
◦ Planform Area, Ft ² (M ²)	929 (86)	841 (78.2)	9.5%
◦ OWE, Lbs. (Kg)	24,600 (11,159)	20,910 (9460)	15%
◦ TOGW, Lbs. (Kg)	77,950 (35,358)	67,210 (30,486)	13.8%
◦ Cost (10 ⁶ Dollars)			
◦ RDT & E	242.863	224.163	7.7%
◦ Investment	123.859	114.820	7.3%
◦ Operative	178.077	167.463	6%
◦ Total Program	544.799	506.446	7%

(U) The active TPS, water wick, used in this study capitalizes on the integrated performance of shingles which re-radiate approximately 98% of the total aerodynamic heat load, high temperature insulation, water evaporation (which absorbs nearly all of the heat transferred to the interior of the vehicle), and an efficient cryogenic insulation (fuel areas only), to obtain low unit weights while retaining high volumetric efficiencies. This system takes advantage of the high heat of vaporization of water (about 1000 Btu/lb, 2324.4 joules/gm) as a heat sink to reduce insulation requirements.

(U) To aid in understanding the above, an example is presented in Figure 4-114. With a passive system, 2 inches (5.08 cm) of high temperature insulation (1.33 lb/ft², 0.6 kg/m²) is required to limit the heat transfer across the insulation to 200 Btu/ft² hr (360 watts/m²) such that a maximum structural temperature of 300°F (422°K) is not exceeded. With the active system, high temperature insulation requirements can be reduced to 0.55 inch, 1.4 cm (an additional 0.4 inch, 1.02 cm, of thickness is required for air gap and water wick) with a corresponding increase in the heat transfer to the interior of the aircraft (730 Btu/ft² hr, 2300 watts/m²) which is absorbed by the conversion of water to steam which is expelled overboard. As indicated in Figure 4-114, total active systems weight including insulation, water and wick is 0.74 lb/ft² (0.336 kg/m²). It should also be noted that structural temperatures, when protected with an active water wick system, are maintained at approximately 100°F (311°K) which corresponds to the boiling point of water at the reduced pressures experienced at the cruise altitude.

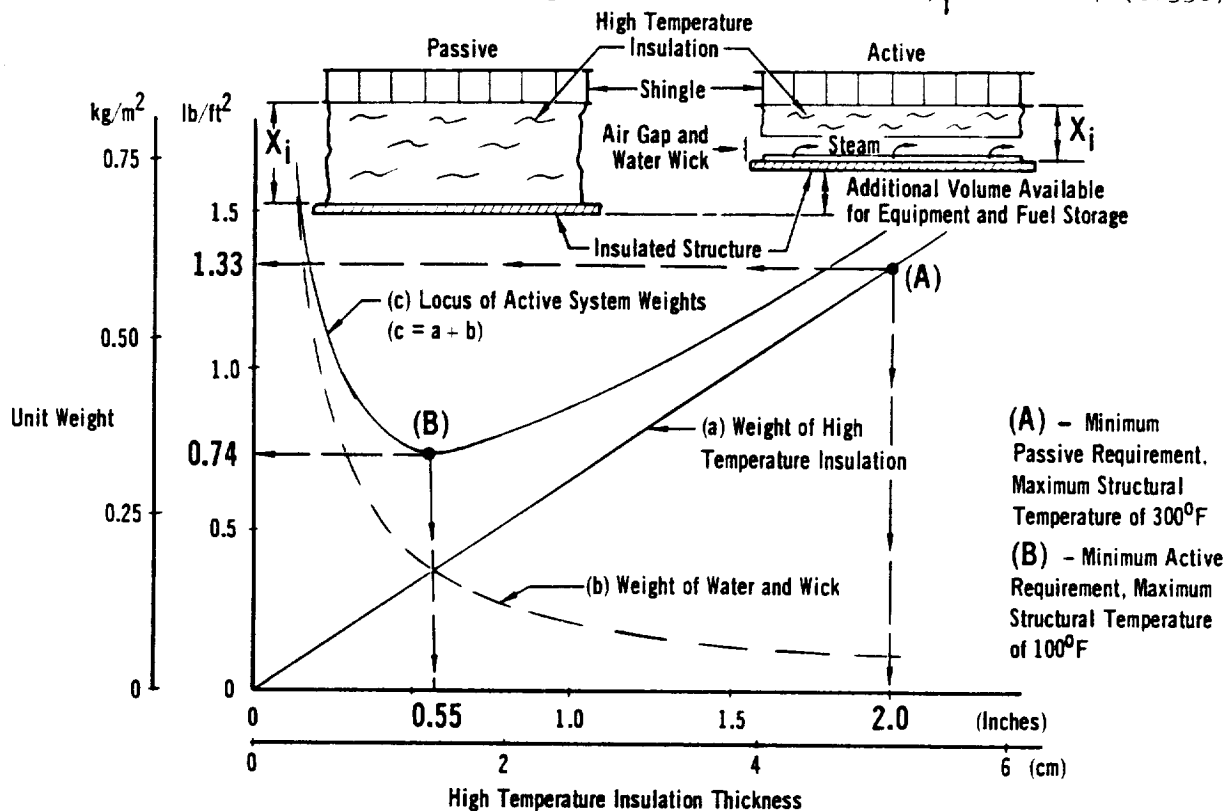
(U) FIGURE 4-114
INCREASED EFFICIENCY WITH ACTIVE TPS

ANALYSIS CONDITIONS:

- o Steady State Analysis of Non-Fuel Area
- o Heat Pulse = 1800°F (1260°K) for 30 Minutes

SOLUTION:

	(A) Passive	(B) Active
o Maximum Structural Temperature; °F, (°K)	300(422)	100(311)
o Heat Transfer Across High Temperature Insulation; Btu/ft ² hr, (Watts/m ²)	200(630)	730(2300)
o Thickness; in., (cm)		
(a) High Temperature Insulation	2.0(5.08)	0.55(1.40)
(b) Air Gap	---	0.25(0.63)
(c) Water Wick	---	0.15(0.38)
Total Thickness (X_i)	2.0(5.08)	0.95(2.41)
o Unit Weight; lb/ft ² , (kg/m ²)		
(a) High Temperature Insulation	1.33(0.6)	0.37(0.168)
(b) Air Gap	---	---
(c) Water and Wick	---	0.37(0.168)
Total Unit Weight	1.33(0.6)	0.74(0.336)



(U) Some of the advantages of passive and active systems are presented in Figure 4-115. An active system would require servicing between flights. For a research aircraft which has no requirements for rapid turnaround, reservicing requirements are of little consequence. For an operational system, it would be an important consideration. Previous studies indicate that the water wick system can be reserviced in about 20 min.

(U) Other advantages of an active system are:

- o Good growth potential - A comparison of passive and active TPS requirements as a function of test time and Mach number is presented in Figure 4-116.

- o Minimize effects of heat shorts; i.e., the amount of coolant can easily be increased locally to compensate for increased heat transfer.

- o Nearly constant-temperature structure - Since the active system uses a phase change process, structural temperatures and temperature gradients are limited to small excursions about some nominal value.

- o No structural cool down requirements at completion of mission - Due to the time lag associated with a passive system, maximum structural temperatures are generally experienced after the aircraft has landed. This then requires long waiting periods or forced cooling before structural temperatures are low enough to permit normal inspection, reservice or repair of internal areas. With an active system, ground operations can commence almost immediately.

(U) Except for subjective differences in technical risk, the above facts would indicate that an active system should be selected for the Mach 12 class of research aircraft. Assessing the difference in technical risk is, at best, difficult. However, the water wick principle has been demonstrated and performed successfully under rather severe conditions during the BGRV flight test programs. The BGRV system is identical with the water wick system discussed herein, with one exception; it used a water gel, which is a "one shot" concept, whereas the water wick is reserviced after each flight. It can be argued that any active system is inherently less reliable. On the other hand, an active system is more desirable because its capacity can readily be increased, with little penalty to aircraft performance, to meet unexpected new demands. It is MCAIR's contention that with proper sensors to monitor its performance, an active system does not present an undue risk and hence substantial cost savings should be realized by using an active system on the Mach 12 vehicles.

(U) A cursory analysis indicated that the advantages of reconfiguring the Mach 6 vehicles with an active TPS would be less pronounced (approximately a 3% reduction in S_p , OWE and TOGW) than observed for the Mach 12 vehicles (Figures 4-112 and 4-113). Even so, an active system was selected for the Mach 6 vehicles to initiate the Phase III Study.

(U) Two Research Objectives, R.O. 28 and 43, deal specifically with the development of reusable thermal protection systems, for cryogenic fuels and oxidizers, and the primary structure, respectively. A ranking by the scientific community (Figure 4-117) indicates the importance of conducting the necessary research to satisfy these objectives. Although much of this research can be and would be

(U) FIGURE 4-115
REASONS FOR SELECTION OF ACTIVE THERMAL PROTECTION SYSTEM

PASSIVE

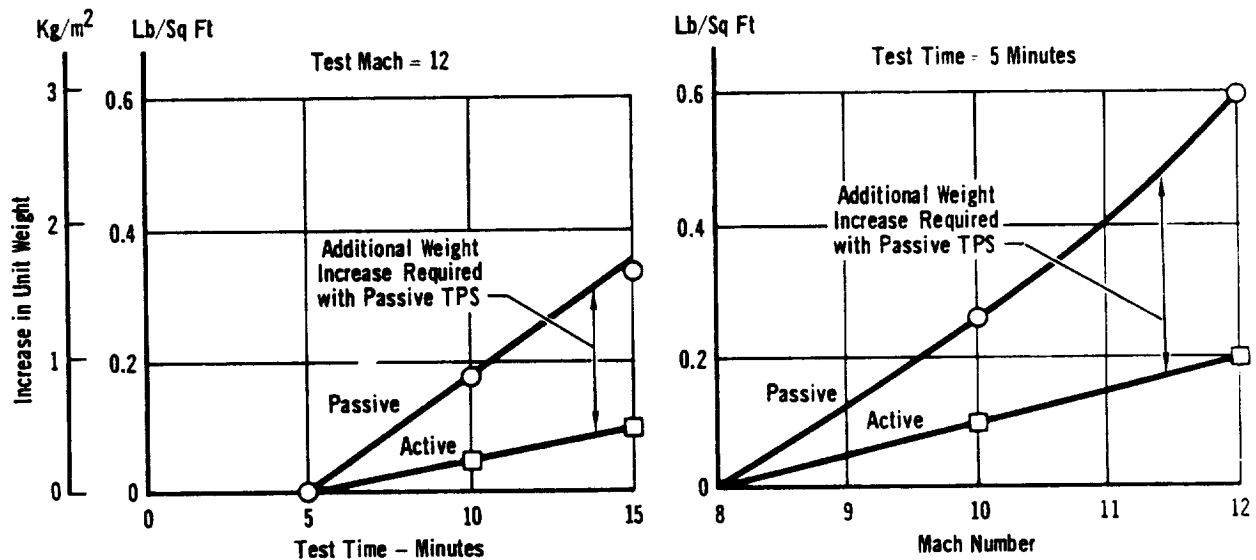
- Conventional Approach
- No Reservice

ACTIVE

- New but demonstrated concept available (Water Wick)
- Reservicing times compatible with normal turn-around times for operational vehicles
- Minimizes effects of heat shorts
- Near constant structural temperatures
- No structural cool down required prior to inspection and/or repair of internal areas

SELECT ACTIVE SYSTEM

(U) FIGURE 4-116
GROWTH POTENTIAL OF ACTIVE THERMAL PROTECTION SYSTEM



accomplished in ground facilities, final development and/or demonstration will require flight test. During Phase III detailed studies will be conducted to show methods by which the research vehicle can be used as a test bed for conducting research on the thermal protection concepts of interest.

(U) FIGURE 4-117
IMPORTANCE OF DEVELOPING REUSABLE
THERMAL PROTECTION SYSTEMS

Operational System	Rank of R.O. #28	Rank of R.O. #43
L ₁ } Launch	1st out of 67	4th out of 67
L ₂ }	1st out of 66	4th out of 66
L ₃ }	2nd out of 62	4th out of 62
L ₄ }	2nd out of 66	5th out of 66
C ₁ } Commercial	2nd out of 64	5th out of 64
C ₂ }	2nd out of 62	5th out of 62
M ₁ } Military	Not Applicable	3rd out of 52
M ₂ }	2nd out of 63	6th out of 63
M ₃ }	2nd out of 63	6th out of 63

R.O. #28 - Develop efficient reusable thermal protection systems for cryogenic fuels and oxidizer tankage.

R.O. #43 - Develop reusable thermal protection systems for the primary structure.

(U) Conclusion - Vehicle size and weight and hence total program costs were significantly reduced with the selection of an active TPS for the Mach 12 research aircraft. At Mach 6, the effects of an active versus a passive TPS on aircraft size and performance were less pronounced. A ranking of Research Objectives by the scientific community clearly indicated the importance of conducting research in the area of reusable TPS.

4.4.5 (U) RESEARCH CAPABILITY - Test Mach number, test duration, and payload envelope of the flight research vehicles were varied to determine their impact on overall research vehicle design and performance capability (research value). The effects on the vehicle cost and research value were then compared.

4.4.5.1 (U) Test Mach Number Sensitivity - Mach number tradeoffs were conducted for four of the Phase II configurations: the Mach 6 air launched rocket/ramjet vehicle (Configuration B207), the Mach 12 air launched rocket vehicle (Configuration B233), the Mach 12 horizontal takeoff rocket vehicle (Configuration B260), and the Mach 12 horizontal takeoff turbojet/convertible scramjet vehicle (Configuration B257). Design studies were used to define the Mach effects on each vehicle and the corresponding impact on program cost.

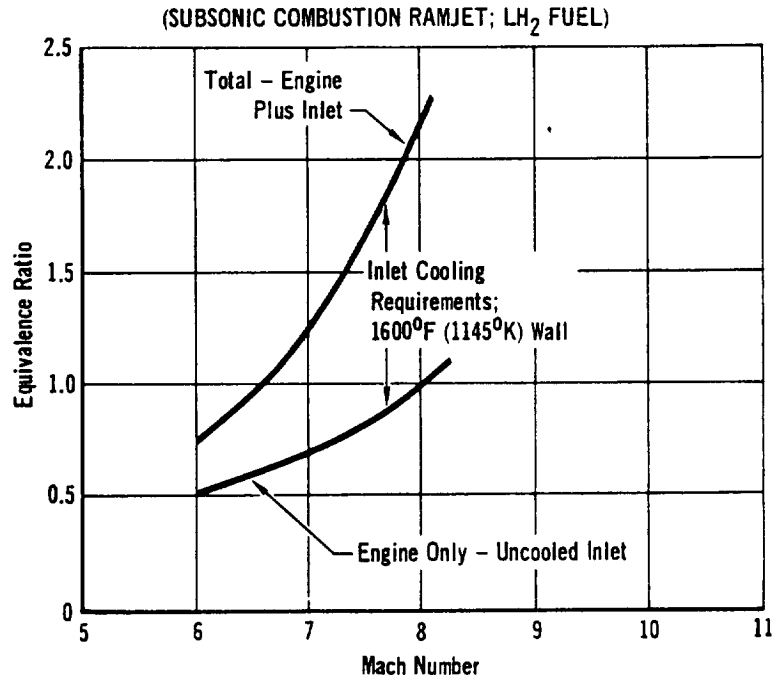
(U) The principal effect of increased Mach number was increased vehicle size to accommodate greater propellant requirements for higher speed flight and to compensate for the generally decreased Isp associated with higher Mach numbers on airbreathing vehicles.

(U) A clear example is depicted in Figures 4-118 and 4-119, which illustrate the impact of increased Mach number and associated higher stagnation temperatures on coolant flow requirements (and hence Isp) for the regeneratively cooled ramjet. Figure 4-118 presents the increase in cooling equivalence ratio (coolant fuel/stoichiometric fuel flow) (ϕ) required for engine and inlet cooling to maintain a structural wall temperature limit of 1600°F (1145°K). Figure 4-119 illustrates the effect of equivalence ratio on Isp at various Mach numbers. The minimum equivalence ratio for engine and inlet is superimposed to identify maximum Isp as a function of Mach number. For example, at Mach 6, cooling fuel requirements are low, equivalence ratio is less than 1, and ramjet Isp is high. At Mach 7, ϕ becomes greater than 1, and extra fuel is required just to cool the inlet and engine. As a result, the Isp at Mach 7 is only 2/3 of the value at Mach 6. The downward performance trend continues at an increasing rate with Mach number, illustrated by the extremely poor Isp at Mach 8.

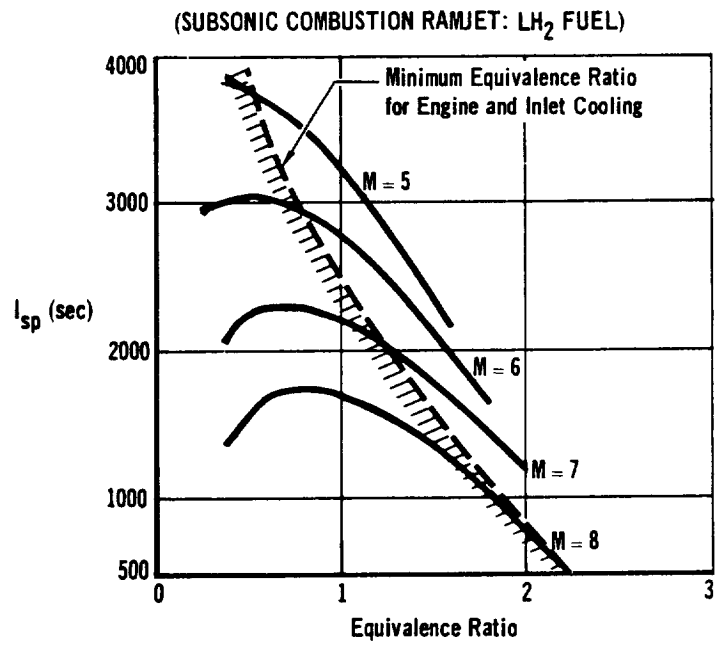
(U) Figure 4-120 shows the impact in terms of research vehicle program cost. The large increase in cost from Mach 6 to Mach 8 for the air launched rocket ramjet configuration (B207) is a result of a number of factors: 1) the increase in boost fuel required to reach the higher Mach number requires a larger physical size and, therefore, weight; 2) the more severe temperature environment at the higher Mach number requires heavier materials and insulation; 3) the airbreather engine inlets have a lower ram recovery at higher Mach numbers and the inlet cooling requirement above Mach 6.5 lowers the engine Isp. The increased cost for the Mach 12 convertible scramjet vehicle (B257) can be traced to factors similar to the three noted above. The Isp is slightly lower, but the primary factor is the boost fuel volume requirement which increases in proportion to the cruise altitude and Mach number. The more severe thermal environment at Mach 12 will also increase the vehicle's empty weight. For all rocket configurations, the cost increase is reflected in increased boost propellant volume and, therefore, vehicle size and weight.

(U) The increased research value of higher Mach numbers is illustrated in Figure 4-121, where the incremental increase is identified relative to a Mach 6 capability. The curve depicts an integrated assessment established by the HYFAC

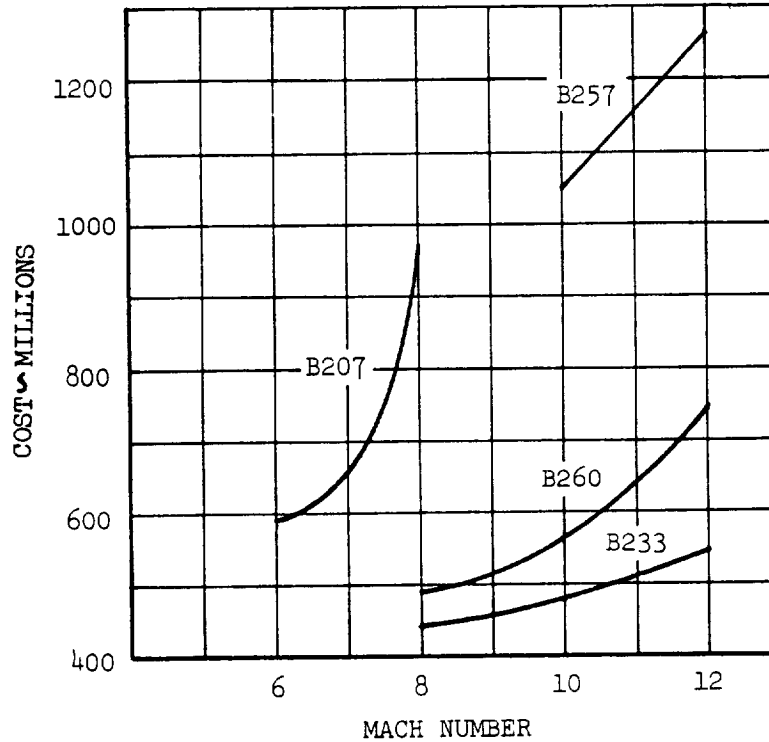
(U) FIGURE 4-118
INCREASE IN EQUIVALENCE RATIO WITH REGENERATIVELY
COOLED INLETS



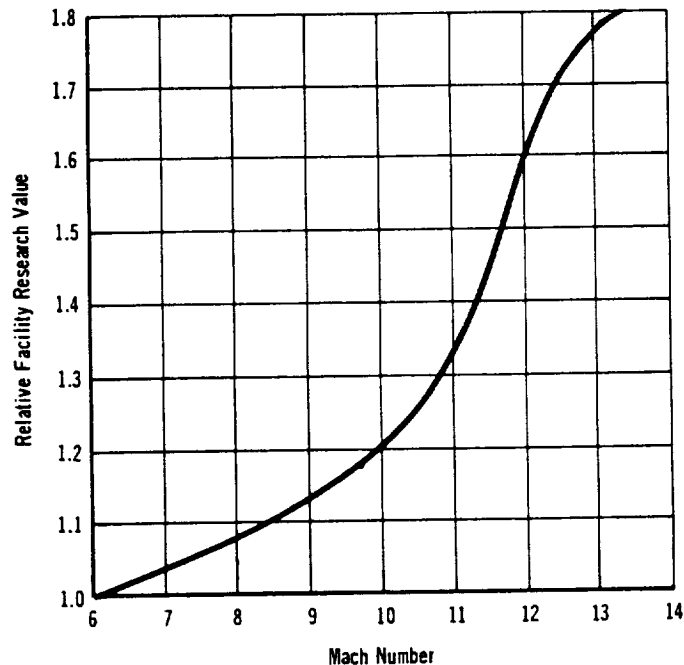
(C) FIGURE 4-119
DECREASE IN SPECIFIC IMPULSE DUE TO COOLING



(U) FIGURE 4-120
EFFECT OF MAXIMUM MACH NUMBER ON FLIGHT FACILITY COST



(U) FIGURE 4-121
EFFECT OF MAXIMUM MACH NUMBER ON FACILITY RESEARCH VALUE



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study team. The relatively modest improvement in research value between Mach 6 and Mach 8 reflects the moderate change in the influence of aerodynamic parameters within the speed regime. As flight speeds increase above Mach 8, however, thermodynamic effects also become more pronounced, contributing to the increasing value through the Mach 12 regime. The relatively large increase in research value in the Mach 12 region is achieved because this is the speed regime of proposed operational systems L₂, L₃, L₄, C₂, M₂ and M₃. Research in this Mach number environment is essential due to the increasing influence of the real gas effects as the flight speed approaches the regime of very thick hypersonic boundary layers and correspondingly thick laminar sublayers. The effects of these aerodynamic and thermodynamic conditions on the vehicle performance must be known before operational systems can be confidently developed. The increased research value at Mach 12 also reflects the use of a scramjet engine which leads to relatively high productivity in this region.

(U) Another significant factor at Mach 12 involves tradeoffs in fuel consumption vs cooling requirements. Mach 12 research in this area will permit development of a regenerative cooling system under realistic flow field conditions. Assessments of the thermal profile, and resulting refinements of the regenerative cooling system, can greatly improve engine performance. This relates directly to vehicle configuration/sizing requirements, which can influence the overall operating envelope.

(U) Research value continues to increase beyond the Mach 11-13 range reflecting the influence of aerodynamic and thermodynamic parameters as orbital velocities and altitudes are approached.

(U) Relative research value was assessed on the basis of cost and research value for the specific Mach 6 and Mach 12 flight research vehicles mentioned above. The research return for the investment in the air launched rocket/RJ Mach 6 vehicle (B207) drops off significantly when this vehicle is used at Mach 8. This decrease is caused by poor engine/cooling performance and a limited increase in research value. The horizontal takeoff Mach 12 rocket vehicle (B260) increases significantly in size between Mach 8 and Mach 12, but the research return varies only slightly.

(U) On the other hand, the airbreathing Mach 12 TJ/CSJ (B257) provides a pronounced increase in research value, when Mach number is increased from Mach 8, and thus provides a significant return per investment in vehicle size.

(U) The air launched rocket vehicle, (B233), because of its initially smaller size and lower investment base, is shown to be particularly attractive in terms of research value. The vehicle is capable of accomplishing research to Mach 12 with only a moderate increase in size and cost compared to the HTO vehicles.

(U) As a result of these comparisons, it is apparent that for the flight research concepts investigated, the maximum research return for the corresponding investment is at Mach 6 for the rocket/RJ vehicle (B207), Mach 12 for the air launched all rocket vehicle (B233), and at Mach 12 for the HTO TJ/CSJ vehicle (B257). For Mach 12 HTO all rocket vehicle (B260), the research return is fairly flat between $M = 8$ and $M = 12$, however, $M = 12$ would appear a more reasonable selection for an advanced research vehicle.

4.4.5.2 (U) Test Time - Determination of minimum test time was based on system flight stability data from the X-15 program and on the external temperature stabilization time for the shingles. Approximately 30 seconds was required to achieve system stability after pushover in the X-15; the remaining time can be considered as steady state. MCAIR studies have shown that temperature stabilization on the outer surface of a thin gage metal shingle occurs between 30 seconds to 60 seconds after the start of cruise flight at cruise altitude. Variation in surface temperatures for different materials will be minor if all shingles are of thin gage.

(U) In a 5 minute flight, the first 30 seconds would be to achieve system stability after pushover at cruise Mach number and altitude. The next 60 seconds would permit surface temperature stability of the shingles and would be valuable transient data to assist in determining heat transfer rates and aerodynamic heating rates on the surface of the aircraft. The last 3-1/2 minutes would be considered steady state data.

(U) In the case of a Mach 12 vehicle, the outer surface of the thin gage shingle will reach its maximum temperature or at least 95% of the maximum temperature within 1 minute after attaining cruise Mach number and altitude. Temperature variations, because of various materials used in shingle construction, will be insignificant if shingles are made of thin gage metal. A flight with 5 minutes at cruise Mach number and altitude will provide the same data acquisition as before: 30 seconds stabilization, 60 seconds of transients and 3-1/2 minutes of steady state data.

(U) Thermal protection system (TPS) requirements are a function of exposure time. Increased test time requires a corresponding increase in size and weight of the TPS, which then scales to vehicle size increases. Insulation thicknesses and weights were defined for Mach 6 and Mach 12 configurations as shown in Figure 4-122. No changes in shingle or insulation material concepts were necessary because maximum surface temperatures were not affected.

(U) FIGURE 4-122 PASSIVE TPS INSULATION REQUIREMENTS VARIATION WITH TEST TIME

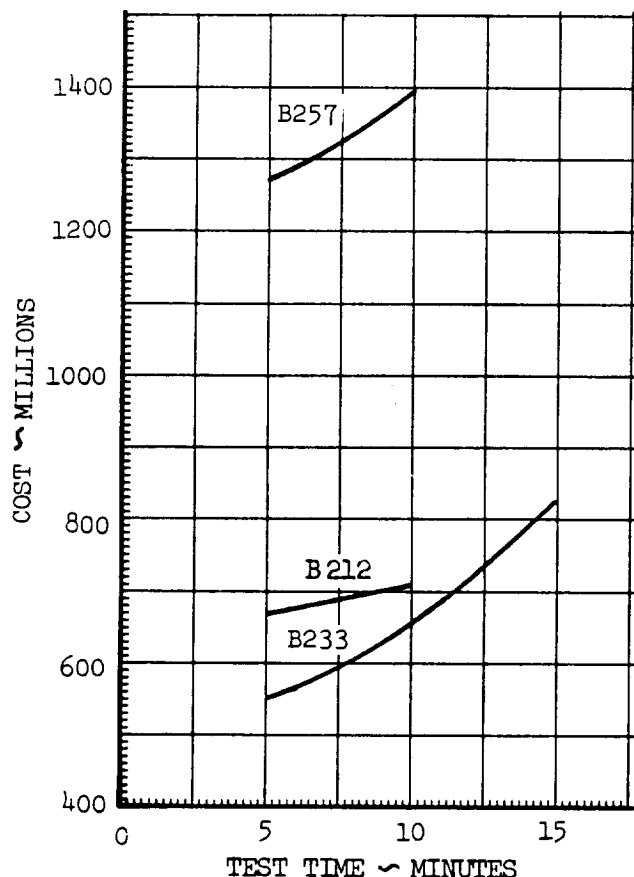
Test Time	Insulation Thickness		Insulation Weight	
	in	cm	psf	kg/m ²
Mach 6 Configuration				
5 Minutes (Baseline)	.69	1.75	.92	4.48
10 Minutes	.81	2.06	1.08	5.26
15 Minutes	.92	2.34	1.23	5.99
Mach 12 Configuration				
5 Minutes (Baseline)	1.54	3.91	1.75	8.52
10 Minutes	1.69	4.29	1.93	9.40
15 Minutes	1.84	4.67	2.09	10.18

(U) Costs of the increases in TPS with additional test time are shown in Figure 4-123 for the HTO Mach 12 airbreathing vehicle (B257), the Mach 12 air launched rocket vehicle (B233), and the airbreathing Mach 6 (B212) vehicle. These

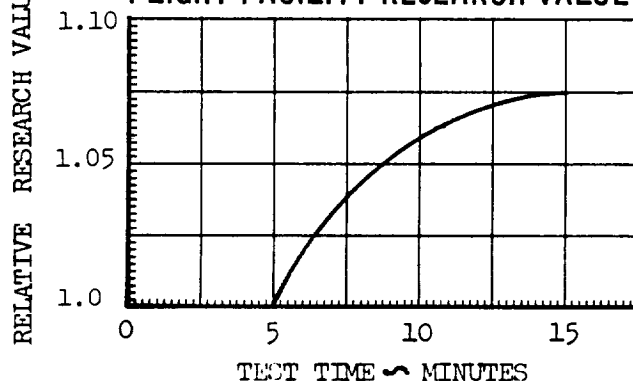
costs also reflect the obvious larger vehicle size required to satisfy new endurance limits. Cost increments are higher for the rocket vehicle due to the greater fuel volume required for cruise operation.

(U) The increase in research value with test time was assessed by the study team and is illustrated in Figure 4-124. As time is increased above 5 minutes a modest increase in research value is obtained, however, above 10 minutes the increase is very small. The research return relative to the required investment reveals that only the Mach 6 vehicle maintains a constant return for different test times. An additional consideration for the Mach 12 vehicles is the associated test range requirements for increase test time, as discussed in more detail in Section 4.4.6.

(U) FIGURE 4-123
EFFECT OF IMPROVED PASSIVE TPS ON
FLIGHT FACILITY COST

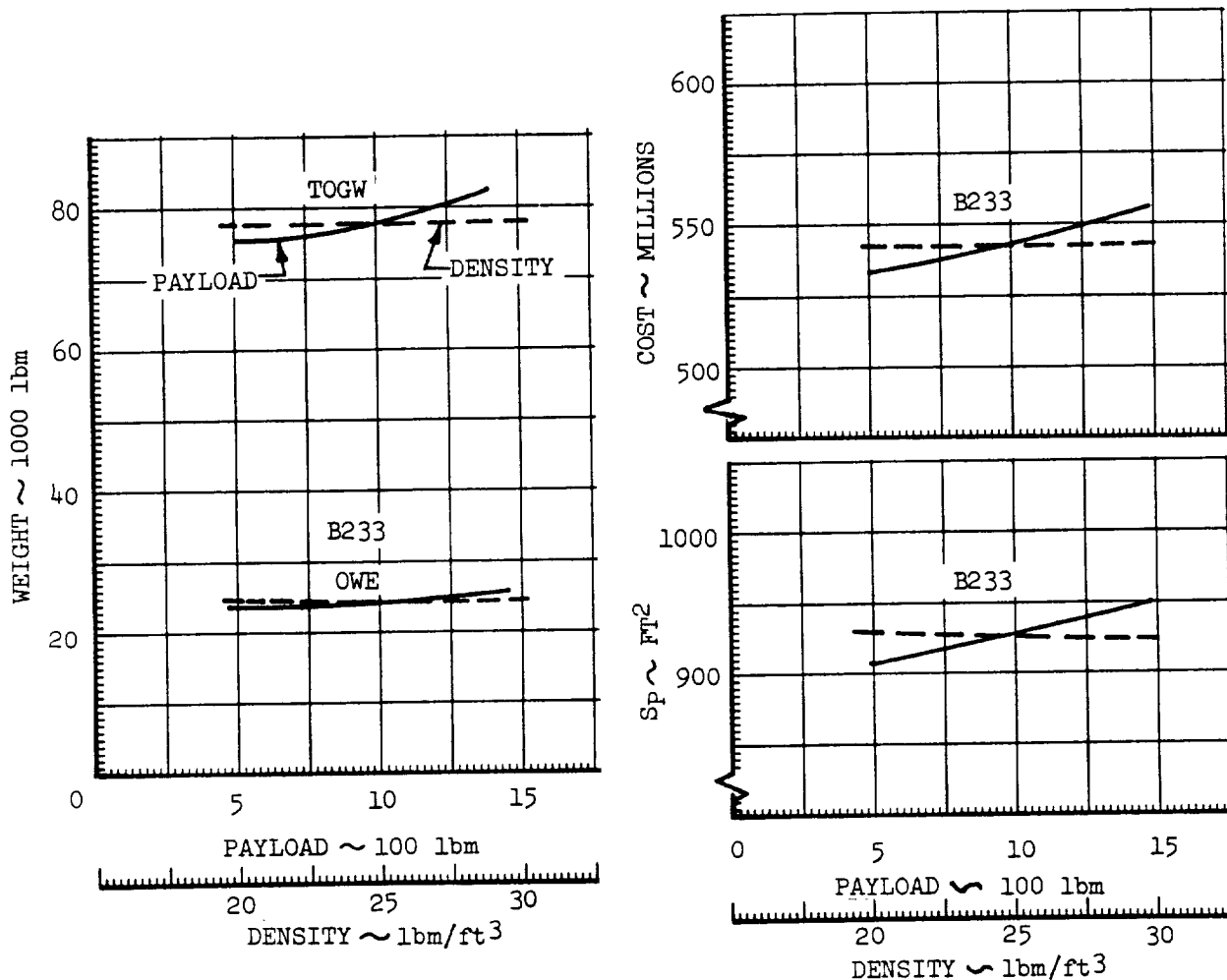


(U) FIGURE 4-124
EFFECT OF IMPROVED PASSIVE TPS ON
FLIGHT FACILITY RESEARCH VALUE



4.4.5.3 (U) Payload - Planform size, OWE, TOGW, and cost of the research aircraft, as shown in Figure 4-125, are relatively insensitive to payload variations between 500 and 1500 lb (226 and 680 kg) and to packaging densities between 20 and 30 lb/ft³ (.321 and .481 gm/cm³).

(U) FIGURE 4-125
EFFECTS OF IMPROVED PAYLOAD ON FLIGHT FACILITY COST



(U) The requirements of the nose structure of a Mach 12 vehicle (based on angular constraints combined with the cross section necessary for cockpit and nose landing gear) create a volume that will accept up to a 1500 lb (680 kg) data package with no enlargement of the structure. A greater payload requires a longer nose or relocation of the forward fuel bulkhead, either of which will cause an increase in size and cost.

(U) The actual research value of an increased payload is hard to assess numerically, but using 1000 lb (453 kg) as a baseline, an increase to 1500 (680 kg) represents a 50% increase in data return for only a 3% increase in program cost.

4.4.5.4 (U) Conclusions - Conclusions from these parametrics are as follows:

- o Design for Mach 12. For high research value and return on investment, growth potential, and program flexibility, a Mach 12 program is very desirable.

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- o In the Mach 6 to Mach 8 regime, design to Mach 6. The primary propulsion system testing by the vehicle will be airbreathing ramjet type engines. The engine and inlet for this type of propulsion system can be cooled efficiently up to Mach 6.0. Increasing to above Mach 6.5, the equivalence ratio exceeds 1 and the engine I_{sp} is decreased making the engine/inlet combination progressively less efficient with increased Mach number.
- o Test time should be 5 minutes of steady state cruise, especially for rocket powered research aircraft. Longer times do not show significant value increase, but do show significant increases in vehicle size and program cost. Test time is somewhat variable with the propulsion system used; with airbreathing propulsion a 10 minute time results in an improved research value at a small increase in cost. Another consideration, which is discussed more fully in Section 4.4.6, is test and tracking range limits.
- o Payload has been chosen as 1500 lb (680 kg). At a weight of 1.0 lb per data sensor, this will provide 1500 data points on any flight. This payload has a small effect on size and cost compared to the greatly enhanced research value and increased data return.

4.4.6 (U) FLIGHT OPERATIONS - The study of flight operations for hypersonic research vehicles is needed to ensure that those flight research vehicles selected have a high probability of achieving the research objectives. The best design concept encompassing shape, structures, propulsion, performance, etc. would be of little value if operational constraints severely limited its capabilities.

(U) Thus, the overall objectives of the flight operations study are:

- o To determine the operational feasibility of the conceptual variations.
- o To determine the operational requirements.
- o To define any constraints, limitations, or problems of otherwise acceptable concepts.
- o To define a preliminary operating plan.

(U) The Phase I study effort regarding operations, reported in Volume II, treated a wide spectrum of design and operational concepts with specific Mach numbers and test times. It defined general operating requirements for the various concepts and identified candidate facilities required for design missions. This data and other criteria were used to select the most promising flight facilities for Phase II.

(U) The Phase II flight operations study reported here treats the selected concepts and the effects of tradeoffs in design features and performance capabilities. The study reevaluated the requirements defined during Phase I and identified and evaluated new requirements, particularly those which affect the development and operating capability of the candidate vehicles.

4.4.6.1 (U) Basing - Basing is the most important consideration in the operating plan. It selects the center of operations and defines the test ranges which must be compatible with the performance capabilities of the test vehicle. A basing study involves both determining the number of bases required and their suitable location in order to accommodate the mission capabilities of the candidate test vehicles.

(U) The basing study performed during Phase I concluded that two base operations were necessary for the Mach 6 and Mach 12 five-minute cruise vehicles and it recommended test ranges and operating bases. Other studies were performed during Phase II in order to:

- o Reevaluate the requirements for the selected Phase II flight facilities because of refinements in vehicle design and mission profiles.
- o Determine the requirements of these vehicles as a result of tradeoff studies affecting mission capabilities.
- o Provide a more detailed evaluation of the basing requirements during the flight envelope expansion phases.
- o Determine the effects of designing the various flight facilities to operate from a single base.

The factors which affect basing concepts and determine the test range requirements are discussed below.

4.4.6.1.1 (U) Range - Range is the most dominant factor with respect to basing. The distance required to perform a specific design mission dictates the number of bases required, how far apart bases have to be, and if certain missions are possible under the guide lines of operation specified. Range requirements are a function of the mission profiles and the performance capabilities.

(a) (U) Mission Profiles - The mission profile as established for Phase I, summarized in Volume II, consists of three distinct segments: (1) acceleration and climb to cruise altitude and Mach number; (2) cruise distance for five minutes at cruise Mach number; and (3) descent to touchdown. Two possibilities were shown for the descent segments: a max L/D equilibrium glide, and a minimum range obtained by a 3.5 "g" power-off wind-up turn. Subsequent discussions with NASA personnel at the Flight Research Center (FRC) resulted in Phase II study efforts directed toward guidelines suggested by FRC. Test runs should be made in-bound toward the landing site. High Mach number turns should not be made as a part of the routine operations in the flight plan. However, high Mach number maneuvers as part of the test program are acceptable. NASA personnel expressed apprehension about the possibility of unknown problems occurring, such as material failure, aircraft instability and pilot disorientation occurring during high Mach number turns. These limitations affect only the single base operations, since two-base missions are routinely flown toward the landing site. Speed brakes were added to the vehicles in order to allow the pilot to adjust the glide distance to reach the landing terminal point, nominally Mach No. = 1.0 at 40,000 ft (12,200 m) directly over the landing site. See Section 4.2.7 for speed brake methodology. Figure 4-126 depicts a typical Phase II flight profile and defines the basis on which the total nominal ranges were calculated. Figure 4-127 shows the effect of speed brakes on the total range.

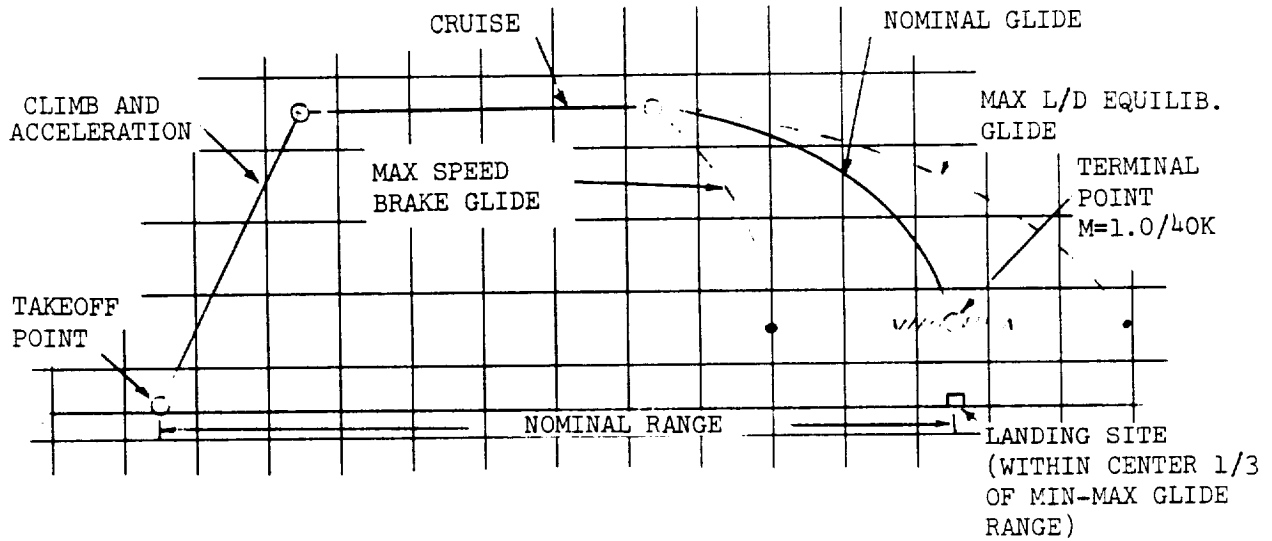
(b) (U) Performance Capabilities - The range requirements of the test vehicles is a function of the distance required to accelerate and climb to the test Mach No./altitude, the test Mach No., the test time spent at the test Mach No., and the nominal glide range. The climb and acceleration distance is a direct function of the type of acceleration engine (rocket or airbreather). Tradeoff studies of the T/W ratio, test Mach Numbers, and test time were performed on several of the Phase II vehicles. The data in Section 4.4.1 shows T/W has only a minor affect on total range requirements. The effects of test time for the two types of acceleration engines are shown in Figure 4-128. Figure 4-129 summarizes the range requirements by flight profile segment

4.4.6.1.2 (U) Test Range Features - In the Phase I study the important characteristics which must be considered in the test base/range selection were defined. These characteristics (or criteria) fall in one of two general categories.

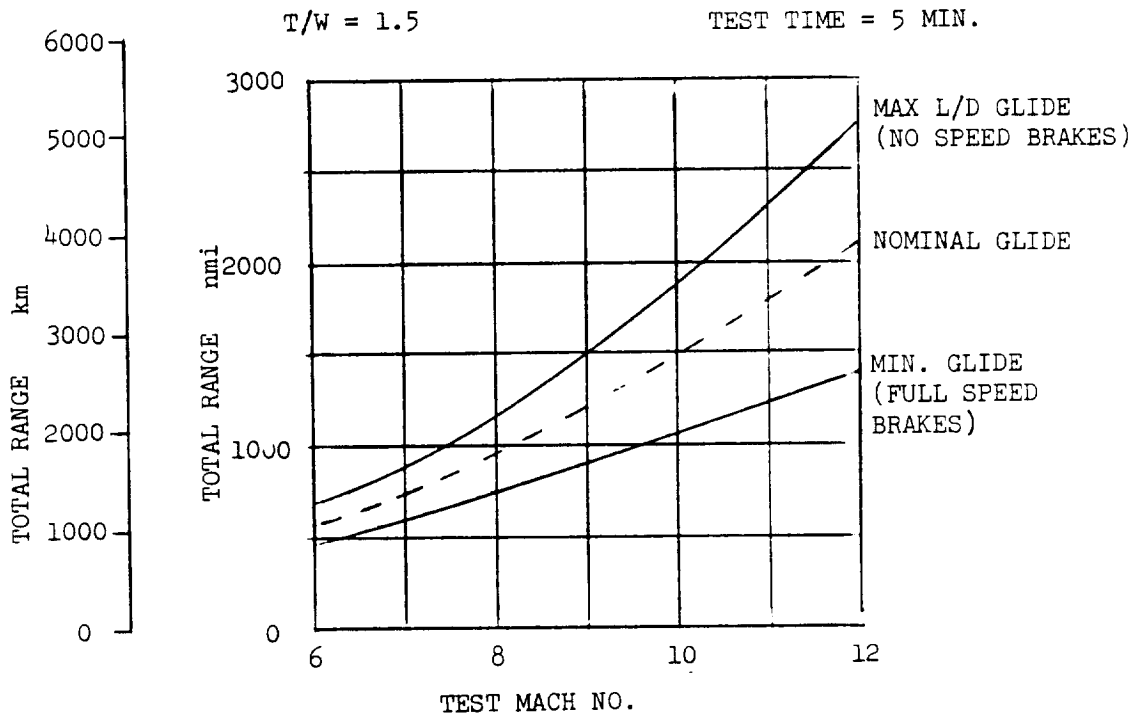
(a) (U) Natural features which include:

- o Weather
- o Population density

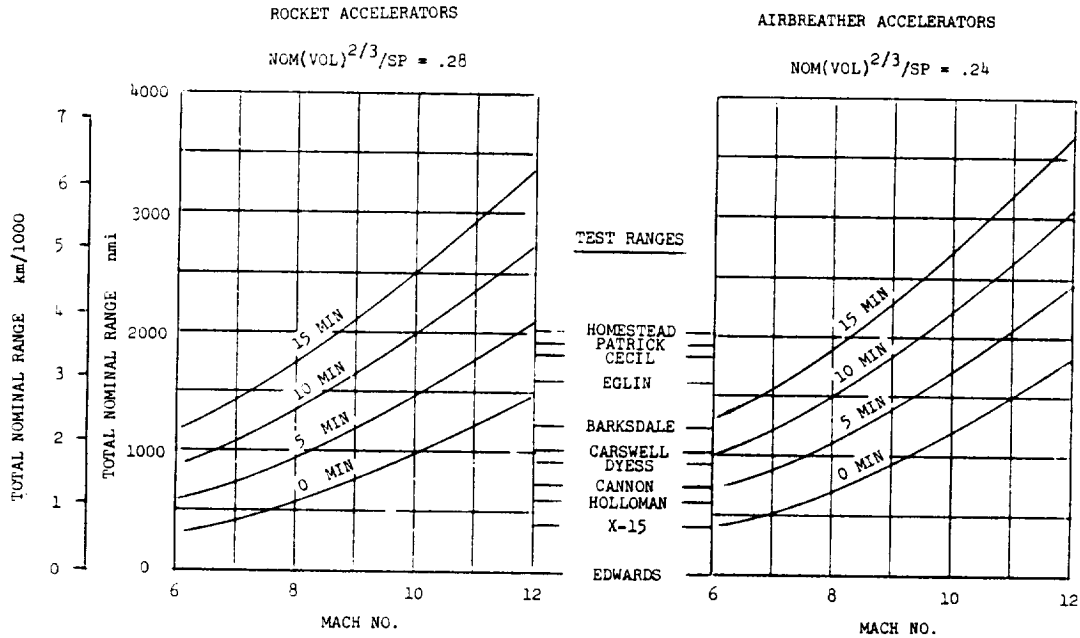
(U) FIGURE 4-126
TYPICAL PHASE II FLIGHT PROFILE



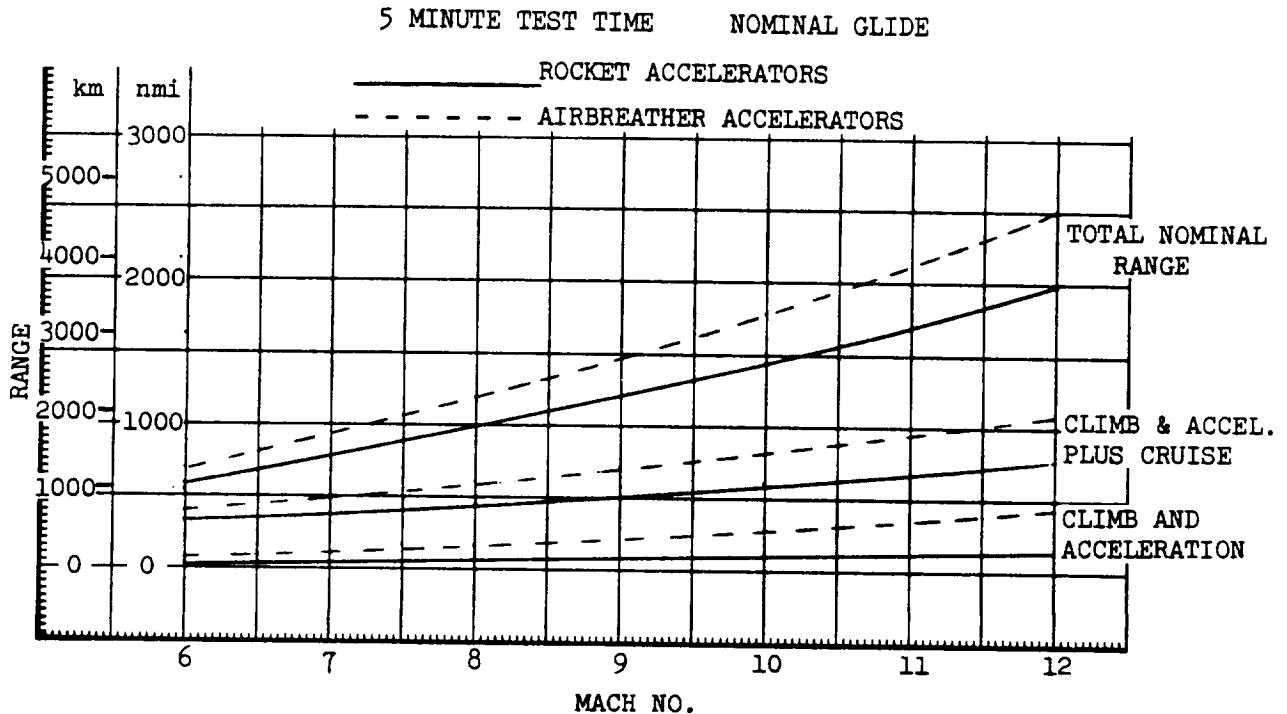
(U) FIGURE 4-127
SPEEDBRAKE EFFECT ON TOTAL RANGE



(U) FIGURE 4-128
TEST TIME EFFECT ON RANGE



(U) FIGURE 4-129
RANGE SUMMARY



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- o Geographic location
- o Available emergency landing sites
- o Suitable takeoff and landing bases

(b) (U) Support features and facilities which make a range attractive if they already exist, but could be provided at some additional costs. These include:

- o Tracking and communications facilities
- o Data acquisition facilities
- o Maintenance facilities, including support equipment and fuel storage facilities

(U) The criteria of group (a) tend to dictate the basing and test range selections. Thus, all the recommended ranges for Phase II are in the southwest or over the extreme southern portion of the United States. Flights over northern or northeast sections were considered unacceptable because of weather conditions, high population density, limited emergency landing sites, and little or no existing facilities (tracking, communication, and maintenance). Figure 4-130 shows the candidate bases, test ranges, available emergency landing sites, and tracking/telemetry facilities. Figure 4-131 shows the population density of the candidate test ranges.

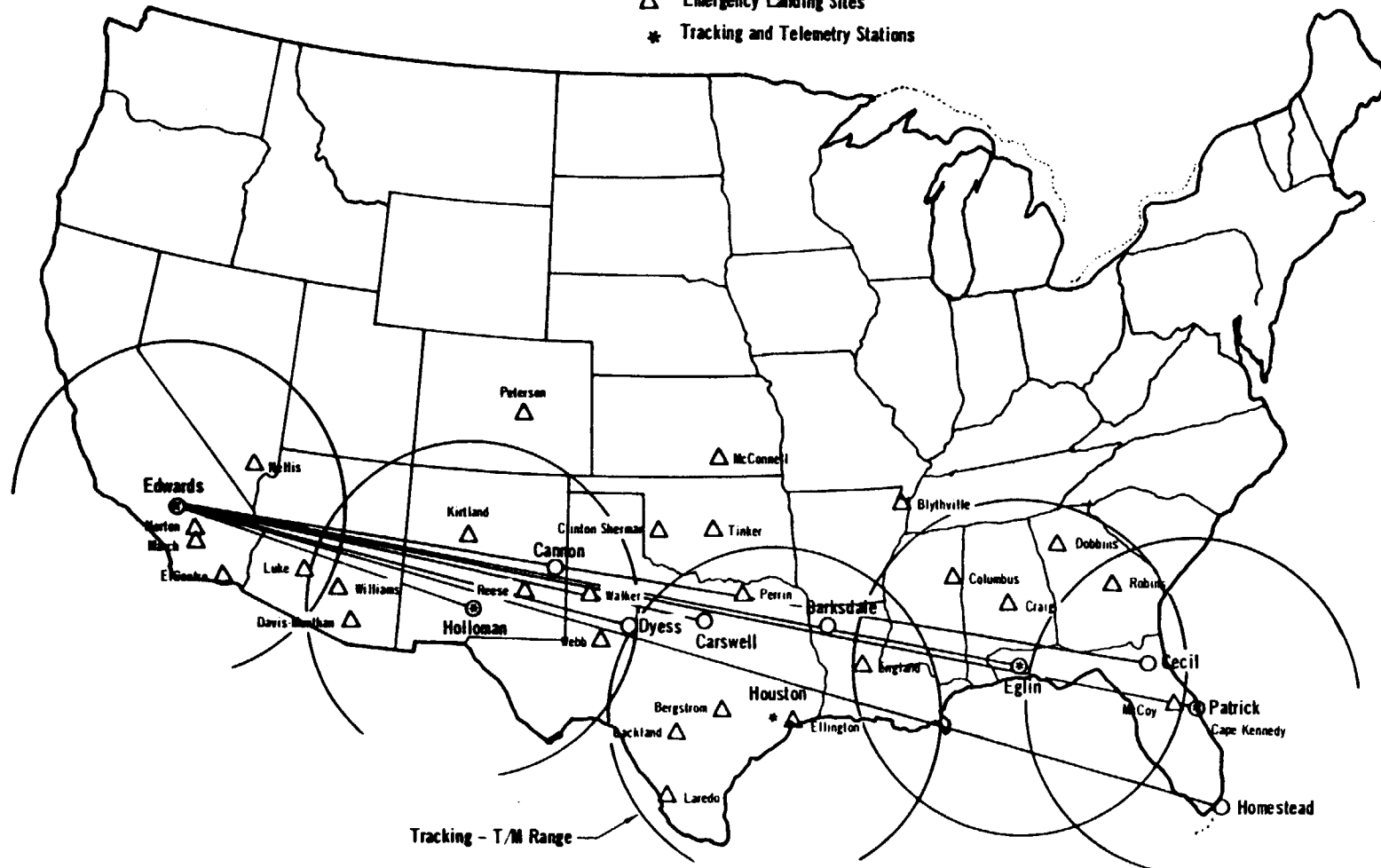
(U) Edwards AFB, California, was selected as the center of operations and the landing site for all of the vehicles which were retained for Phase II. Holloman AFB, New Mexico, which also has many of the features and facilities of Edwards AFB, was considered as a suitable landing site for vehicles with turbojet engines. The base is located at the edge of the White Sands test range, making the landing site visible and identifiable from long distances.

(U) Candidate launch bases were selected on the basis of range from the landing site (Edwards AFB, California), geographic location, and available facilities. Figure 4-132 identifies these bases and the specific design missions which could be accomplished from each. Other mission capabilities can be determined from Figure 4-128.

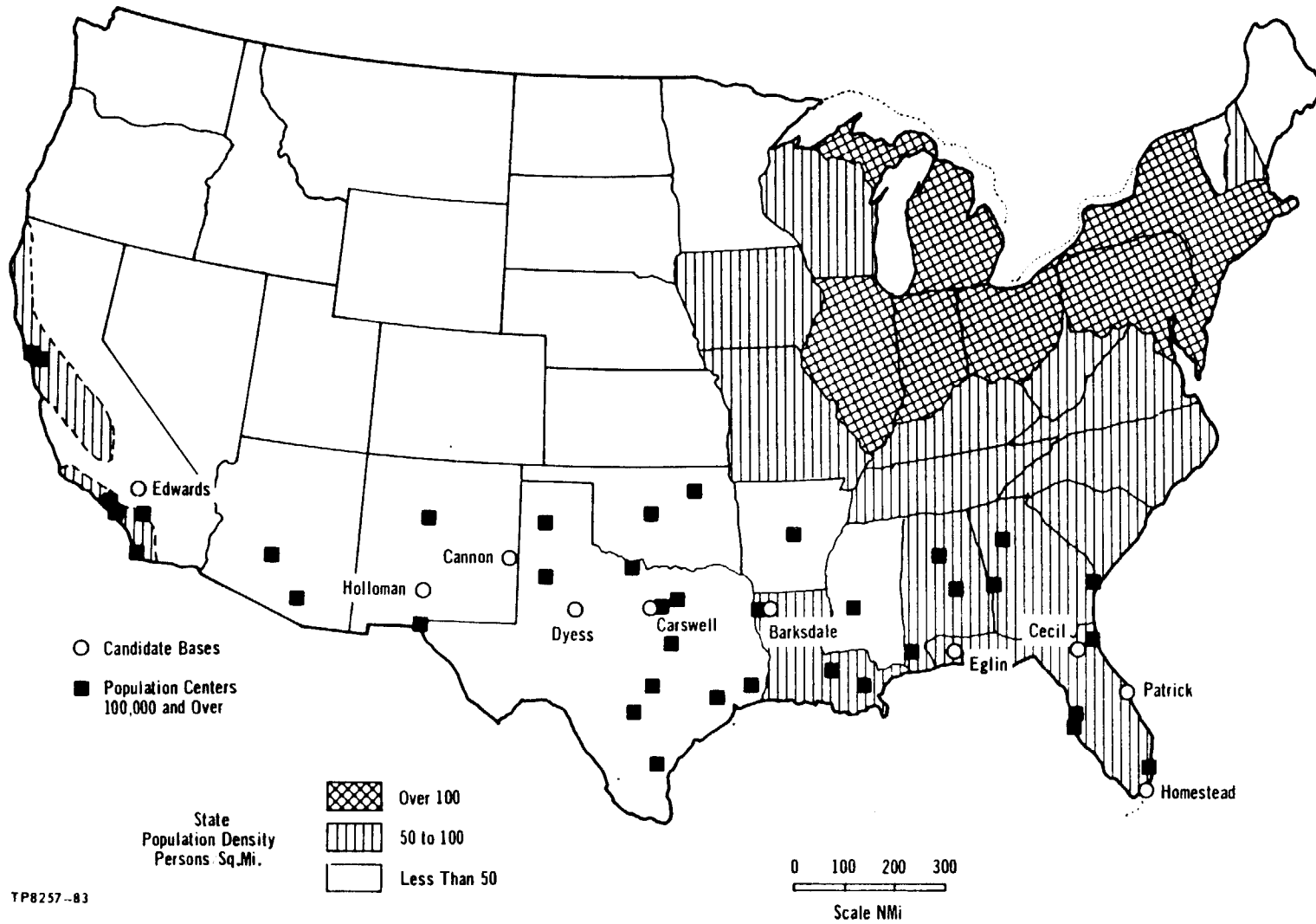
4.4.6.1.3 (U) Operational Constraints - The candidate test ranges defined in Figures 4-130, 4-131, and 4-132 show that Mach 6 to 8 vehicles can operate in the sparsely populated Southwest where the weather is generally mild and relatively predictable. The Mach 10 to 12 vehicles require bases in Louisiana or Florida where operations will be subjected to the more severe and unpredictable weather conditions of the Gulf Coast. Flights from bases in northern Florida would overfly more heavily populated areas than flights originating from other bases. Flights from Homestead AFB, Florida, either have to operate over water for a considerable distance (650 nmi/1200 km) or fly north and make a high Mach number turn toward Edwards AFB. Disadvantages of over water flight are: (1) crew survivability problems if bailout over water occurs; (2) necessity to provide search and rescue craft on standby at sea for each mission; and (3) the inability to reach an emergency landing site if a power loss occurs during acceleration between Mach 3 and 8.

(U) FIGURE 4-130
TEST RANGE FACILITIES

- Candidate Bases
- △ Emergency Landing Sites
- * Tracking and Telemetry Stations



(U) FIGURE 4-131
TEST RANGE PROPULATION DENSITY



(U) FIGURE 4-132
TEST RANGE MISSION CAPABILITY

BASE	RANGES TO EDWARDS AFB n mi/km	DESIGN CRUISE MISSION CAPABILITIES			
		ROCKET ACCELERATORS		AIRBREATHER ACCELERATORS	
		MACH	TIME MIN	MACH	TIME MIN
Holloman AFB, New Mexico	600/1110	6 8	5 0	7 -	0
Cannon AFB, New Mexico	720/1340	7 8.5	5 0	6 8	5 0
Dyess AFB, Texas	915/1690	6 8 9.5	10 5 0	7 9	5 0
Carswell AFB, Texas	1020/1890	7 10	10 0	6 9.3	10 0
Barksdale AFB, La.	1220/2250	6 8 11	15 10 0	6 7 10	15 10 0
Eglin AFB, Fla.	1600/2960	7 10 12	15 5 0	7 8 10 11	15 10 5 0
Cecil Fld NAS, Fla.	1830/3380	8 12	15 3	12 -	0
Patrick AFB, Fla.	1935/3580	12 -	4	8 12	15 1
Homestead AFB, Fla.	2020/3740	10 12	10 5	10 12	5 2

(U) The operating constraints of basing for the Mach 10 to 12 missions are serious; however, they must be viewed with respect to the overall program. Before a Mach 10 or 12 mission is attempted, the vehicles will have undergone a development program. Many flights will have been flown in an incremental build-up in Mach No. and test time, and a high degree of confidence and reliability will have been established in the vehicle and its systems. Also, the number of design missions to be flown must be considered. Many research objectives can be obtained from flights at lesser Mach No. and test times which could be staged from intermediate bases. In the total program of approximately 200 flights, probably only a modest percentage would require design missions. Also, the effects of the above constraints differ for the various vehicles studied. The operational considerations and constraints are discussed below.

(a) (U) Airlaunch Vehicles - The airdrop vehicles provide a unique versatility to the operations planning for a hypersonic test vehicle, since the mother ship is essentially an airborne second base. This provides a high degree of flexibility during early flight development and permits a buildup in Mach No. and test time to relatively high levels while operating from a single ground base.

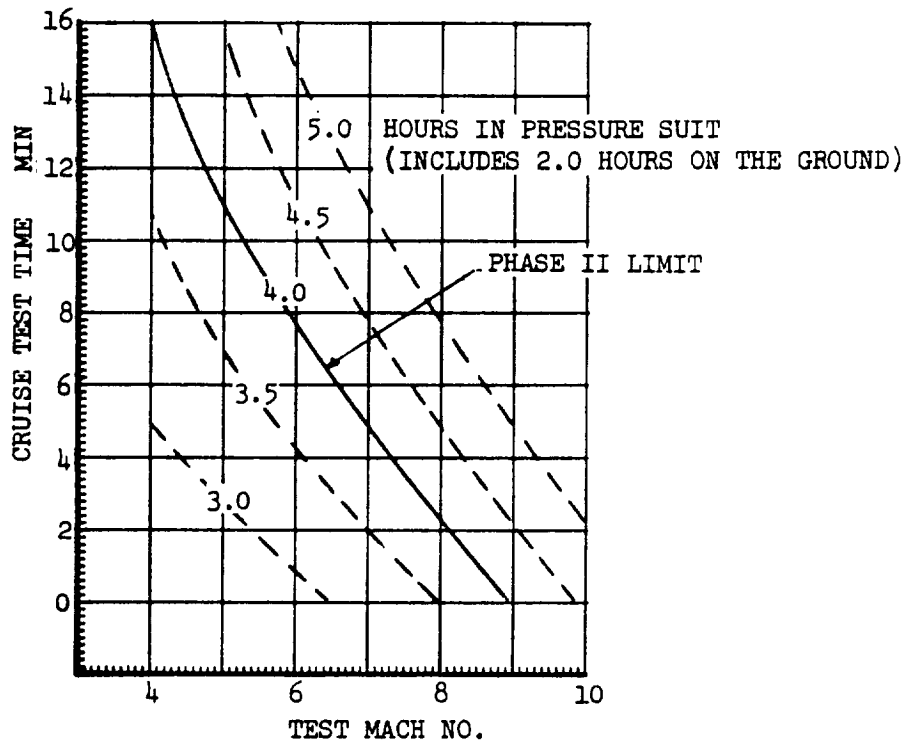
(U) The airdrop feature will permit assessment of the power-off glide and flight characteristics from drops directly over the landing site without the need to power the vehicle into the air for the first flight. Initially, powered flights can be performed on the existing X-15 test range with its numerous drop zones permitting incremental envelope expansion up to Mach 7 or sustained Mach 6 for two minutes. Using the Holloman AFB to Edwards AFB test range will permit speed build up to Mach 8 and sustained cruise of Mach 7 for 3 minutes or Mach 6 for 5 minutes.

(U) Missions from Holloman AFB can be based at Edwards AFB. Using the C5A, all airdrop missions could be staged from a single base; however, missions requiring longer ranges become inhibited by constraints. The primary constraint is the long duration that the pilot has to spend in a pressure suit, including time on the ground prior to takeoff, (an estimated 2.0 hours based on X-15 experience), and because of the relatively slow outbound ferry (Mach = .8) of the mother ship. Discussions with NASA at FRC resulted in establishing a nominal limit of four hours total suit time for Phase II planning. Figure 4-133 shows mission performance capabilities (Mach No. and test time) as a function of pressure suit time. These times are based on the test mission being flown. A mission aborted prior to drop would result in a considerably longer period in the suit. Also, returning long distances following an abort could limit missions to the early part of the day to ensure a return during daylight.

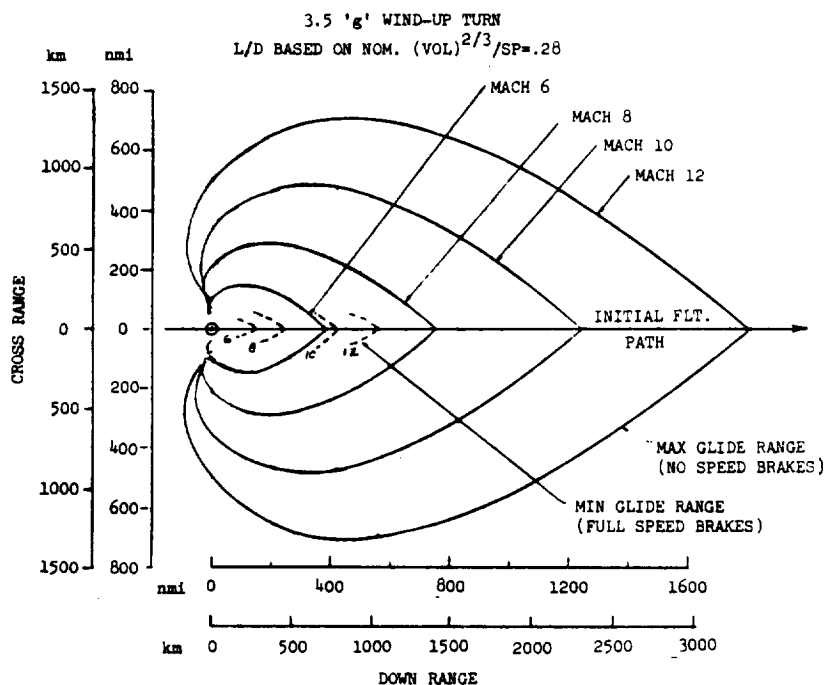
(U) Additional buildup in vehicle performance can be achieved prior to Mach 12 missions by airdrops in the vicinity of Carswell AFB, Texas, where Mach 8.5 can be sustained for five minutes. Drops near Eglin AFB, Florida, will permit testing at Mach 10 for five minutes and a speed buildup to Mach 12.

(U) Emergency landing capability is another important operational consideration. Figure 4-134 shows the emergency landing foot prints for unpowered glides from various Mach numbers. These data indicate the rather limited footprints from Mach numbers below 6 or 8. Missions flown from eastern bases where the candidate emergency landing sites are fewer and farther between may be constraining. Rocket accelerated vehicles, in particular the air launched vehicles have the advantage of being able to attain high Mach numbers in a relatively short time and distance (Mach 8 in 2.5 minutes and 90 nmi/160 km compared to 5 or 6 minutes and 200 nmi/370 km for turbojet accelerated vehicles). Thus, airlaunching should result in better operational flexibility in regards to assuring adequate emergency landing sites.

(U) FIGURE 4-133
EFFECTS OF PRESSURE SUIT TIME CONSTRAINT ON
SINGLE BASE AIRLAUNCH MISSIONS



(U) FIGURE 4-134
EMERGENCY DESCENT LANDING FOOTPRINT



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(U) Further basing options result from the use of the C-5 mother ship, including its ability to carry a considerable amount of support equipment on-board. In all options, Edwards AFB is assumed to be the main base and landing site for all flights. Following are some of the basing options considered.

- (1) Set up a permanent operating base, including complete maintenance facilities, at each intermediate base as it is used during the program.
- (2) Set up one permanent base at an optimum location and ferry to drop zones over other intermediate bases.
- (3) Use Edwards AFB as the only permanent base and ferry to intermediate sites and land prior to the mission. Support equipment necessary for pre-flight, routine maintenance, and launch would be carried in the C-5.

These, and other operating considerations will be studied in more detail during Phase III.

(b) (U) HTO Rocket Accelerators - HTO vehicles with rocket accelerator engines present operating problems with respect to early development flights where it will be necessary to incrementally expand the flight envelope. Several development plans were considered which are discussed in the following paragraphs.

- (1) Single base, test inbound - This is the most attractive plan from an operational safety standpoint; however, data presented in Phase I showed that mission capability would be very limited because the rocket accelerated vehicles cannot economically cruise outbound at subsonic speeds and accelerate inbound. Speeds to Mach 3 are about the maximum for this flight profile.
- (2) Single base, test outbound - Testing to Mach 8 is feasible for this profile; however, a complex maneuver with a high Mach No. turn is required to return and land. This type of profile was discussed in Paragraph 4.4.6.1.1(a) and found objectionable.
- (3) Dual base, test one way - This profile offers the advantage of testing inbound toward the landing site; however, there are several disadvantages because of increased risk factors. The nearest recommended take-off base to Edwards AFB is Holloman AFB 600 nmi(1110 km) away. This would require initial flights be flown over a relatively long distance with little or no prior flight evaluation of airplane flying qualities, glide characteristics, and systems operation. Also, the long range could dictate larger incremental jumps in the Mach No. envelope expansion than would otherwise be desired.
- (4) Dual base, test in both directions - This option has an advantage over no. (3) in that the expense of transporting the vehicle to the second base would be eliminated. However, it would have the additional disadvantages of increased operational complexity, and the added risk associated with landing on a runway as a routine operation.

The recommended development plan would probably be a combination of (1) and (3), i.e., establish some confidence level with the vehicle on single base inbound missions then expand to two base operations.

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(U) Probably the most difficult problem with HTO rocket accelerator vehicles would be in transporting them to the takeoff base. Use of a C-5 aircraft to transport the vehicle from Edwards AFB to the takeoff base was included in the operational plan described in Phase I. The HTO vehicles, being larger than airdrop vehicles, will require a special pylon on the C-5 to allow them to be canted (in roll attitude) from the horizontal position. Another consideration would be strap-on turbojets for ferry capability; however, this would significantly increase program cost for procurement, design, manufacture, and the development testing.

(c) (U) HTO Airbreather Accelerator - The HTO airbreathers employing turbojet engines which can be operated much more efficiently and economically at subsonic airspeeds, have a decided operational advantage over the HTO rocket accelerators. Development and evaluation of the low speed flight characteristics would be similar to conventional jet aircraft. Speed buildups could be performed by subsonic cruise outbound and testing inbound on the X-15 test range; however, the maximum mission capabilities on the X-15 range would be considerably less than the rocket accelerated airlaunch vehicles because of the slower acceleration of airbreathers. Speeds to Mach 3 or 4 might be achieved, permitting a limited assessment of the cruise engine (RJ or CSJ) operation, particularly the starting characteristics. Further envelope expansion would require two base operations. Flights from Holloman AFB to Edwards AFB would provide tests to Mach 6 and Mach 7 which would be a significant evaluation of an RJ engine, but a very limited evaluation for CSJ operation. Additional buildup in Mach No. from 7 to 9 can be accomplished from Cannon AFB, New Mexico, and Dyess AFB, Texas. Envelope expansion beyond Mach 9 will require basing in Louisiana and Florida; thereby requiring a higher percentage of total flights to be flown from these areas.

(U) The maximum mission capability for an all airbreather while operating on the longest defined test range (Homestead AFB, Florida to Edwards AFB, California) is Mach 11 for five minutes or Mach 12 for two minutes. Operating an airbreather out of Homestead AFB presents more of a problem than for the rocket vehicles because it will take from five to six minutes and 200 nmi (370 km) of flight over water to attain Mach 8 (the minimum speed required for glide to an emergency landing site on land). Also, the CSJ engine will probably not have the reliability factor of the rocket engines, although this is offset by the potential availability of reliable turbojet engines to cruise to a landing site.

(U) The airbreather vehicles should not have the severe problem of getting to a second base that the HTO rocket vehicles have. The availability of turbojet engines gives them the potential to ferry to the takeoff bases, especially since this capability can be evaluated during single base missions at Edwards AFB. It is also conceivable that test missions could be flown in both directions. However, landings on conventional runways rather than dry lake beds would probably require that some fuel be conserved for a limited loiter capability as well as go-around. Thus, eastbound test missions might be more limited in test Mach No. and time than westbound missions.

(d) (U) Unmanned Vehicles - Vehicle B284 is an unmanned version of the all rocket, Mach 12, air launched B233 which has a nominal range of 2100 nmi (3890 km) and utilizes the Homestead AFB to Edwards AFB test range. B284 is also assumed to use this test range at this point because only bases and test ranges in CONUS

have been considered acceptable for a flight research vehicle. However, unmanned flights over CONUS would be questionable because of the safety factors involved. Landing at an emergency site would probably not be feasible since unmanned vehicle recovery, even at a large landing site like Edwards AFB, would be extremely complex and require special equipment. It would therefore seem necessary to be able to get the vehicle over open water if an emergency occurs. This might not always be possible on a CONUS test range. The unmanned research vehicle, therefore, would be very constrained from an operational standpoint.

4.4.6.1.4 (U) Single Base Tradeoff Study - The Phase II baseline vehicles were all designed (size, weight, fuel, etc.) to perform a straight line mission utilizing two bases. If these vehicles could operate from a single base, flight operations would be less complex and operating costs would be reduced. However, performing the design mission from a single base requires additional fuel and increases vehicle size and weight, and costs. A study was performed during Phase II to determine the cost tradeoffs of single base vs dual base operation. The single base vehicles were designed to perform the test mission outbound from the takeoff point and the return cruise inbound in order to minimize the growth effects (see Section 4.2.7 for methodology). The significant results of this study are discussed below.

(a) (U) Mach 6/Air Launch/Rocket Accelerator - Data for the baseline vehicle showed that the total range requirement for this vehicle's mission (five minute test time) is 600 nmi (110 km) which makes it feasible to operate single base as it is. Therefore, it was unnecessary to investigate a single base version.

(b) (U) Mach 12/Air Launch/All Rocket (B233) - The single base design of this vehicle is primarily constrained by size. The baseline vehicle has a planform area of 929 ft² (86.3 m²) and volume of 3840 ft³ (108.6 m³) compared to 2114 ft² (196.3 m²) and 13,100 ft³ (370.5 m³) required for a single base design. This is prohibitive for carriage on a mother ship.

(c) (U) Mach 12/HTO/All Rocket (B260) - The single base design of this vehicle suffers a severe growth penalty compared to the dual base baseline version. The planform area is 98% greater, the volume is 180% larger, and the OWE is 73% heavier. Since system cost is related to vehicle size and weight, costs become the primary constraint for this vehicle. The total system cost of the dual base B260 vehicle is \$742 million of which \$13.4 million (approximately 2%) is the additional cost attributable to the second operating base for AGE, AGE maintenance, AGE spares, general purpose maintenance, and vehicle transportation costs. The total system cost for a single base design of B260 is \$1,113 million, a 50% increase over the dual base baseline vehicle.

(d) (U) Mach 6-12/HTO/Airbreather - Both the Mach 6 vehicle (B212) and the Mach 10-12 vehicle (B257) are included here since neither one had the ability to grow to meet the single base mission requirements. Configuration B257, as sized, is very sensitive to any change in fuel required. Attempts to increase the fuel volume available, beyond that required to perform the HYFAC mission, results in a rapid increase in vehicle size. It is evident that any attempt to operate Configuration B257F in a single base mode will result in an outlandishly large vehicle; therefore, this concept was not pursued. For Configuration B212, some increase in fuel volume can be obtained by stretching the fuselage while maintaining a constant

wing area; however, the fuel required to operate configuration B212 in the single base mode is more than that obtainable in this manner. Attempts to grow the entire vehicle resulted in stretching the turbojet wing nacelles with a severe weight and performance penalty.

4.4.6.1.5 (U) Recommended Basing - "Near Optimum" Vehicles - The Phase II study effort on the seven (7) baseline vehicles and parametric tradeoffs resulted in the selection of a near optimum configuration for each vehicle. The basing recommendations for these vehicles are shown in Figure 4-135.

(U) FIGURE 4-135
RECOMMENDED BASING - NEAR OPTIMUM VEHICLES

VEHICLE DESIGN CONCEPT	MISSION		RANGE n mi/km	BASING	TEST BASES		TEST RANGE CAPABILITY MACH/TIME
	MACH	TIME MIN			LANDING	T.O./LAUNCH	
B207 All Body Air Launch RKT/RJ	6	10	900/1670	DUAL	EDWARDS	X-15 RANGE HOLLOMAN DYESS	6/2 6/5 6/10
B212 Wing Body HTO TJ/RJ	6	10	1025/1890	DUAL	EDWARDS	X-15 RANGE HOLLOMAN CARSWELL	3-4/0 6/3 6/10
B232 All Body Air Launch RKT/SJ	12	5	2100/3890	DUAL	EDWARDS	X-15 RANGE HOLLOMAN CARSWELL EGLIN HOMESTEAD	6/2; 7/0 6/5; 8/0 8/5; 10/0 10/6; 12/1 12/5
B233 All Body Air Launch RKT B284 (Unmanned B233)	12	5	2100/3890	DUAL	EDWARDS	SAME AS B232	SAME AS B232
B257 Bl. Wing Body HTO TJ/CSJ	12	5	2430/4490	DUAL	EDWARDS	X-15 RANGE HOLLOMAN CARSWELL EGLIN HOMESTEAD	3-4/0 6/3; 7.5/0 8/3; 9/1 10/5 12/2 1
B260 All Body HTO RKT	12	5	2100/3890	DUAL	EDWARDS	X-15 RANGE HOLLOMAN CARSWELL EGLIN HOMESTEAD	3/0 6/5; 8/0 8/5; 10/0 10/6; 12/1 12/5

NOTES: 1 Maximum all air breather mission capability of Homestead AFB to Edwards AFB test range.

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4.4.6.1.6 (U) Conclusions - The following conclusions were drawn from the basing and operating study:

(a) (U) Single basing is practical and feasible for airlaunch vehicles limited to Mach 6 capability. All other concepts require dual basing for both the speed buildup development program and the design mission.

(b) (U) Test ranges and bases located in the southwestern and southern portions of the U.S. offer the best operational potential considering features, facilities, and costs.

(c) (U) Edwards AFB with its large landing site, many facilities, and X-15 test range is an outstanding choice as a main operating base for early development testing and as a landing site for all flights.

(d) (U) Rocket powered air launch vehicles and turbojet HTO vehicles offer the greatest flexibility and least operating constraints during the initial low speed flight evaluation and envelope expansion in the high supersonic/low hypersonic flight regime. Rocket powered HTO/VTO vehicles would present the greatest difficulties in this area.

(e) (U) Rocket accelerated vehicles have an operational advantage over airbreather accelerators because of greater engine reliability, better acceleration capabilities, shorter range requirements, and maximum mission capabilities within CONUS.

(f) (U) Single base designs are not feasible for Mach 6 to 12 airbreathers, or Mach 12 airlaunch vehicles, and are not cost effective for HTO rocket accelerators.

(g) (U) Unmanned vehicles would be severely constrained by operational problems.

4.4.6.1.7 (U) Recommendations - The following recommendations are made, based on the above conclusions:

(a) Vehicles with sustained Mach 12 capability should be airlaunched with rocket acceleration engines.

(b) Vehicles with HTO capability should employ turbojet engines.

(c) All vehicles should employ manned control.

4.4.6.2 (U) Powered Landing - The relative merits of providing additional fuel for powered landings was studied during Phase II. The results obtained and conclusions drawn were found to be primarily a function of the type of propulsion system (rocket or airbreather).

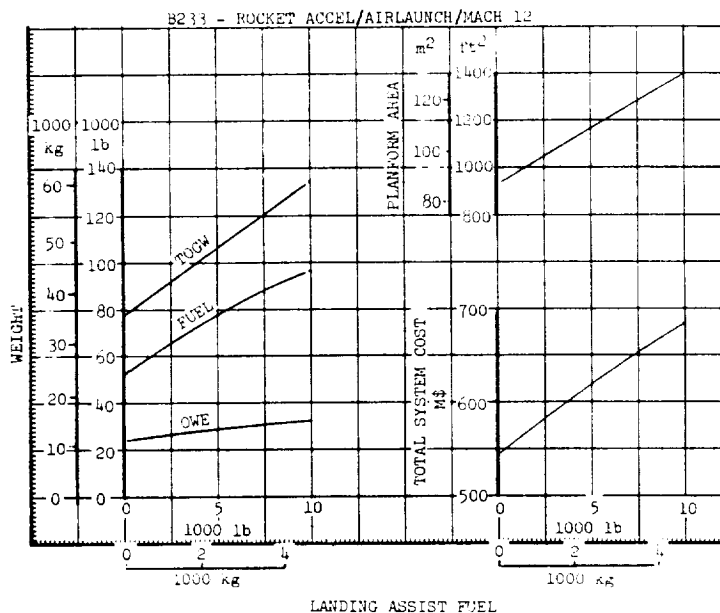
4.4.6.2.1 (U) Airbreathers - The all airbreather vehicles were studied to determine the affects of providing enough additional fuel for one go-around since turbo-jet engines can operate efficiently and economically at subsonic air speeds. Trade-off versions employing JP fuel for TJ operation were selected in lieu of the baseline vehicles for this study to minimize the vehicle volume increase (one lb of LH₂ is equivalent to three lb of JP in performance capability but requires approximately ten times the volume). The versions selected were: (1) B212A9-Mach 6, HTO, TJ/RJ; and (2) B257F91-Mach 12, HTO TJ/CSJ. Figure 4-136 illustrates the effect of providing go-around capability to these vehicles. These data indicate that go-around capability for B212 can be achieved for a relatively small cost increase. B257 on the other hand suffers a penalty in both weight and cost increases because of its sensitivity to any change in fuel required.

4.4.6.2.2 (U) Rockets - All baseline rocket accelerated vehicles were designed for unpowered glide and landing. A study was performed on the Mach 12, airlaunch, all rocket vehicle (B233) to determine the effects of providing landing power assist. Two tradeoffs were evaluated: (1) 5000 lb (2268 kg) of landing assist fuel; and (2) 10,000 lb (4536 kg) of landing assist fuel. These amounts of fuel will provide sustained level flight of 250 knots (128.5 m/s) at sea level for 30 seconds and one minute, respectively. The net gain in landing performance would be the capability to extend the glide range on final approach by 12,600 ft (3840 m) and 25,200 ft (7680 m) without go-around capability. This gain is achieved at high cost as shown in Figure 4-137.

(U) FIGURE 4-136
AIRBREATHER GO AROUND COST SUMMARIES

	B212A9		B257F91	
	WITHOUT GO AROUND	WITH GO AROUND	WITHOUT GO AROUND	WITH GO AROUND
OWE (lb/kg)	32,217/14,620	32,771/14870	68,400/31,000	79,000/35,850
FUEL- LH ₂ "	2,788/1,268	4,425/2,010	19,900/9,030	23,300/10,580
JP "	2,345/1,065	4,570/2,075	17,000/7,720	27,000/12,270
TOGW "	38,480/17,500	41,816/19,000	105,310/47,800	129,300/58,700
TOTAL SYS. COST(MIL \$)	665	671	1,131	1,236

(U) FIGURE 4-137
POWERED LANDING COSTS



(U) The increase in planform area would eliminate this consideration from B233, an airlaunch vehicle which is restricted in size. However, the values obtained could be applied to an HTO vehicle which is not as size limited. NASA at FRC has indicated that experiences with the lifting body glide vehicles have shown that the ability to extend the glide is an attractive feature, even considering the landings are performed on a large dry lake bed. Thus, the decision is one of tradeoff in added capability vs the high cost. There are also other factors to consider. The additional fuel for landings is only required on a design mission (5 min @ Mach 12), and since a large number of flights will be flown at reduced Mach number or time, some fuel would be available for most of the landings. For example: After a five minute mission at Mach = 8, 10,000 lb (4536 kg) of fuel would remain; and at Mach 10, 4000 lb (1815 kg) of fuel would remain. Downstream in the program when design missions are flown, the operating and piloting techniques would be well established, which should minimize the requirement. To utilize the rocket for landing assist, it will be necessary to maintain the engine in a chill-down mode during the landing approach. This is necessary to assure rapid engine start and throttling capability. Based on the above considerations, the high increase in cost is not justified for the limited increase in performance capability.

4.4.6.2.3 (U) Conclusions and Recommendations - The following conclusions and recommendations are made with respect to power for landing.

(a) (U) Powered landing with go-around capability is recommended for Mach 6 turbojet powered vehicles because it improves the operating capability with a minimum effect on cost and performance.

(b) (U) It is not recommended that rocket powered vehicles or Mach 12 air-breathers be designed for powered landing because: (1) the capability will be available for all but maximum design missions (small percentage of total flights); and (2) the resultant size growth of airlaunched vehicles is large.

4.5 DATA SUMMARIES

(U) This section presents summaries of the design characteristics and performance, the weights, and the costs of the actual configurations analyzed in the parametric studies.

4.5.1 (U) DESIGN CHARACTERISTICS AND PERFORMANCE - Various vehicle tradeoff studies were performed to determine the effects on vehicle size. These studies resulted in a change in either mission definition or vehicle design.

(U) The tradeoff studies affecting mission definition are:

Test Mach number variation (A1)
Test time variation (A3)
Single base operation (A6).

(U) The tradeoff studies affecting vehicle design are:

Engine thrust to weight variation (A2)
Payload weight and density variation (A4)
Design load factor variation (A5)
Thermal protection system variation (A7)
Fuel tank variation, integral and non-integral (A8)
Fuel variation (A9)
Landing assist effect (A11)
Off-shelf engine effect (B)
Wing-leading edge sweep variation. (D and E)

(U) The alpha-numeric characters within the parenthesis designate the vehicle tradeoff study. For example, B233A-A1 indicates a test Mach number variation was studied on the B233A configuration.

(U) A summary of these tradeoffs, and their effects on vehicle size is presented in Figures 4-138 through 4-144. The discussion section for each of the tradeoffs is cross-referenced on the figures.

4.5.2 (U) WEIGHTS SUMMARY - Seven base vehicle configurations were sized for equal performance based on the design criteria described in Section 4.1. Sixty-five tradeoff studies were performed on selected vehicles to meet the base vehicle performance and design criteria, as indicated in the following table:

<u>Configuration</u>	<u>Number of Tradeoff Studies</u>
B207A	8
B212A	9
B232A	2
B233A	26
B257F	12
B260A	6
B284A	2

**(U) FIGURE 4-138
TRADE-OFF RESUME
B207 Configuration**

AIR LAUNCH ALL BODY CONFIG. LH₂ FUEL
ROCKET ENGINE FOR ACCEL. & CLIMB RAM-JET CRUISE

TRADE-OFF DESIGNATION	DESIGN MACH NO.	S_p ft. ²	VOLUME TOTAL ft. ³	VOLUME FUEL ft. ³	$VOL^{2/3}/S_p(L/D)_{MAX}$	OWE lb.	TOGW lb.	PROPELLANT WT. ACCEL. & CRUISE CLIMB lb.	WT. lb.	W_F/W_{TO}	$(W/S_p)_{TO}$ lb./ft. ²	TRADE-OFF DESCRIPTION		SECTION REFERENCE
A	6	426	1,700	775	0.335	3.19	22,360	38,430	14,920	650	0.405	90.3	BASLINE CONFIGURATION	4.3
A1-1	↓	420	1,671	733	↓	↓	21,440	36,166	14,120	606	0.408	86.0	TEST MACH NO. VARIATION	MACH NO. = 6 4.4.5.1
A1-2	7	570	2,633	1,092	3.03	3.03	24,350	45,300	19,800	1,150	0.463	79.5	↓	MACH NO. = 7 4.4.5.1
A1-3	8	1,200	3,070	3,510	2.90	2.90	48,500	105,530	50,750	6,280	0.540	88.0	↓	MACH NO. = 8 4.4.5.1
A2-1	6	435	1,760	800	3.13	3.13	22,360	39,020	15,510	650	0.414	89.8	THRUST TO WEIGHT VARIATION	T/W = 1.25 4.4.2.2
A2-2	↓	423	1,687	770	↓	↓	23,000	38,425	14,770	655	0.401	91.0	↓	T/W = 2.00 4.4.2.2
A6	↓	640	3,140	13,820	↓	↓	27,100	49,332	19,700	2,532	0.451	77.2	SINGLE BASE OPERATION	4.4.6.1
B	↓	520	2,386	1,008	0.343	3.09	24,200	42,912	18,000	712	0.436	82.5	OFF-SHELF ENGINE (3) ENG. RL-10 (MOD)	T/W = 1.59 4.4.1
REVISION A	6	502	2,178	921	0.335	3.19	27,180	45,663	17,710	773	0.405	91.0	BASLINE CONFIG. REVISION	4.3

(Presented in International Standard Units)

TRADEOFF DESIGN DESIGNATION MACH NO.	S_p m ²	VOLUME TOTAL m ³	VOLUME FUEL m ³	$VOL^{2/3}/S_p(L/D)_{MAX}$	OWE kg	TOGW kg	PROPELLANT WT. ACCEL. & CLIMB kg	CRUISE kg	W_F/W_{TO}	$(W/S_p)_{TO}$ kg/m ²	TRADEOFF DESCRIPTION	SECTION REFERENCE		
A	6	39.6	48.11	21.9	0.335	3.19	10,267	17,428	6766	295	0.405	440.9	BASLINE CONFIGURATION	4.3
A1-1	↓	39.0	47.29	20.7	↓	↓	9,723	16,401	6403	275	0.408	419.9	TEST MACH NO VARIATION	MACH NO. = 6 4.4.5.1
A1-2	7	53.0	74.7	30.9	3.03	3.03	11,043	20,543	8979	522	0.463	388.2	↓	MACH NO. = 7 4.4.5.1
A1-3	8	111.5	228.4	99.3	2.90	2.90	21,994	47,857	23015	2848	0.540	429.7	↓	MACH NO. = 8 4.4.5.1
A2-1	6	40.4	49.8	22.6	3.19	3.19	10,367	17,695	7034	295	0.414	438.4	THRUST TO WEIGHT VARIATION	T/W = 1.25 4.4.2.2
A2-2	↓	39.3	47.74	21.8	↓	↓	10,430	17,425	6698	297	0.401	444.3	↓	T/W = 2.00 4.4.2.2
A6	↓	59.4	88.9	391.3	↓	↓	12,292	22,376	8936	1148	0.451	376.9	SINGLE BASE OPERATION	4.4.6.1
B	↓	48.3	67.52	28.5	0.343	3.09	10,975	19,460	8163	323	0.436	402.8	OFF-SHELF ENGINE (3) ENG. RL-10 (MOD)	T/W = 1.59 4.4.1
REVISION A	6	46.6	61.6	26.1	0.335	3.19	12,329	20,713	8033	351	0.405	444.3	BASLINE CONFIG. REVISION	4.3

**(U) FIGURE 4-139
TRADE-OFF RESUME
B212 Configuration**

TRADE-OFF DESIGNATION	Sp ft ²	VOLUME TOTAL ft. ³	VOLUME FUEL ft. ³	(L/D)MAX	OWE lb.	TOGW lb.	PROPPELLANT WT. ACCEL. & CRUISE CLIMB lb. lb.		FUEL lb.	W _F /W _{TO}	(W/Sp) _{TO} lb./ft. ²	TRADE-OFF DESCRIPTION	SECTION REFERENCE
							CLIMB	CRUISE					
A	1,004	3,048	1,005	3.25	32,540	37,223	3,573	1,110	LM2	0.126	37.1	BASE-LINE CONFIGURATION	4.3
A3-1	1,222	4,337	1,430		34,850	41,520	4,330	2,340		0.161	34.0	TEST TIME VARIATION t = 10 MIN.	4.4.5.2
A5-1	978	2,896	955		31,040	35,474	3,400	1,052		0.124	35.3	DESIGN LOAD FACTOR VARIATION 3.5g STRUCT., 3.5g TPS	4.4.4.1
A5-2	1,040	3,260	1,075		33,940	38,950	3,880	1,130		0.127	37.4	5.0g STRUCT., 5.0g TPS	4.4.4.1
A5-3	1,042	3,275	1,080		34,200	39,245	3,900	1,145		0.128	37.5	7.33g STRUCT., 3.5g TPS	4.4.4.1
A5-4	1,094	3,579	1,180		35,820	41,320	4,300	1,200		0.133	37.7	7.33g STRUCT., 5.0g TPS	4.4.4.1
A7	1,060	3,427	1,130		38,850	43,543	4,850	43		0.112	41.0	THERMO PROTECTION SYS. VARIATION (HOT STRUCT) - 10 SEC CRUISE	4.4.4.3
A8	1,040	3,260	1,075		33,900	38,910	3,850	1,160		0.129	37.4	FUEL TANK VARIATION NON-INTEGRAL TANK	4.4.4.2
A9	942	2,684	885		32,217	38,480	1,788 LM2 2,345 JP	1,130	LM2-RJ JP-TJ	0.162	40.2	FUEL VARIATION TURBO-JETS USE JP	4.4.3.1
REVISION A	1,145	3,882	1,280	3.25	43,000	48,980	4,540	1,440	LM2	0.122	42.8	BASELINE CONFIG. REVISION	4.3

(Presented in International Standard Units)

TRADEOFF DESIGNATION	Sp m ²	VOLUME TOTAL m ³	VOLUME FUEL m ³	(L/D)max	OWE kg	TOGW kg	PROPPELLANT WT. ACCEL. & CRUISE CLIMB kg kg		FUEL kg	W _F /W _{TO}	(W/Sp) _{TO} kg/m ²	TRADEOFF DESCRIPTION	SECTION REFERENCE
							CLIMB	CRUISE					
A	93.27	86.26	28.44	3.25	14,760	16,884	1620	503	LM2	0.126	181.1	BASELINE CONFIGURATION	4.3
A3-1	113.8		40.47		15,808	18,833	1964	1061		0.161	166.0	TEST TIME VARIATION t = 10 min	4.4.5.2
A5-1	90.9		27.0		14,080	16,091	1542	477		0.124	177.1	DESIGN LOAD FACTOR VARIATION 3.5g STRUCT, 3.5g TPS	4.4.4.1
A5-2	96.7		30.42		15,395	17,667	1760	513		0.127	182.5	5.0g STRUCT, 5.0g TPS	4.4.4.1
A5-3	97.0		30.56		15,513	17,801	1769	519		0.128	183.4	7.33g STRUCT, 3.5g TPS	4.4.4.1
A5-4	101.8		33.39		16,248	18,742	1950	544		0.133	184.6	7.33g STRUCT, 5.0g TPS	4.4.4.1
A7	98.6		31.98		17,622	19,751	2200	19.5		0.112	200.6	THERMO PROTECTION SYS. VARIATION (HOT STRUCT)-10 sec CRUISE	4.4.4.3
A8	96.7		30.42		15,377	17,649	1756	526		0.129	182.5	FUEL TANK VARIATION NON-INTEGRAL TANK	4.4.4.2
A9	87.6		25.05		14,613	17,454	811 LM2 1064 JP	513	LM2-RJ JP-TJ	0.162	198.6	FUEL VARIATION TURBOJETS USE JP	4.4.3.1
REVISION A	106.4	109.9	36.25	3.25	19,505	22,217	2059	653	LM2	0.122	209.0	BASELINE CONFIG. REVISION	4.3

**(U) FIGURE 4-140
TRADE-OFF RESUME
B232 Configuration**

TRADE-OFF DESIGN DESIGNATION MACH NO.		S P ft ²	VOLUME TOTAL ft ³	VOLUME FUEL ft ³	VOL FUEL ft ³	2/3 P ft ³	(L/D) _{MAX}	AIR LAUNCH ALL BODY CONFIG. LH ₂ FUEL ROCKET ENGINE FOR ACCEL. & CLIMB SCRAM-JET FOR CRUISE				TRADE-OFF DESCRIPTION	SECTION REFERENCE	
								ONE	TOW	PROPELLANT WT.	W _F /W _{TO} (W/S) _P TO			
								ACCEL & CRUISE CLIMB-lb	lb	lb/ft ²				
A	12	947	4,260	2,416	0.278	2.98	26,730	78,010	49,900	1,380	0.658	82.4	BASE-LINE CONFIGURATION	4.3
B	↓	1,187	6,000	3,537	↓	↓	31,870	100,449	66,900	1,679	0.684	84.7	OFF-SHELF ENGINE (A) ENG. T/W = 1.359 RL-10 (MOD)	4.4.1

(Presented in International Standard Units)

TRADE-OFF DESIGNATION	DESIGN MACH NO.	S_p m^2	VOLUME TOTAL m^3	VOLUME FUEL m^3	VOL $\frac{2/3}{P}$	$\frac{L}{D}_{MAX}$	OWE kg.	TOGW kg.	PROPELLANT WT. ACCEL & CRUISE CLIMB kg	W_F/W_{TO} kg.	$(W/S_p)_{TO}$ kg/m^2	TRADE-OFF DESCRIPTION	SECTION REFERENCE	
A	12	88.0	121	68.43	0.278	2.98	12,125	35,385	22,634	626	0.658	402.3	BASE-LINE CONFIGURATION	4.3
B	↓	110.3	170	100	↓	↓	14,156	45,563	30,345	762	0.684	413.5	OFF-SHELF ENGINE (6) ENG. T/W = 1.359 RL-10 (MOD)	4.4.1

(U) FIGURE 4-141
TRADE-OFF RESUME
B233 Configuration

AIR LAUNCH ALL BODY CONFIG. LH ₂ FUEL													SECTION		
TRADE-OFF DESIGNATION	DESIGN MACH NO.	S P ft ²	VOLUME TOTAL ft ³	VOLUME FUEL ft ³	VOL ^{2/3} _P	(L/D) _{MAX}	ROCKET ENGINE					W _P /W _{TO}	(W/S _P) _{TO} lb/ft ²	TRADE-OFF DESCRIPTION	REFERENCE
							OWE lb	TOGW lb	ACCEL & CLIMB lb	CRUISE lb	PROPELLANT WT.				
A	12	929	3,840	2,276	0.263	3.10	24,600	77,950	49,030	4,320	0.685	84.0	BASE - LINE CONFIGURATION	4.3	
A1-1	8	570	1,837	982		3.70	17,450	40,460	14,900	3,110	0.570	71.0	TEST MACH NO. VARIATION	MACH NO. = 8 4.4.5.1	
A1-2	10	739	2,700	1,484		3.39	20,170	54,990	31,240	3,580	0.635	74.5		MACH NO. = 10 4.4.5.1	
A2-1	12	940	3,880	2,326		3.10	24,560	79,160	50,290	4,310	0.690	84.3	THRUST TO WEIGHT VARIATION	T/W = 1.25 4.4.2.2	
A2-2		932	3,855	2,300			25,080	79,080	49,590	4,410	0.682	85.0		T/W = 2.00 4.4.2.2	
A3-1		1,239	5,830	3,540			30,700	113,790	71,300	11,790	0.731	91.9	TEST TIME VARIATION	t = 10 MIN 4.4.5.2	
A3-2		1,815	10,400	5,920			41,800	180,820	112,800	26,220	0.770	99.5		t = 15 MIN 4.4.5.2	
A4-1		912	3,717	2,202			23,880	75,650	47,570	4,200	0.684	82.9	PAYLOAD WT. VARIATION	600 lb 4.4.5.3	
A4-2		947	3,924	2,340			25,320	80,260	50,490	4,450	0.685	84.8		1400 lb 4.4.5.3	
A4-3		925	3,785	2,265			24,550	77,770	48,910	4,310	0.684	84.1	PAYLOAD DENSITY VARIATION	25 lb/ft ³ 4.4.5.3	
A4-4		921	3,800	2,265			24,500	77,590	48,790	4,300	0.685	84.3		30 lb/ft ³ 4.4.5.3	
A5-1		921	3,770	2,242			23,980	76,590	48,400	4,210	0.687	83.2	DESIGN LOAD FACTOR VARIATION	3.5g STRUCT., 3.5g TFS 4.4.4.1	
A5-2		959	4,000	2,412			26,000	82,550	51,975	4,575	0.674	86.1		5.0g STRUCT., 5.0g TFS 4.4.4.1	
A5-3		944	3,903	2,350			25,290	80,440	50,700	4,450	0.687	85.4		7.3g STRUCT., 3.5g TFS 4.4.4.1	
A5-4		976	4,110	2,489			26,860	85,260	53,650	4,750	0.685	87.4		7.3g STRUCT., 5.0g TFS 4.4.4.1	
A6		2,114	13,100	7,490			46,190	222,120	138,000	37,930	0.791	105.1	SINGLE BASE OPERATION	4.4.6.1	
A7-1		841	3,295	1,442			20,910	67,210	42,630	3,670	0.689	80.0	THERMAL PROTECTION SYS. VARIATION	ACTIVE COOLING 4.4.4.3	
A7-2		996	4,245	2,290			24,190	77,830	49,400	4,240	0.690	78.0		PASSIVE INSULV. 4.4.4.3	
A9		970	4,070	2,457			25,500	80,911	50,944	4,467	0.685	83.4	FUEL VARIATION	BBP FUEL 4.4.3.2	
A11-1		1,164	5,352	3,318			28,790	106,690	66,945	5,955	0.730	91.5	LANDING ASSIST	5,000 lb PROPELLANT 4.4.6.2	
A11-2		1,400	7,070	4,350			33,030	135,030	84,400	7,600	0.755	96.5		10,000 lb PROPELLANT 4.4.6.2	
B		1,202	5,630	3,478			28,660	101,910	67,940	5,310	0.720	84.6	OFF-SHELF ENGINE	(5) ENG. T/W = 1.117 4.4.2.2	
C-1		1,175	5,410	3,360			28,680	99,580	65,590	5,310	0.713	84.7	RL-10 (MOD)	(6) ENG. T/W = 1.375 4.4.2.2	
C-2		1,200	5,590	3,470			29,730	102,830	67,600	5,500	0.713	85.3		(7) ENG. T/W = 1.55 4.4.2.2	
D		1,110	4,710	2,686	0.253	3.20	28,220	91,220	58,240	4,760	0.690	82.1	WING L.S. SWEPT VARIATION	75 DEG. 4.4.2.1	
E		1,005	4,360	2,484	0.265	3.10	26,380	84,730	53,720	4,630	0.689	84.3		77 DEG. 4.4.2.1	

MCDONNELL AIRCRAFT

4-172

(U) FIGURE 4-141 (Continued)
TRADE-OFF RESUME
B233 Configuration

(Presented in International Standard Units)

TRADEOFF DESIGN DESIGNATION	MACH NO.	Sp	VOLUME TOTAL	VOLUME FUEL	VOLUME 2/3 Sp	W/D MAX	ONE	TOGW	PROPELLANT WT.	Wp/WTO	(W/Sp) TO	TRADEOFF DESCRIPTION	SECTION REFERENCE
		m ²	m ³	m ³			kg	kg	ACCEL. & TIME-kg	CRUISE kg	kg/m ²		
A	12	86.3	108.7	64.51	0.263	3.10	11,158	35,356	22,240	1,958	0.685	BASE-LINE CONFIGURATION	4.4.5.1
A1-1	8	53.0	51.99	27.8		3.70	7,915	18,352	9,026	1,510	0.570	TEST MACH NO. VARIATION MACH NO. = 8	4.4.5.1
A1-2	10	68.7	76.41	42.00		3.32	9,149	24,943	14,170	1,624	0.635	MACH NO. = 10	4.4.5.1
A2-1	12	87.3	109.8	65.83		3.10	11,140	35,906	22,811	1,955	0.690	THRUST TO WEIGHT VARIATION T/W = 1.25	4.4.2.2
A2-2		86.6	109.1	65.09			11,376	35,870	22,494	2,000	0.682	T/W = 2.00	4.4.2.2
A3-1		115.1	155.0	100.2			13,925	51,614	32,341	5,347	0.731	TEST TIME VARIATION t = 10 MIN.	4.4.5.2
A3-2		168.6	294.32	167.5			18,960	82,019	51,165	11,891	0.770	t = 15 MIN.	4.4.5.2
A4-1		84.7	105.2	62.32			10,832	34,314	21,577	1,905	0.684	PAYLOAD WT. VARIATION 276.2 kg	4.4.5.3
A4-2		88.0	111.0	66.22			11,385	36,405	22,902	2,018	0.685	335.0 kg	4.4.5.3
A4-3		85.9	107.1	64.10			11,136	35,276	22,185	1,955	0.684	PAYLOAD DENSITY VARIATION 400.2 kg/m ³	4.4.5.3
A4-4		85.6	107.5	64.10			11,113	35,194	22,131	1,950	0.685	480.6 kg/m ³	4.4.5.3
A5-1		85.6	106.7	63.45			10,877	34,741	21,954	1,909	0.687	DESIGN LOAD FACTOR VARIATION 3.5g STRUCT. 3.5gTFS	4.4.4.1
A5-2		89.1	113.2	68.26			11,793	37,444	23,575	2,075	0.674	5.0g STRUCT. 5.0gTFS	4.4.4.1
A5-3		87.7	110.5	66.51			11,471	36,487	22,992	2,018	0.687	7.3g STRUCT. 7.3g TFS	4.4.4.1
A5-4		90.7	116.3	70.44			12,183	38,673	24,330	2,154	0.685	7.3g STRUCT. 5.0g TFS	4.4.4.1
A6		196.4	370.7	212.0			20,951	100,752	62,582	17,201	0.791	SINGLE BASE OPERATION	4.4.6.1
A7-1		78.1	93.25	40.81			9,485	30,486	19,332	1,664	0.689	THERMAL PROTECTION SYS. VARIATION ACTIVE COOLING	4.4.4.3
A7-2		92.5	120.1	64.81			10,972	35,303	22,403	1,923	0.690	PASSIVE INSULATION	4.4.4.3
A9		90.1	115.2	69.53			11,567	36,701	23,103	2,026	0.685	FUEL TEMPERATURE VARIATION MBP MTEL	4.4.3.2
A11-1		108.1	151.5	93.90			13,059	48,394	30,359	2,791	0.730	LANDING ASSIST 2,468 kg PROPELLANT	4.4.6.2
A11-2		130.1	200.1	123.1			14,982	61,249	38,275	3,447	0.755	1,535.9 kg PROPELLANT	4.4.6.2
B		111.7	159.3	98.45			13,000	46,226	30,810	2,408	0.720	OFF-SHELF ENGINE (5) ENG. T/W = 1.117	4.4.2.2
C-1		109.2	153.1	95.09			13,009	45,169	29,745	2,408	0.713	RL-10 (MED) (6) ENG. T/W = 1.375	4.4.2.2
C-2		111.5	158.2	98.20			13,485	46,643	30,656	2,494	0.713	(7) ENG. T/W = 1.55	4.4.2.2
D		103.1	113.3	76.01	0.253	3.20	12,800	41,377	26,411	2,159	0.690	WING L/E. SWEEP VARIATION ALE = 75 DEG.	4.4.2.3
E		93.4	123.4	70.30	0.265	3.10	11,966	38,433	24,362	2,100	0.684	ALE = 77 DEG.	4.4.2.3

**(U) FIGURE 4-142
TRADE-OFF RESUME
B257 Configuration**

TURBO-JET (MACH 0-3) CONVERTIBLE SCRAM-JET (MACH 1-12)

TRADE-OFF DESIGNATION	DESIGN MACH NO.	TYPE CONFIG.	S_p ft. ²	VOLUME TOTAL ft. ³	VOLUME FUEL ft. ³	(L/D) _{MAX}	OWE lb.	TOW lb.	PROPELLANT WT. ACCEL. & CRUISE CLIMB LB.	FUEL	NO. OF TURBINES TJ ENG.	W/F_{TO}	$(W/S_p)_{TO}$ lb/ft ²	TRADE - OFF DESCRIPTION	SECTION REFERENCE	
F	12	BWB	2,860	17,400	7,560	3.69	82,600	117,830	32,000	3,230	LH ₂	5	0.299	41.2	BASE-LINE CONFIGURATION	4.3
F1-1	10		1,780	8,500	3,690	3.93	60,000	77,200	15,400	1,800		4	0.223	43.4	TEST MACH NO. VARIATION MACH NO=10	4.4.5.1
F2-1	12		2,760	16,500	6,980	3.69	87,600	119,847	29,000	3,247		6	0.269	43.4	THRUST TO WT. VARIATION T/W=1.17	4.4.2.2
F2-2			2,250	12,000	5,200		78,000	119,500	38,900	2,600	LH ₂ /JP	5	0.347	53.1	T/W=0.97	4.4.2.2
F2-3			2,540	14,400	6,000		88,000	131,200	39,950	3,250		6	0.329	51.7	T/W=1.03	4.4.2.2
F3-1			3,440	22,900	9,700		99,000	144,202	37,202	8,000	LH ₂	6	0.313	41.9	TEST TIME VARIATION t=10 MIN	4.4.5.2
F3-2			3,100	19,600	8,240		93,200	131,600	33,400	5,000		6	0.292	42.5	t=7.5MIN.	4.4.5.2
F5-1			2,530	14,400	6,320		76,660	106,120	26,335	3,125		5	0.278	41.9	LOAD FACTOR VARIATION 3.5g STRUCT. 3.5g TPS	4.4.4.1
F5-2			2,900	17,700	7,500		93,100	128,050	31,450	3,500		6	0.273	44.2	5.0g STRUCT. 5.0g TPS	4.4.4.1
F5-3			2,840	17,100	7,223		90,470	124,170	30,020	3,680		6	0.271	43.7	7.33g STRUCT. 3.5g TPS	4.4.4.1
F5-4			3,000	18,600	7,900		96,800	133,575	33,275	3,500		6	0.275	44.5	7.33g STRUCT. 5.0g TPS	4.4.4.1
F9-1			1,980	9,900	4,600		68,400	105,310	34,710	2,200	LH ₂ /JP	4	0.350	53.3	FUEL VARIATION JP FUEL FOR TJ	4.4.3.1
F9-2			2,920	17,800	7,560		90,000	123,415	30,000	3,415	LH ₂	6	0.271	42.3	MRP FUEL	4.4.3.2
F9-3			2,280	12,200	5,300		74,200	101,177	24,477	2,500		5	0.267	44.4	SLUSH FUEL	4.4.3.2
D		AB	3,900	21,700	7,580	3.76	112,000	173,820	58,300	3,520	LH ₂ /JP	7	0.356	44.6	WING L.E. SWEEP VAR.	77 DEG 4.4.2.1
E			3,040	16,150	7,540	3.60	96,200	161,821	62,521	3,100		6	0.406	53.2		75 DEG 4.4.2.1

(Presented in International Standard Units)

HORIZONTAL TAKEOFF TURBOJET (MACH 0-3) CONVERTIBLE SCRAMJET (MACH 1-12)

TRADEOFF DESIGN		TYPE	S _p m ²	VOLUME	VOLUME	(L/D) max	OWE	TOW	PROPELLANT WT.	FUEL	NO. OF TJ ENGINES	W/F _{TO}	P/W _{TO} kg/m ²	TRADEOFF DESCRIPTION	SECTION REFERENCE	
DESIGNATION	MACH NO.	CONFIG.		TOTAL m ³	FUEL m ³		kg	kg	ACCEL & CLIMB kg							CRUISE kg
F	12	BWB	266	492	214	3.69	37,466	53,447	14,515	1465	LH ₂	5	0.299	201.2	BASELINE CONFIGURATION	4.3
F1-1	10		165	241	104	3.93	27,216	35,017	6,985	816	↓	4	0.223	211.9	TEST MACH NO. VARIATION MACH NO. = 10	4.4.5.1
F2-1	12		256	467	196	3.69	39,735	54,362	13,154	1473	↓	6	0.269	211.9	THRUST TO WT. VARIATION T/W = 1.17	4.4.2.2
F2-2			209	340	147		35,380	54,204	17,645	1179	LH ₂ /JP	5	0.347	259.3	T/W = 0.97	4.4.2.2
F2-3			236	408	170		39,916	59,511	18,121	1474	↓	6	0.329	252.4	T/W = 1.03	4.4.2.2
F3-1			320	648	275		44,906	65,409	16,785	3629	LH ₂	6	0.313	204.6	TEST TIME VARIATION t = 10 min	4.4.5.2
F3-2			288	555	233		42,275	59,693	15,150	2268	↓	6	0.292	207.5	t = 7.5 min	4.4.5.2
F5-1			235	408	172		34,772	48,135	11,945	1417		5	0.278	204.6	LOAD FACTOR VARIATION 3.5g STRUCT, 3.5g TPS	4.4.4.1
F5-2			269	501	212		42,229	58,083	14,265	1588		6	0.273	215.8	5.0g STRUCT, 5.0g TPS	4.4.4.1
F5-3			264	484	204		41,037	56,323	13,617	1669		6	0.271	213.4	7.33g STRUCT, 3.5g TPS	4.4.4.1
F5-4			279	526	224		43,908	60,589	15,093	1588		6	0.275	217.3	7.33g STRUCT, 5.0g TPS	4.4.4.1
F9-1			184	280	130		31,026	47,768	15,744	998	↓	4	0.350	260.2	FUEL VARIATION JP FUEL FOR TJ	4.4.3.1
F9-2			271	504	214		40,823	55,980	13,608	1549	LH ₂	6	0.271	206.5	MRP FUEL	4.4.3.2
F9-3			212	345	150		33,857	45,893	11,103	1134	↓	5	0.267	216.8	SLUSH FUEL	4.4.3.2
D		AB	362	614	215	3.76	50,802	78,843	26,444	1597	LH ₂ /JP	7	0.356	217.8	WING L.E. SWEEP VAR. A _{LE} = 77 DEG.	4.4.2.1
E			282	457	213	3.60	43,636	72,401	28,359	1406	↓	6	0.406	259.7	A _{LE} = 75 DEG.	4.4.2.1

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**(U) FIGURE 4-143
TRADE-OFF RESUME
B260 Configuration**

HTO/VTO ALL BODY CONFIG. LH₂ FUEL
ROCKET ENGINE

TRADE-OFF DESIGN DESIGNATION MACH NO.	S _p	VOLUME TOTAL ft. ³	VOLUME FUEL ft. ³	$VOL^{2/3}/S_p$	(L/D) _{MAX}	OWE	TOGW	PROPELLANT WT. ACCEL & CRUISE CLIMB LB LB	W_F/W_{TO}	(W/S_p) _{TO}	TRADE-OFF DESCRIPTION	SECTION REFERENCE
	ft. ²					lb	lb			lb/ft ²		
A	12	1,405	6,990	4,655	0.260	3.14	37,900	147,250 106,800 6,550	0.742	105.0	BASE-LINE CONFIGURATION HTO	4.3
A/VTO	12	1,405	6,990	4,655		3.14	37,900	147,250 105,640 3,710	0.742	105.0	VTO t = 2.83 MIN	
A1-1	10	1,025	4,350	2,810		3.47	28,600	94,700 61,100 5,000	0.700	92.5	TEST MACH NO. VARIATION MACH NO = 10	4.4.5.1
A1-2	8	810	3,050	1,890		3.62	24,650	68,900 39,800 4,450	0.643	85.0	MACH NO = 8	4.4.5.1
A6	12	2,785	19,500	12,780		3.14	65,550	364,550 248,500 50,480	0.820	130.8	SINGLE BASE OPERATION	4.4.6.1
B	↓	1,708	9,380	6,470		↓	44,800	189,050 136,100 8,150	0.763	110.8	OFF-SHELF ENGINE (1) ENG. T/W = 1.40 J2S	4.4.1

(Presented in International Standard Units)

TRADE-OFF DESIGN DESIGNATION MACH NO.	S _p	VOLUME TOTAL m ³	VOLUME FUEL m ³	$VOL^{2/3}/S_p$	(L/D) _{MAX}	OWE	TOGW	PROPELLANT WT. ACCEL & CRUISE CLIMB kg kg	W_F/W_{TO}	(W/S_p) _{TO}	TRADE-OFF DESCRIPTION	SECTION REFERENCE
	m ²					kg	kg			kg/m ²		
A	12	130.5	197.8	131.7	0.260	3.14	17,191	66,791 46,629 2,971	0.742	572.7	BASE-LINE CONFIGURATION HTO	4.3
A/VTO	12	130.5	197.8	131.7		3.14	17,191	66,791 47,917 1,683	0.742	572.7	VTO t = 2.83 MIN	
A1-1	10	95.22	123.1	79.52		3.47	12,972	42,955 27,214 2,268	0.700	451.6	TEST MACH NO. VARIATION MACH NO = 10	4.4.5.1
A1-2	8	75.25	86.3	53.49		3.62	11,181	31,253 18,053 2,018	0.643	411.0	MACH NO = 8	4.4.5.1
A6	12	258.7	551.85	361.67		3.14	29733	165,357 112,716 22,897	0.820	634.6	SINGLE BASE OPERATION	4.4.6.1
B	↓	158.7	265.5	183.1		↓	20,321	85,752 61,734 3,697	0.763	541.0	OFF-SHELF ENGINE (1) ENG. T/W = 1.40 J2S	4.4.1

(U) FIGURE 4-144
TRADE-OFF RESUME
B284 Configuration

TRADE-OFF DESIGN DESIGNATION MACH NO.	S_p ft ²	VOLUME TOTAL ft ³	VOLUME FUEL ft ³	$VOL^{2/3}/S_p$ (L/D) _{MAX}	ONE lb	TOW lb	AIR LAUNCH ALL BODY CONFIG. LH ₂ FUEL ROCKET ENGINE UNBURNED		w_p/w_{TO}	$(w/s_p)_{TO}$ lb/ft ²	TRADE-OFF DESCRIPTION	SECTION REFERENCE
							PROPELLANT WT. ACCEL. & CRUISE CLIMB lb	CLIMB lb				
A	12	881	2,190	0.277	3.00	23,190	47,180	4,220	0.688	84.7	BASE-LINE CONFIGURATION	4.3
B	↓	1,182	3,398	0.285	2.92	27,810	66,020	5,480	0.720	83.9	OFF-SHELF ENGINE (5) ENG. T/W = 1.145 RL-10 (MOD)	4.4.1

(Presented in International Standard Units)

TRADE-OFF DESIGN DESIGNATION MACH	S_p m ²	VOLUME TOTAL m ³	VOLUME FUEL m ³	$VOL^{2/3}/S_p$ (L/D) _{MAX}	ONE kg.	TOW kg.	PROPELLANT WT.		w_p/w_{TO}	$(w/s_p)_{TO}$ kg./m ²	TRADE-OFF DESCRIPTION	SECTION REFERENCE
							ACCEL. & CRUISE CLIMB kg.	CLIMB kg.				
A	12	11.3	21.93	0.277	3.00	10,519	21,686	1,914	0.688	113.5	BASE-LINE CONFIGURATION	4.3
B	↓	169.3	96.16	0.285	2.92	12,614	29,946	2,436	0.720	109.6	OFF-SHELF ENGINE (5) ENG. T/W = 1.145 RL-10 (MOD)	4.4.1

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Due to the large number of configurations and tradeoff studies, the detail weight discussion below is limited to the B233A (a M = 12.0 air launched LR-129 rocket boost/cruise configuration) as being representative of the seven configurations.

(U) Changes in design criteria/ground rules were made in Phase II that increased the weight of the Phase I vehicles. The Phase II vehicles used precooled hydrogen fuel, while NBP hydrogen was used for Phase I. A passive thermal protection system was incorporated in Phase II instead of an active water wick system, which was used in Phase I. A 1% I_{sp} degradation for Phase II rocket performance was used while a degradation of 5% was used for Phase I. The following comparison of the M = 12.0 233 air launched, LR-129 rocket boost/cruise configuration illustrates the increments of this weight growth:

ITEM	CRITERIA		Δ WEIGHT From 224-233 lb/kg
	224-233 Phase I	B233A Phase II	
OWE (lb/kg)	21980 (9970)	24600 (11159)	3620 (1189)
Fuel	NBP	Pre-cooled	-900 (-408)
Thermal Protection System	Active	Passive	3690 (1674)
Rocket I_{sp} Degradation	5%	1%	-970 (-440)
Additional Propellant Volume	--	34 ft ³ /.96 m ³	800 (363)

(U) Detail balance calculations were performed on all Phase II base configurations. The aft aerodynamic C.G. limit of the B233A configuration was set at 65.5% of total body length with the C.G. at launch weight located at this aft limit. A 2.0% positive static margin is provided. The C.G. has moved forward 2.7% at end of cruise and landing. Similar aerodynamic C.G. limits were determined (with 2.0% positive static margin) for all baseline configurations and suitable balance conditions were determined.

(U) In addition to a responsive and proper weight and balance evaluation of the Phase II vehicles, a correct material distribution is required for airframe (investment cost) evaluation. The structural material distribution used in the weight/cost analysis was allocated as shown below:

<u>GROUP</u>	<u>MATERIAL</u>	<u>APPLICATION</u>
Wing/Body	Aluminum	Skin, stringers, frames, longerons, etc.
	High Strength Steel	High temperature bulkheads, ftgs., etc.
	Titanium	Bulkheads, fittings, etc.
	Other	Windshield glass, etc.
Movable Wing Tips	Rene' 41	M=6 Basic Structure
	TD-NiCr	M=6 Leading Edges & M=12 Basic Structure
	Columbium	M=12 Leading Edge

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Vertical Tail	Titanium	M=6 Basic Structure
	Rene' 41	M=6 Leading Edge and Rudder
	TD NiCr	M=12 Basic Structure (Incl. L.E. & Rudder)
Landing Gear	High Strength Steel	Shock struts, trunnions, etc.
	Conventional	Wheels, brakes, and tires
	Materials	
	Steel	Main landing gear skids

The B233A air launched configuration utilizes steel main landing gear skids. Conventional wheels, tires, and brakes are used where applicable, sized to the design criteria described in Section 4.1.

(U) The weight of the instruments, cockpit furnishings, and hydraulic and electrical systems are based on existing technology. The electronics system weight is 715 lb (324 kg) installed which does not include any test monitoring equipment (considered payload weight). The environmental control system and the auxiliary power systems are based on an integrated cryogenic APU-cooling system similar to that proposed for the X-20 Dyna-soar.

(U) Figures 4-145 through 4-151 present the material distribution for all baseline vehicles. Note that a 2% (weight empty) contingency is included. Allowances for vent gases and helium pressurant are also included. Figure 4-152 compares the Material Weight Breakdowns for the B233 configuration tradeoff studies. Figures 4-153 through 4-156 present group weight statements of the 65 configurations sized for the Phase II program. In addition to the group weight statements, key characteristics are also presented including wing planform area, type and number of engines, and type of fuel and oxidizer. Noted at the bottom of each figure is a description of the type of tradeoff study performed. Ground rules for the base configurations B207 and B212 presented on Figure 4-153 are:

	Configuration B207	Configuration B212
Cruise Speed	M = 6.0	M = 6.0
Cruise Time	5 min	5 min
T/W (TOGW)	1.5	1.5
Base Operation	Dual	Dual
Boost Engine	One Rubberized LR-129	Two Hardware GE-F100 TJ
Cruise Engine	Two-dimensional RJ	Two-dimensional RJ
Inlet Design	Hot Structure	Hot Structure
Structure Load Factor	5.0 g	5.0 g
TPS Load Factor	3.5 g	3.5 g
Tank Design	Integral	Integral
Boost Propellant	LOX/LH ₂	LH ₂

(U) FIGURE 4-145
MATERIAL WEIGHT BREAKDOWN - lb
B207A Configuration Base Vehicle

Manned, Air Launched, All Body Rocket/Vehicle

Group	STRUCTURE						SYSTEMS/EQUIPMENT				SHIELDS			ROSE CONE/INSULATION					TOTAL
	Alum Alloys	Cb Alloys	Super Alloys	Titanium Alloys	Other	Wheels Tires Brakes	Hyd. Equip.	Equip.	Expend	Titanium	Rein. Al	T-DNiCr	Cb	Tantalum	Rose Cone	Insul.	Foam	Wick	
Wing/Body	2656		300	770	520					223	765	188			80	1049	112		6753
Rotating Tips			440																440
Vertical Tail(4)			595																595
MLG			142			50	72												264
MLG			505				27												532
Control Actuation							315												315
Engine Section																			=
Propulsion:																			
(1) X-129								1040											1040
(1) P-D RJ								3140											3140
Air Induction		2560	660	900			90									390			4600
Fuel System								420											420
APU (Incl Tankage)								465											465
Engine Controls								70											70
Lube Cooling																			=
Instruments								175											175
Hydraulics							278												278
Electrical								300											300
Electronics								715											715
Furnishings	20							330								50			400
ECS								250											250
Contingency	210		87	126															423
Ballast																			
Weight Empty																			21175
Crew & Equipment								240											240
Payload								1000											1000
Vent Prop./Oxidiser									65										65
Wetium Pressurant									70										70
APU Fuel									310										310
OME																			22860
Fuel (LH2)									2800										2800
Oxidiser (LOX)									12770										15175
TOGW	2886	2560	2819	1796	520	50	782	8145	16015	223	765	188			80	1489	112		38430

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(U) FIGURE 4-145 (Continued)
MATERIAL WEIGHT BREAKDOWN
B207A Configuration Base Vehicle
(Presented in International Standard Units - kg)

Group	STRUCTURE					SYSTEMS/EQUIPMENT				SHIELDS					NOSE CONE/INSULATION				TOTAL
	Alum. Alloys	Cu Alloys	Super Alloys	Titanium Alloys	Other	Wheels/Tires/Brakes	Hyd. Equip.	Equip.	Expend.	Titanium	Rene 41	T-DBiCr	Cu	Tantalum	Nose Cone	Insul.	Foam	Wick	
Wing/Body	1205		177	349	236					101	347	85			36	476	51		3063
Rotating Tids			200																200
Vertical Tail(s)			270																270
HLG			64			23	33												120
MLG			229				12												241
Control Activation							143												143
Engine Section																			
Propulsion:																			
(1) LR-129								472											472
(1) 2-D RJ								1424											1424
Air Induction		1161	300	408			41									177			2087
Fuel System								191											191
APU (incl. Tankage)								211											211
Engine Controls								32											32
Lube Cooling																			
Instruments								79											79
Hydraulic							126												126
Electrical								136											136
Electronics								324											324
Furnishings		9						149								23			181
ECS								113											113
Contingency		96		39	57														192
Ballast																			
Weight Empty																			9605
Crew & Equipment								109											109
Payload								454											454
Vent Prop./Oxidizer									29										29
Helium Pressurant									32										32
APU Fuel									140										140
OWE																			10369
Fuel (LH2)									1270										1270
Oxidizer (LOX)									5793										5793
TOTV	1310	1161	1279	814	236	23	355	3694	7264	101	347	85			36	676	51		17432

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(U) FIGURE 4-146
MATERIAL WEIGHT BREAKDOWN - 1b
B212A Configuration Base Vehicle

Manned, NTO, Wing Body, Turbojet/Ramjet

Group	STRUCTURE					Wheels Tires Brakes	SYSTEMS/EQUIPMENT					SHIELDS					NOSE CONE/INSULATION				TOTAL
	Alum Alloys	Cp Alloys	Super Alloys	Titanium Alloys	Other		Hyd. Equip.	Equip.	Expend	Titanium	Refr. Al	T-DNiCr	Cp	Tantalum	Nose Cone	Insul.	Foam	Wick			
Wing/Body	2209	1760	590	880	520				190	1644	858			80	1555	97		10383			
Rotating Tips			848															848			
Vertical Tail(s)			1240															1240			
MLG			99			62	40											201			
MLG			441			294	100											835			
Control Actuation							476											476			
Engine Section			158	819														977			
Propulsion:																					
(2) 100% F-100-GE-100								4600										4600			
(2) 2-0 RJ								3705										3705			
Air Induction		1040	312	875			350								957			3534			
Fuel System								379										379			
APU (Incl. Tankage)								481										481			
Engine Controls								70										70			
Lube Cooling				50														50			
Instruments								175										175			
Hydraulics							441											441			
Electrical								300										300			
Electronics								715										715			
Furnishings	20							330							50			400			
ECS								250										250			
Contingency	189		171	253														613			
Ballast																					
Weight Empty																		30673			
Crew & Equipment								240										240			
Payload								1000										1000			
Vent Prop./Oxidiser									85									85			
Helium Pressurant									91									91			
APU Fuel									451									451			
OME																					
Fuel (LM2)									4683									4683			
Oxidiser (LOX)																		-			
TOW	2418	2800	3859	2877	520	356	1407	12245	5310	190	1644	858		80	2562	97		37223			

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(U) FIGURE 4-146 (Continued)
MATERIAL WEIGHT BREAKDOWN
B212A Configuration Base Vehicle
(Presented in International Standard Units- kg)

Group	STRUCTURE						SYSTEMS/EQUIPMENT				SHINGLES			NOSE CONE/INSULATION				TOTAL
	Alum Alloys	Co Alloys	Super Alloys	Titanium Alloys	Other	Wheels Tires Hoses	Hyd. Equip.	Equip.	Expend Titanium	Rene 41	T-DNiCr	Co	Tantalum	Nose Cone	Insul.	Foam	Wick	
Wing/Body	1002	798	268	399	236				86	747	389			36	705	44		1710
Rotating Taps			385															385
Vertical Tail(s)			562															562
MLG			45			28	18											91
MLG			200			134	45											379
Control Actuation							216											216
Engine Section			72	371														443
Propulsion:																		
(2) 100% F-100-GE-100 TV								2087										2087
(2) 2-D NJ								1681										1681
Air Induction		472	142	397			158								434			1603
Fuel System								172										172
APU (Incl. Tankage)								218										218
Engine Controls								32										32
Lube Cooling				23														23
Instruments								79										79
Hydraulic							200											200
Electrical								136										136
Electronics								324										324
Furnishings	9							149							23			181
ECS								113										113
Contingency	86		78	114														278
Ballast																		
Weight Empty																		13913
Grav. & Equipment								109										109
Payload								454										454
Vent Prop./Oxidiser									38									38
Helium Pressurant									41									41
APU Fuel									205									205
OWR																		14760
Fuel (LH2)																		2124
Oxidiser (LOX)									2124									-
TOTAL	1097	1270	1752	1304	236	162	637	5523	2408	86	747	389		36	1162	44		16884

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(U) FIGURE 4-147
MATERIAL WEIGHT BREAKDOWN - lb
B232A Configuration Base Vehicle

Manned, Air Launched, All Body, Rocket/ SJ

Group	STRUCTURE					Wheels Tires Brakes	SYSTEMS/EQUIPMENT			SHIELDS				ROSE CORR/INSULATION				TOTAL	
	Alum Alloys	Cb Alloys	Super Alloys	Titanium Alloys	Other		Hyd. Equip.	Equip.	Expend	Titanium	Reze Al	T-DMAr	Cb	Tantalum	Rose Corr	Insul.	Foam		Wick
Wing Body	4555		220	860	520					114	1012	220	1137		150	3060	220		11968
Rotating Tip		115	1035																1150
Vertical Tail(s)			1530																1530
ELG			161			59	80												300
MLG			555				35												590
Control Actuation							390												390
Engine Section																			
Propulsion:																			
(1) LR-129								1560											1560
(1) CB								3070											3070
Air Induction																			
Fuel System								895											895
APU (Incl. Tankage)								670											670
Engine Controls								90											90
Lube Cooling																			
Instruments								175											175
Hydraulics							360												360
Electrical								300											300
Electronics								715											715
Furnishings	20							330								50			400
ECS								250											250
Contingency	226		94	137															457
Ballast																			
Weight Empty																			24870
Crew & Equipment								240											240
Payload								1000											1000
Vent Prop./Oxidizer									215										215
Helium Pressurant									100										100
APU Fuel									305										305
OWE																			26730
Fuel (LM2)									8500										8500
Oxidizer (LOX)									42780										42780
TOTAL	4701	115	3595	997	520	59	865	9295	51900	114	1012	220	1137	-	150	3110	220		78010

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(U) FIGURE 4-147 (Continued)
MATERIAL WEIGHT BREAKDOWN
B232A Configuration Base Vehicle
(Presented in International Standard Units - kg)

Group	STRUCTURE						SYSTEMS/EQUIPMENT				SHINGLES			NOSE CONE/INSULATION				TOTAL	
	Alum Alloys	Co Alloys	Super Alloys	Titanium Alloys	Other	Wheels Tires Brakes	Hyd. Equip.	Equip.	Expend	Titanium	Rene 41	T-DNiCr	Co	Tantalum	Nose Cone	Insul.	Foam		Wick
Wing/Body	2020		100	590	236					52	459	100	516		68	1388	100		5429
Rotating Tips		52	470																522
Vertical Tail(s)			694																694
MLG			73			27	36												136
MLG			253				15												268
Control Actuation							177												177
Engine Section																			
Propulsion:																			
(1) LR-129								708											708
(1) CSJ								1393											1393
Air Induction																			
Fuel System								406											406
APU (Incl. Tankage)								304											304
Engine Controls								41											41
Lube Cooling																			
Instruments								79											79
Hydraulics							163												163
Electrical								136											136
Electronics								324											324
Furnishings	9							149								23			181
ECS								113											113
Contingency	102		42	63															207
Ballast																			
Weight Empty																			11281
Crew & Equipment								109											109
Payload								454											454
Vent. Prop./Oxidizer									98										98
Helium Pressurant									45										45
AHU Fuel									138										138
OWE																			12125
Fuel (LH ₂)									3855										3855
Oxidizer (LOX)									19405										19405
TOW	2131	52	1632	453	236	27	391	4216	23541	52	459	100	516		68	1411	100		35385

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(U) FIGURE 4-148
MATERIAL WEIGHT BREAKDOWN - 1b
B233A Configuration Base Vehicle

Air Launched, All Body, All Rocket, Base Vehicle

		STRUCTURE					SYSTEMS/EQUIPMENT				SHINGLES					ROSE CORE/INSULATION				TOTAL	
GROUP		Alum. Alloys	Cb Alloys	Super Alloys	Titanium Alloys	Tantalum Alloys	Other	Wheels & Tires	Hyd. Equip.	Equip.	Expend.	Titanium	Rene 41	T-DM;Cr	Cb	Tantalum	Rose Core	Insul.	Foam	Wick	
Wing/Body		4655		220	880		520					80	1030	155	1380		150	3420	238		12728
Rotating Tip			115	1035																	1150
Vertical Tail (a)				1545																	1545
WLG				192				70	98												360
WLG				630					40												670
Control Actuation									403												803
Engine Section																					
Propulsion:										1560											1560
LB-129																					
Air Induction										915											915
Fuel System										660											660
APU (incl. Tankage)										60											60
Engine Controls																					
Lube Cooling										175											175
Instruments									380												380
Hydraulics										300											300
Electrical										715											715
Electronics										330								50			800
Parashutes		20								250											250
RCS																					454
Contingency		228		90	136																
Ballast																					
Weight Empty																					22725
Crew & Equipment										240											240
Payload										1000											1000
Vent. Prop./Oxidizer										240											240
Helium Pressurant										91											91
APU Fuel										304											304
																					24600
Fuel (LH ₂)										7590											7590
Oxidizer (LOX)										45760											45760
TOTAL		4903	115	3712	1016		520	70	921	6205	53985	80	1030	155	1380		150	3470	238		77950

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(U) FIGURE 4-148 (Continued)
MATERIAL WEIGHT BREAKDOWN
B233A Configuration Base Vehicle
(Presented in International Standard Units - kg)

GROUP	STRUCTURE						SYSTEMS/EQUIPMENT				SHINGLES					MOORE CORE/INSULATION				TOTAL
	Alum Alloys	Co Alloys	Super Alloys	Titanium Alloys	Tantalum Alloys	Other	Wheels & Tires	Hyd. Equip.	Equip.	Expend.	Titanium	Rene 41	C-DMCr	Co	Tantalum	Moore Core	Insul.	Foam	Wick	
Wing/Body	2112		100	399		236					36	467	70	626		68	1551	108		5773
Rotating Tips		52	470																	522
Vertical Tail(s)			701				32	44												701
MLG			88					18												164
MLG			286					183												304
Control Actuation																				183
Engine Section																				
Propulsion:																				
LR-129									708											708
Air Induction																				
Fuel System									415											415
APU (Incl. Tankage)									299											299
Engine Controls									27											27
Lube Cooling									79											79
Instruments								173												173
Hydraulics									136											136
Electrical									324											324
Electronics									149								23			181
Furnishings	9								113											113
ECS																				206
Contingency	103		41	62																
Ballast																				
Weight Empty																				10308
Crew & Equipment									109											109
Payload									454											454
Vent Prop./Oxidizer										109										109
Helium Pressurant										41										41
APU Fuel										138										138
OWE																				11159
Fuel (LH ₂)										3443										3443
Oxidizer ² (LOX)										20756										20756
TOTW	2224	52	1686	461		236	32	418	2813	24487	36	467	70	626		68	1574	108		35358

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(U) FIGURE 4-149
MATERIAL WEIGHT BREAKDOWN - lb
B257F Configuration Base Vehicle

Manned, WFO, Blended Wing Body, Turbojet/CSJ

Group	STRUCTURE					SYSTEMS/EQUIPMENT					SHINGLES			MOSE CONE/INSULATION					TOTAL
	Alum Alloys	Cb Alloys	Super Alloys	Titanium Alloys	Other	Wheels Tires Brakes	Hyd. Equip.	Equip.	Expend	Titanium	Rene 41	T-DHICr	Cb	Tantalum	Mose Cone	Insul.	Foam	Wick	
Wing Body	11840		3640	2750	520					917	1225	533	4820	500	150	6250	855		34000
Rotating Tips		365	3285																3650
Vertical Tail(s)			4630																4630
MLG			190			122	58												370
MLG			722			912	216												1850
Control Actuation							1080												1080
Engine Section																			
Propulsion:																			
(5) F-100-6S-100 TJ								13625											13625
(1) CSJ								4420											4420
Air Induction			4230	4700			470												9400
Fuel System								1060											1060
APU (Incl. Damage)								810											810
Engine Controls								150											150
Lube Cooling				200															200
Instruments								175											175
Hydraulics							1170												1170
Electrical								300											300
Electronics								715											715
Furnishings	20							330								50			400
ECS								450											450
Contingency	806		310	484															1600
Ballast																			
Weight Empty																			80055
Crew & Equipment								240											240
Payload								1000											1000
Vent Prop./Oxidiser									700										700
Helium Pressurant									235										235
APU Fuel									370										370
OWE																			82600
Fuel (LM2)										35230									35230
Oxidiser (LOX)																			
TOGW	12666	365	17007	8134	520	1034	2994	23275	36535	917	1225	533	4820	500	150	6300	855		117830

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(U) FIGURE 4-149 (Continued)
MATERIAL WEIGHT BREAKDOWN
B257F Configuration Base Vehicle
(Presented in International Standard Units - kg)

Group	STRUCTURE					SYSTEMS/EQUIPMENT					SHINGLES				NOSE CONE/INSULATION				TOTAL
	Alum Alloys	Cb Alloys	Super Alloys	Titanium Alloys	Other	Wheels Tires Brakes	Hyd. Equip.	Equip.	Expend	Titanium	Rene 41	T-DWICr	Cb	Tantalum	Nose Cone	Insul.	Foam	Wick	
Wing/Body	5370		1651	1247	236					416	556	242	2186	227	68	2835	388		15412
Rotating Tips		166	1490																1616
Vertical Tail(s)			2100																2100
MLG			86			55	27												148
MLG			328			413	98												839
Control Actuation							490												490
Engine Section																			
Propulsion:																			
(5) F-100-G4-100 TJ								6180											6180
(1) CSJ								2005											2005
Air Induction			1919	2132			213												4264
Fuel System								481											481
APU (Incl. Package)								368											368
Engine Controls								68											68
Lube Cooling				91															91
Instruments								79											79
Hydraulics							531												531
Electrical								136											136
Electronics								324											324
Furnishings	9							149								23			181
ECS								204											204
Contingency	365		141	220															726
Ballast																			
Weight Empty																			30313
Crew & Equipment								109											109
Payload								454											454
Vent Prop./Oxidizer									317										317
Helium Pressurant									107										107
APU Fuel									168										168
OWE																			37468
Fuel (LH2)									15980										15980
Oxidizer (LOX)																			
TOGW	5744	166	7715	3690	236	468	1359	10557	16572	416	556	242	2186	227	68	2858	388		53448

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(U) FIGURE 4-150
MATERIAL WEIGHT BREAKDOWN - lb
B260A Configuration Base Vehicle

Manned, ETO, All Body, All Rocket

Group	STRUCTURE						SYSTEMS/EQUIPMENT				SHINGLES				ROSE CORE/INSULATION				TOTAL
	Alum Alloys	Cu Alloys	Super Alloys	Titanium Alloys	Other	Wheels Tires Brakes	Syd. Equip.	Equip.	Expend	Titanium	Rene 41	T-DNiCr	Cu	Tantalum	Rose Core	InnuL	Fomm	Wick	
Wing/Body	7584	310		1240	520					310	1153	366	2238		150	5670	385		19926
Rotating Tips		158	1422																1580
Vertical Tail(s)			2580																2580
MLG			212			160	88												460
MLG			1200			1248	312												2760
Control Actuation							620												620
Engine Section																			-
Propulsion:																			2580
(1) LR-129								2580											-
Air Induction																			1280
Fuel System								1280											740
APU (incl. Tankage)								740											55
Engine Controls								55											-
Lube Cooling																			175
Instruments								175											565
Hydraulics							565												300
Electrical								300											715
Electronics								715								50			400
Furnishings	20							330											250
RCS								250											714
Contingency	353		146	215															
Ballast																			
Weight Empty																			35700
Crew & Equipment								240											240
Payload								1000											1000
Vent. Prop./Oxidizer									460										460
Helium Pressurant									170										170
APU Fuel									330										330
OWB																			37900
Fuel (LH2)									15620										15260
Oxidizer (LOX)									93730										93730
TOGW	7957	158	5870	1455	520	1408	1585	7665	110310	310	1153	366	2238	-	150	5720	385		147250

(U) FIGURE 4-150 (Continued)
MATERIAL WEIGHT BREAKDOWN
B260A Configuration Base Vehicle
(Presented in International Standard Units - kg)

Group	STRUCTURE					SYSTEMS/EQUIPMENT					SHINGLES				NOSE CONE/INSULATION				TOTAL
	All Alloys	Cb Alloys	Super Alloys	Titanium Alloys	Other	Wheels Tires Brakes	Hyd. Equip.	Equip.	Expend.	Titanium	Rene 41	T-DNiCr	Cb	Tantalum	Nose Cone	Insul.	Foam	Wick	
Wing/Body	3440		141	562	236					141	523	166	1015		68	2572	175		9039
Rotating Tips		72	645																717
Vertical Tail(s)			1170																209
MLG			96			73	40												1252
MLG			544			566	142												281
Control Actuation							281												-
Engine Section																			1170
Propulsion:									1170										-
(1) LR-129																			-
Air Induction																			581
Fuel System																			336
APU (Incl. Tankage)																			25
Engine Controls																			-
Lube Cooling																			79
Instruments																			257
Hydraulics							257												136
Electrical																			324
Electronics																23			181
Furnishings																			113
ECS																			324
Contingency	161		66	97															
Ballast																			
																			16194
Weight Empty																			
Crew & Equipment									109										109
Payload									454										454
Vent Prop./Oxidizer										209									209
Helium Pressurant										77									77
APU Fuel										149									149
																			17192
OWE																			
Fuel (LH2)									7085										7085
Oxidizer (LOX)									42516										42516
TOGW	3610	72	2662	659	236	639	720	3476	50036	141	523	166	1015		68	2595	175		66793

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(U) FIGURE 4-151
MATERIAL WEIGHT BREAKDOWN - 1b
B284A Configuration Base Vehicle

Unmanned, Air Launched, All Body, All Rocket

Group	STRUCTURE						SYSTEMS/EQUIPMENT				SHINGLES				MOSE CONE/INSULATION				TOTAL
	Alum Alloys	Cb Alloys	Super Alloys	Titanium Alloys	Other	Steel Pipes Brakes	Hyd. Equip.	Equip.	Expend	Titanium	Rene 41	T-DMICr	Cb	Tantalum	Mose Cone	Inaul.	Foam	Wick	
Wing Body	3944	108	200	810	520					69	1012	215	1373		150	3260	230		11783
Rotating Tip		108	972																1080
Vertical Tail(s)			1470																1470
MLG			189			26	80												244
MLG			525				40												565
Control Actuation							390												390
Engine Section																			-
Propulsion								1540											1540
(1) LH-123																			-
Air Induction																			910
Fuel System								910											910
APU (Incl. Tankage)								745											745
Engine Controls								55											55
Lube Cooling																			-
Instruments							370												370
Hydraulics								450											450
Electrical								1080											1080
Electronics																			-
Purpishings								300											300
ECS																			428
Contingency	290		88	50															-
Ballast																			-
Weight Empty																			23410
Crew & Equipment								1000											1000
Payload									240										240
Vent Prop./Lidizer									90										90
Helium Pressurant									450										450
APU Fuel																			-
OWE																			23190
Fuel (LH2)									7350										7350
Oxidizer (LOX)									44050										44050
Totals	4234	108	1384	860	520	26	889	6080	52180	69	1012	215	1373		150	3260	230		74590

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(U) FIGURE 4-151 (Continued)
MATERIAL WEIGHT BREAKDOWN
B284A Configuration Base Vehicle
(Presented in International Standard Units - kg)

Group	STRUCTURE					SYSTEMS/EQUIPMENT					SHINGLES				NOSE CONE/INSULATION				TOTAL
	Alum Alloys	Cb Alloys	Super Alloys	Titanium Alloys	Other	Wheels Tires Brakes	Hyd. Equip.	Equip.	Expend	Titanium	Rein. 41	T-DWICr	Cb	Tantalum	Nose Cone	Insul.	Foam	Wick	
Wing/body	1789			367	236					31	459	98	623		68	1479	104		5345
Rotating Tips		49	91																490
Vertical Tails(s)			441																667
MLG			59			12	40												111
MLG			238				18												256
Control Actuation							177												177
Engine Section																			-
Propulsion:																			-
(1) LR-129								698											698
Air Induction																			-
Fuel System								413											413
APU (Incl. Tankage)								338											338
Engine Controls								25											25
Lube Cooling																			-
Instruments																			-
Hydraulic							168												168
Electrical								204											204
Electronics								490											490
Furnishings																			-
ECS								136											136
Contingency	131		40	23															194
Ballast																			-
Weight Empty																			9712
Crew & Equipment																			-
Payload								454											454
Vent Prop./Oxidizer									109										109
Helium Pressurant									40										40
APU Fuel									204										204
OWE																			10519
Fuel (LH ₂)									3334										3334
Oxidizer (LOX)									19981										19981
TOGW	1920	49	1536	390	236	12	403	2758	23668	31	459	98	623		68	1479	104		33834

(U) FIGURE 4- 152
MATERIAL DISTRIBUTION COMPARISON - lb
B233 Configuration Trade Offs

All Body, All Rocket Configuration

CONFIGURATION	STRUCTURE						SYSTEMS/EQUIPMENT			SHINGLES				NOSE/CONE INSULATION				TOTAL
	ALUM. ALLOYS	Cu ALLOYS	SUPER ALLOYS	TITANIUM ALLOYS	TANTALUM ALLOYS	OTHER	WIRE, CABLE, RPD AND TIRE EQUIP.	EQUIP.	EXPEND.	TITANIUM SHEET	T-D HICr	Cu	TANTALUM	NOSE CONE	INSULATION	FOAM	WICK	
B233A (Base)	4903	115	3712	1016		520	70	921	6205	53985	80	1030	155	1380	150	3470	238	77950
" A1-1 (M=8.0)	4143		2505	814		520	38	571	4845	23460	92	720	1023	110	1470	149		60460
" A1-2 (M=10.0)	4226	42	2119	941		520	47	872	5520	35345	300	835	1628	115	2270	190		54990
" A2-1 (T/W=1.25)	5012	115	3676	1015		520	66	912	6115	55235	78	1028	155	1380	150	3466	237	79160
" A2-2 (T/W=2.0)	4926	116	3731	1023		520	69	873	6615	54630	126	1030	155	1383	150	3485	239	79080
" A3-1 (10 min. cruise)	6160	153	4785	1354		520	78	1125	7145	83955	147	1030	155	1451	150	4862	320	113790
" A3-2 (15 min. cruise)	7918	226	6763	1654		520	93	1410	8860	140270	390	1030	155	2711	150	8200	470	180820
" A4-1 (600 lb P.L.)	4844	115	3631	994		520	70	912	5762	52389	69	1030	155	1354	150	3420	235	75650
" A4-2 (1000 lb P.L.)	4970	115	3793	1036		520	70	930	6648	55579	92	1030	155	1407	150	3524	241	80260
" A4-3 (P.L.=25 lb/ft ²)	4894	115	3710	1015		520	70	921	6201	53843	77	1030	155	1374	150	3458	237	77770
" A4-4 (P.L.=30 lb/ft ³)	4887	115	3691	1014		520	70	917	6202	53706	77	1027	155	1370	150	3452	237	77590
" A5-1 (Load Factor)	4556	103	3513	1003		520	64	926	6230	53255	5	1030	155	1370	150	3470	240	76590
" A5-2 (Load Factor)	4795	670	3216	1065	250	520	62	953	6415	57240	120	1000	214	1120	150	3565	245	82550
" A5-3 (Load Factor)	4918	142	3940	1040		520	64	934	6350	55980	19	1030	155	1402	150	3550	246	80440
" A5-4 (Load Factor)	5063	690	3546	1098	256	520	68	958	6510	59105	125	1012	215	1126	150	3610	250	85260
" A6 (Single Base)	7380	285	8923	1570		520	163	1591	10510	177420	890	1030	155	3180	150	7810	543	222120
" A7-1 (Active Cooling)	4660	104	3382	909		520	68	855	5970	47495	21	1030	155	1248	150	390	224	67210
" A7-2 (Low p insulation)	5595	124	3780	1075		520	71	934	6200	54280	126	1030	155	1482	150	2040	268	77830
" A9 (HBP Fuel)	5309	120	3851	1041		520	70	940	6325	56066	108	1030	155	1443	150	3540	243	80911
" A11-1 (5000 lb Ldg Fuel)	5424	145	4569	1127		520	80	1058	7250	78670	213	1030	155	1670	150	4330	299	106690
" A11-2 (10000 lb Ldg Fuel)	6042	173	5371	1510		520	81	1103	7815	102915	430	1030	155	2165	150	5210	360	135030
" B (Hardware Engine)	5766	151	4650	1236		520	73	1037	6575	74045	102	1030	155	1750	150	4370	300	101910
" C-1 (T/W=1.375)	5537	145	4547	1276		520	75	1052	6886	71685	102	1030	155	1750	150	4370	300	99580
" C-2 (T/W=1.55)	5855	148	4644	1302		520	76	1102	7247	73915	121	1030	155	1790	150	4465	310	102830
" D (A ₁₀ =75°)	5931	137	4260	1212		520	72	1041	6220	63715	221	1030	155	1693	150	4160	266	91220
" E (A ₁₀ =77°)	5878	104	3315	1057		520	60	913	6370	59045	121	1208	235	1599	150	3860	390	84730

MCDONNELL AIRCRAFT

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(U) FIGURE 4-152 (Continued)
MATERIAL DISTRIBUTION BREAKDOWN
B233 Configuration Trade-Offs
(Presented in International Standard Units - kg)

CONFIGURATION	STRUCTURE						SYSTEMS/EQUIPMENT				SHIELDS					ROSE CONF/INSULATION				TOTAL
	Alum Alloys	Cb Alloys	Super Alloys	Titanium Alloys	Tantalum Alloys	Other	Hyd. Equip.	Equip.	Expend.	Titanium	Rene 41	T-DMCr	Cb	Tantalum	Rose Cone	Insulation	Form	Wick		
B233A (Base)	2224	52	1684	461		236	32	418	2814	24488	36	467	70	626	68	1574	108		35358	
B233A1-1 (M = 8.0)	1879		1136	369		236	17	259	2198	10642	42	327	464		50	667	67		18353	
B233A1-2 (M = 10.0)	1917	19	961	427		236	21	395	2504	16033	136	379	738		61	1030	86		24943	
B233A2-1 (T/W = 1.25)	2273	52	1667	460		236	30	414	2774	25056	35	466	70	626	68	1572	108		35207	
B233A2-2 (T/W = 2.0)	2235	53	1692	464		236	31	396	3001	24784	57	467	70	627	68	1581	109		35871	
B233A3-1 (10 Min. Cruise)	2794	69	2171	614		236	35	510	3241	38083	67	467	70	840	68	2205	145		51615	
B233A3-2 (15 Min. Cruise)	3591	103	3068	750		236	42	640	4019	63626	177	467	70	1230	68	3720	213		82020	
B233A4-1 (600 Lb. P.L.)	2127	52	1647	451		236	32	414	2614	23764	31	467	70	614	68	1551	107		34315	
B233A4-2 (1400 Lb. P.L.)	2254	52	1721	470		236	32	422	3016	25211	42	467	70	638	68	1598	109		36406	
B233A4-3 (P.L.p = 25Lb/Ft.)	2220	52	1683	460		236	32	418	2813	24423	35	467	70	623	68	1569	107		35276	
B233A4-4 (P.L.p = 30Lb/Ft.)	2211	52	1674	460		236	32	416	2813	24361	35	466	70	621	68	1566	107		35195	
B233A5-1 (Load Factor)	2067	47	1594	455		236	29	420	2826	24156	2	467	70	621	68	1574	109		34741	
B233A5-2 (Load Factor)	2175	304	1459	483	113	236	28	432	2910	25965	54	454	97	508	68	1617	111		37445	
B233A5-3 (Load Factor)	2231	64	1787	472		236	29	424	2880	25393	9	467	70	636	68	1610	112		36488	
B233A5-4 (Load Factor)	2296	313	1608	498	116	236	31	435	2953	26810	57	459	98	511	68	1637	113		38674	
B233A6 Single Base	3348	129	4047	712		236	74	722	4767	80478	404	467	70	1442	68	3543	246		100754	
B233A7-1 (Active Cooling)	2114	47	1534	412		236	31	388	2708	21544	10	467	70	566	68	177	102	13	30486	
B233A7-2 (Low g. Insulation)	2538	56	1715	488		236	32	424	2812	24622	57	467	70	672	68	925	122		35304	
B233A9 WBF Fuel	2408	54	1747	472		236	32	426	2869	25432	49	467	70	655	68	1606	110		36701	
B233A11-1 (50000 Lb. Fuel)		66	2072	511		236	36	480	3289	35685	97	467	70	758	68	1964	136		48395	
B233A11-2 (100000 Lb. Fuel)	2741	79	2436	685		236	37	500	3545	46683	195	467	70	982	68	2363	163		61250	
B233B (Hardware Engine)	2616	69	2109	561		236	13	470	2982	33587	46	467	70	794	68	1982	136		46226	
B233C-1 (T/W = 1.375)	2512	66	2063	579		236	34	477	3123	32516	46	467	70	794	68	1982	136		45169	
B233C-2 (T/W = 1.55)	2656	67	2107	591		236	34	500	3287	33528	55	467	70	812	68	2025	141		46644	
B233D (LE = 75°)	2690	62	1936	550		236	36	472	3003	28902	100	467	70	768	68	1887	130		41377	
B233E (LE = 77°)	2666	47	1504	480		236	27	416	2889	26783	55	503	107	725	68	1751	177		38434	
B233J-1 (Hardware Engine)																				

(U) FIGURE 4-153
GROUP WEIGHTS COMPARISON - lb
M = 6.0 Vehicles

B207 - MANNED, AIR LAUNCHED, ALL BODY ROCKET/RAMJET											B212 - MANNED, FPO, WING BODY, TURBOJET/RAMJET									
CONFIGURATION																				
TRADEOFF STUDY	A-REVISED	A	A1-1	A1-2	A1-3	A2-1	A2-2	A-6	B	A-REVISED	A	A3-1	A5-1	A5-2	A5-3	A5-4	A7	A8	A9	
Wing/Body	8057	6753	5983	7138	16232	6758	6763	9504	7219	13760	10383	12062	9122	11485	11681	13259	15851	11623	9777	
Rotating Tips	518	440	430	750	2560	440	443	660	539	848	848	848	705	1018	1028	945	848	848	848	
Vertical Tail(s)	702	595	585	1010	3340	600	602	895	725	1240	1240	1240	1240	1240	1240	1240	1240	1240	1240	
Wing Landing Gear	299	264	260	310	480	266	264	330	298	247	201	224	195	206	207	214	227	206	205	
Main Landing Gear	385	532	525	665	1210	539	532	743	633	823	835	951	804	869	874	909	997	872	854	
Control Actuation	340	315	310	350	500	318	315	390	348	566	476	514	464	487	489	501	532	484	475	
Engine Section	—	—	—	—	—	—	—	—	—	1917	977	977	977	977	977	977	977	977	977	
Propulsion Group	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	
Engine(s)	1160	1040	1010	1150	2020	950	1090	1180	963	4600	4600	4600	4600	4600	4600	4600	4600	4600	4600	
Engine(s)	3240	3140	3150	3250	6050	3170	3150	3140	3350	3705	3705	3705	3705	3705	3705	3705	3705	3705	3705	
Air Induction	6746	4600	4069	4330	9100	4630	4610	4600	4720	8780	3534	3534	3534	3534	3534	3534	3534	3534	3534	
Fuel System	478	420	405	455	870	427	420	515	485	500	379	587	364	392	407	413	465	390	420	
APU	608	465	460	520	740	465	465	567	510	572	481	575	469	492	494	508	534	491	480	
Engine Controls	70	70	70	70	80	70	70	70	70	70	70	70	70	70	70	70	70	70	70	
Lube Cooling	—	—	—	—	—	—	—	—	—	50	50	50	50	50	50	50	50	50	50	
Instruments	175	175	175	175	175	175	175	175	175	175	175	175	175	175	175	175	175	175	175	
Hydraulics	306	278	275	310	475	280	278	356	316	537	441	475	431	450	454	466	489	448	440	
Electrical	300	300	300	300	300	300	300	300	300	300	300	300	300	300	300	300	300	300	300	
Electronics	715	715	715	715	715	715	715	715	715	715	715	715	715	715	715	715	715	715	715	
Furnishings	400	400	400	400	400	400	400	400	400	400	400	400	400	400	400	400	400	400	400	
ECR	250	250	250	250	250	250	250	250	250	250	250	250	250	250	250	250	250	250	250	
Contingency	509	423	389	452	928	423	423	507	449	820	613	612	582	641	644	678	734	640	607	
Weight Empty	25458	21175	19761	22600	46425	21176	21265	25305	22465	40972	30673	32864	29152	32056	32294	33909	36693	32018	30322	
Crew & Equipment	240	240	240	240	240	240	240	240	240	240	240	240	240	240	240	240	240	240	240	
Payload	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	
Vent Fuel/Oxidizer	76	65	60	80	220	64	65	85	80	120	85	120	80	87	101	93	109	86	90	
Trapped Fuel	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	25	
Pressurant	74	70	69	80	115	70	70	90	75	131	91	140	88	95	102	103	107	94	90	
APU Fuel	332	310	310	350	500	310	360	380	340	537	451	486	440	462	463	475	501	462	450	
OWG	27180	22860	21440	24350	48500	22860	23000	27100	24200	43000	32540	34850	31000	33940	34200	35820	38650	33900	32217	
Fuel	3308	2800	2620	4006	13780	2860	2765	5346	3110	5980	4683	6670	4334	5009	5033	5500	4393	5010	3918/2345	
Oxidizer	15175	12770	12106	17520	43250	13300	12660	16886	15582	—	—	—	—	—	—	—	—	—	—	
TOGW	45663	38430	36166	45876	105530	39020	38425	49332	42912	48785	37223	41520	35334	38949	39233	41320	43543	38910	38480	
Characteristics																				
Sp (ft ²)	—	426	420	570	1200	435	423	640	520	1704	1004	1099	987	1020	1021	1043	1014	1020	977	
Engine (Type)	LA-129	LA-129	LA-129	LA-129	LA-129	LA-129	LA-129	LA-129	LA-129	(2)*	(2)*	(2)*	(2)*	(2)*	(2)*	(2)*	(2)*	(2)*	(2)*	
Engine (Type)	2D RJ	2D RJ	2D RJ	2D RJ	2D RJ	2D RJ	2D RJ	2D RJ	2D RJ	1) 2D RJ	1) 2D RJ	1) 2D RJ	1) 2D RJ	1) 2D RJ	1) 2D RJ	1) 2D RJ	1) 2D RJ	1) 2D RJ	1) 2D RJ	
Fuel	LM ₂	LM ₂	LM ₂	LM ₂	LM ₂	LM ₂	LM ₂	LM ₂	LM ₂	LM ₂	LM ₂	LM ₂	LM ₂	LM ₂	LM ₂	LM ₂	LM ₂	LM ₂	LM ₂	
Oxidizer	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	
Tradeoff Study	Base Rev. M=6.0 (Hot Inlet)	Base M=6.0 (Hot Inlet)	M=6.0 (Regen cooled inlet)	M=7.0 (Regen cooled inlet)	M=8.0 (Regen cooled inlet)	T/W (TOGW) =1.25	T/W (TOGW) =2.0	Single Base Engines	Hardware Engines	Base Rev M=5.0	Base M=6.0	10 min Cruise	3.5g Struct. & 3.5g TPS	5.0g Struct. & 5.0g TPS	7.33g Struct. & 3.5g TPS	7.33g Struct. & 5.0g TPS	Hot Struc. (No Shing.) 10 sec cruise time	Non-Integral Tank	JP Used for Boost	

* F100-GE100 Turbojet

(U) FIGURE 4-153 (Continued)
 GROUP WEIGHTS COMPARISON
 M = 6.0 Vehicles
 (Presented in International Standard Units - kg)

B207 - MANNED, AIR LAUNCHED, ALL BODY, ROCKET/RAMJET										B212 - MANNED, P70, WING BODY, TURBOJET/RAMJET										
TRADEOFF STUDY	A-Revised	A	A1-1	A1-2	A1-3	A2-1	A2-2	A6	B	A-Revised	A	A3-1	A5-1	A5-2	A5-3	A5-4	A7	A8	A9	
Wing/Body	3655	3063	2711	3218	7363	3066	3068	4311	3275	6242	385	4710	5471	4138	5210	5099	6014	7190	5272	4526
Rotating Tips	235	200	195	340	1161	200	201	299	245	562	385	385	320	320	462	466	429	385	385	385
Vertical Tail(s)	319	270	265	458	1515	272	273	406	329	417	562	562	562	562	562	562	562	562	562	562
Nose Landing Gear	136	120	118	141	218	121	120	153	135	257	91	102	88	94	94	94	97	103	93	93
Main Landing Gear	265	241	238	302	549	245	241	337	287	870	379	431	365	394	397	412	452	396	387	387
Control Actuation	154	143	141	159	227	144	143	177	158	443	216	233	210	221	222	227	242	219	215	215
Engine Section	—	—	—	—	—	—	—	—	—	2087	443	443	443	443	443	443	443	443	443	443
Propulsion Group	—	—	—	—	—	—	—	—	—	1681	2087	2087	2087	2087	2087	2087	2087	2087	2087	2087
Engine(s)	1470	1472	1458	1422	2744	1431	1405	1425	1520	3983	1681	1681	1681	1681	1681	1681	1681	1681	1681	1681
Engine(s)	3060	2087	1846	1964	4128	2100	2091	2087	2141	227	1603	1603	1603	1603	1603	1603	1603	1603	1603	1603
Air Induction	217	191	184	206	395	194	191	234	220	259	172	266	165	178	185	188	211	177	191	191
Fuel System	276	211	209	236	336	210	211	257	231	32	218	261	213	223	224	231	242	223	218	218
APU	32	32	32	32	36	32	32	32	32	23	32	32	32	32	32	32	32	32	32	32
Engine Controls	—	—	—	—	—	—	—	—	—	79	23	23	23	23	23	23	23	23	23	23
Lube Cooling	79	79	79	79	79	79	79	79	79	241	79	79	79	79	79	79	79	79	79	79
Instruments	139	126	125	141	216	127	126	162	143	136	200	216	196	204	206	211	221	203	200	200
Hydraulics	136	136	136	136	136	136	136	136	136	324	136	136	136	136	136	136	136	136	136	136
Electrical	324	324	324	324	324	324	324	324	324	181	324	324	324	324	324	324	324	324	324	324
Electronics	181	181	181	181	181	181	181	181	181	113	181	181	181	181	181	181	181	181	181	181
Furnishings	113	113	113	113	113	113	113	113	113	371	113	113	113	113	113	113	113	113	113	113
MCS	231	192	177	205	421	192	192	230	204	18585	278	278	264	291	292	308	313	290	275	275
Contingency	—	—	—	—	—	—	—	—	—	109	13913	14907	13223	14541	14649	15381	16644	14523	13754	13754
Weight Empty	11548	9605	8964	10251	21058	9605	9646	11478	10190	154	109	109	109	109	109	109	109	109	109	109
Crew and Equipment	109	109	109	109	109	109	109	109	109	54	454	454	454	454	454	454	454	454	454	454
Payload	454	454	454	454	454	454	454	454	454	—	38	54	36	39	45	42	49	39	39	39
Vent Fuel/Oxidizer	34	29	27	36	100	29	29	39	36	59	—	—	—	—	—	—	—	—	—	—
Trapped Fuel	—	—	—	—	—	—	—	—	—	244	41	64	40	43	46	47	49	43	39	39
Pressurant	34	32	31	36	52	32	32	41	34	19505	205	220	200	209	210	215	227	209	208	208
APU Fuel	150	140	140	159	227	140	140	172	154	2712	14760	15808	14062	15395	15513	16248	17532	15377	14614	14614
OWE	12329	10369	9725	11045	22000	10369	10433	12293	10977	2124	2124	3025	1966	2272	2283	2495	2219	2273	1778/	1065
Fuel	1501	1270	1188	1817	6250	1297	1254	2425	1420	22217	16184	18833	16028	17667	17796	18743	19751	17650	17455	17455
Oxidizer	6883	5793	5492	7947	19618	6033	5743	7659	7068	93	—	—	—	—	—	—	—	—	—	—
TOW	20713	17432	16405	20809	47868	17699	17430	22377	19465	(2)*	93	102	92	95	95	97	92	95	91	91
Characteristics	—	—	—	—	—	—	—	—	—	21 RJ	(2)*	(2)*	(2)*	(2)*	(2)*	(2)*	(2)*	(2)*	(2)*	(2)*
Sp (M)	40	40	39	47	111	40	39	59	48	LE2	(1)20 RJ	(1)20 RJ	(1)20 RJ	(1)20 RJ	(1)20 RJ	(1)20 RJ	(1)20 RJ	(1)20 RJ	(1)20 RJ	(1)20 RJ
Engine (Type)	12-129	12-129	12-129	12-129	12-129	12-129	12-129	12-129	12-129	—	—	—	—	—	—	—	—	—	—	—
Engine (Type)	20 RJ	20 RJ	20 RJ	20 RJ	20 RJ	20 RJ	20 RJ	20 RJ	20 RJ	—	—	—	—	—	—	—	—	—	—	—
Fuel	120	120	120	120	120	120	120	120	120	—	—	—	—	—	—	—	—	—	—	—
Oxidizer	120	120	120	120	120	120	120	120	120	—	—	—	—	—	—	—	—	—	—	—
Tradeoff Study	Base Rev	Base	M=6.0	M=7.0	M=8.0	T/W	T/W	Single Hardware	Base Engines	Base Rev	Base	10 Min	3.5 g	3.5g	5.0g	7.33g	Not	Non-	JP Used	
	M=5.0	M=6.0	(Regen)	(Regen)	(Regen)	(TOGW)	(TOGW)	Base	Engines	M=6.0	Base	Cruise	Struct	Struct	Struct	Struct	Struct	Integral for	JP Used	
	(No: Inlet)	(No: Inlet)	Cooled	Cooled	Cooled	al.25	al.25	Opera-	tion				3.5g	3.5g	5.0g	7.33g	(No Shine- Tank Boost	Time)		
			Inlet	Inlet	Inlet								TPS	TPS	TPS	TPS				

*110-2E 1.0 Turbojet

(U) FIGURE 4-154 (Continued)
GROUP WEIGHTS COMPARISON
M = 12.0 Air Launched Vehicles
(Presented in International Standard Units - kg)

CONFIGURATION	B213 - MARKED - ALL ROCKET - ARE LAUNCHED - ALL BODY																		
TRADEOFF STUDY	A	A1-1	A1-2	A2-1	A2-2	A3-1	A3-2	A4-1	A4-2	A4-3	A4-4	A5-1	A5-2	A5-3	A5-4	A-6	A7-1	A7-2	A9
Wing/Body	5773	4034	4985	5821	5813	7405	10371	5697	5856	5760	5749	5573	6131	5812	6308	10336	4117	5563	6040
Rotating Tips	502	271	383	522	526	694	1025	512	532	521	520	567	649	644	776	1288	472	560	544
Vertical Tail(s)	701	558	211	701	708	930	1370	687	715	701	699	689	717	712	717	1796	635	680	731
Main Landing Gear	164	88	110	154	160	181	118	161	166	164	162	148	152	149	166	358	158	166	166
Main Landing Gear	304	156	211	290	299	361	460	296	312	304	301	286	297	288	313	721	280	306	311
Control Actuation	181	116	160	183	175	222	279	181	185	183	182	190	193	191	197	299	171	191	187
Engine Section	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-
Propulsion Group	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-
Engine(s)	708	472	567	662	898	962	1397	699	717	706	708	703	753	735	789	1656	662	708	753
Engine(s)	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-
Air Induction	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-
Fuel System	415	213	263	420	420	522	799	405	426	415	414	422	440	440	440	1180	372	404	422
APU	299	129	286	299	290	356	418	299	299	299	299	313	329	318	327	463	282	308	302
Engine Controls	27	18	23	27	27	36	41	27	27	27	27	23	23	23	23	36	27	27	27
Lube Cooling	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-
Instruments	79	79	79	79	79	79	79	79	79	79	79	79	79	79	79	79	79	79	79
Hydraulics	173	111	149	173	164	209	263	171	174	173	172	175	183	177	177	284	160	171	171
Electrical	136	136	136	136	136	136	136	136	136	136	136	136	136	136	136	136	136	136	136
Electronics	324	324	324	324	324	324	324	324	324	324	324	324	324	324	324	324	324	324	324
Furnishings	181	181	181	181	181	181	181	181	181	181	181	181	181	181	181	181	181	181	181
BOS	113	113	113	113	113	113	113	113	113	113	113	113	113	113	113	113	113	113	113
Contingency	206	143	167	204	210	260	357	203	209	205	205	200	218	211	226	395	168	203	214
Weight Empty	10308	7148	8348	10289	10523	12971	17811	10171	10451	10291	10271	10022	10918	10533	11301	19713	8367	10120	10707
Crew and Equipment	109	109	109	109	109	109	109	109	109	109	109	109	109	109	109	109	109	109	109
Payload	454	454	454	454	454	454	454	454	454	454	454	454	454	454	454	454	454	454	454
Vent Fuel/Oxidizer	109	45	68	109	114	159	260	101	111	103	102	102	109	181	113	399	91	104	113
Water	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-
Pressurant	41	41	36	41	42	68	102	41	41	41	40	41	43	41	43	64	39	43	43
APU Fuel	138	118	134	138	134	165	105	138	138	137	137	149	161	154	164	213	131	143	144
OWB	11159	7915	9149	11140	11376	13926	18960	10832	11485	11136	11113	10877	11794	11572	12184	20952	9485	10973	11567
Fuel	3443	1497	2263	3547	3502	5398	9299	3383	3552	3438	3448	3411	3674	3561	3788	11385	3003	3475	3583
Oxidizer	20756	8941	13531	21220	20993	32291	53761	20140	21369	20702	20634	20453	21977	21455	22702	68417	17968	20856	21551
TOWW	35358	18353	24943	35907	35871	51615	82020	34315	36406	35276	35195	34741	37445	36488	38674	100754	30486	35304	36701
Characteristics	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-
Sp (M)	86	51	69	87	87	115	169	85	88	86	86	86	89	88	91	196	78	91	90
Engine (Type)	11LR-129	11LR-129	11LR-129	11LR-129	11LR-129	11LR-129	11LR-129	11LR-129	11LR-129	11LR-129	11LR-129	11LR-129	11LR-129	11LR-129	11LR-129	11LR-129	11LR-129	11LR-129	11LR-129
Engine (Type)	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-
Fuel	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2
Oxidizer	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX
Tradeoff Study	Base M=12.0	M=10.0	M=10.0	T/W (TOGW) =1.25	T/W (TOGW) =2.0	10 Min Cruise	15 Min Cruise	600 lb. Payload	400 lb. Payload	25 lb./ ft ³ Pay- load	80 lb./ ft ³ Pay- load	3.5g TPS	5.0g TPS	7.33g TPS	7.33g TPS	Single Base Opera- tion	Active Cooled Insul. System	Passive Cooled LOW Insul. Density Fuel System	Normal Paint Fuel System

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(U) FIGURE 4-155
GROUP WEIGHTS COMPARISON - lb
M = 12.0 Air Launched Vehicles

CONFIGURATION		B233 - MANNED, AIR BREATHING, AIR LAUNCHED, ALL BODY, ALL ROCKET							B234 - UNMANNED, AIR LAUNCHED, ALL BODY, ALL ROCKET		B235 - MANNED, AIR LAUNCHED, ALL BODY, ROCKET/1 ST	
TRADEOFF STUDY		All-1	All-2	B	C-1	C-2	D	E	A	B	A	B
Wing/Body		14667	17339	15164	14976	15477	15132	14689	11783	14160	11908	14289
Rotating Tips		1450	1730	1510	1450	1480	1370	1040	1080	1450	1150	1430
Vertical Tail(s)		1940	2330	2010	243	1995	1730	1460	1470	1960	1530	1930
Nose Landing Gear		412	415	380	385	390	405	308	390	405	300	342
Main Landing Gear		860	860	740	750	780	768	510	565	680	590	705
Control Actuation		455	455	460	465	478	455	415	390	495	390	450
Engine Section												
Propulsion Group												
Engine(s)		2005	2420	1605	1926	2247	1780	1740	1540	1605	1560	1926
Engine(s)											3070	3850
Air Induction												
Fuel System		1430	1580	1120	1110	1130	1030	915	910	1150	895	1100
APU		745	745	760	760	780	740	655	745	860	670	940
Engine Controls		60	60	80	80	80	60	50	55	70	90	90
Lube Cooling												
Instruments		175	175	175	175	175	175	175			175	175
Hydraulics		432	485	433	438	450	430	395	370	458	360	410
Electrical		300	300	300	300	300	300	300	450	450	300	300
Electronics		715	715	715	715	715	715	715	1080	1080	715	715
Furnishings		400	400	400	400	400	400	400			400	400
ECS		250	250	250	250	250	250	250	300	300	250	250
Contingency		525	616	505	532	544	525	488	428	517	427	598
Weight Empty		26180	30875	26605	26655	27675	26265	24445	21410	25885	24870	29900
Crew and Equipment		240	240	240	240	240	240	240			240	240
Payload		1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000
Vent Fuel/Oxidizer		130	435	305	295	310	270	250	240	300	215	280
Trapped Fuel												
Pressurant		100	110	140	140	145	105	120	90	105	100	125
APU Fuel		340	370	350	350	300	340	325	450	520	305	325
OVE		28790	33030	27000	28680	29730	28220	26380	23190	27810	26730	31870
Fuel		1115	14570	12250	11800	12200	9000	8320			8500	12829
Oxidizer		66770	87430	61000	59100	60900	54000	50030	44050	53650	42780	55750
TOW		106490	135030	101910	99580	102830	91220	84740	74590	99310	78010	100410
Characteristics												
Sp (ft)		1164			1172	1200	1110	1005	881	1182	947	1187
Engine (Type)		1) LR-129	1) LR-129	1) LR-129	1) LR-129	1) LR-129	1) LR-129	1) LR-129	1) LR-129	1) LR-129	1) LR-129	1) LR-129
Engine Type		LR	LR	LR	LR	LR	LR	LR	LR	LR	LR	LR
Fuel		LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX
Oxidizer		LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX
Base (Un-)		500 lb	1000 lb	Hardware	T/W	T/W	A ₁ = 75°	A ₂ = 77°	Base (Un-)	Hardware	Base	Hardware
Landing		Drop	Drop	Drop	(T.G.W)	(T.G.W)	at 15°		Drop	Drop	Drop	Drop
Drop					(T.G.W)							

(U) FIGURE 4-155 (Continued)
GROUP WEIGHTS COMPARISON
M = 12.0 Air Launched Vehicles
(Presented in International Standard Units - kg)

CONFIGURATION	B233 - MANNEQ, AIR BREATHER, AIR LAUNCHED, ALL BODY, ALL ROCKET							B284 - UNMANNED, AIR LAUNCHED, ALL BODY, ALL ROCKET		B232 - MANNEQ, AIR LAUNCHED, ALL BODY, ALL ROCKET					
TRADEOFF STUDY	All-1	All-2	R	C-1	C-2	D	K	A	R	A	R				
Wing/Body	6651	7865	6879	6793	7018	6864	6663	5345	6559	5429	6480				
Relating Tips	658	785	685	658	672	622	672	490	658	522	648				
Vertical Tail(s)	880	1057	912	882	905	785	645	667	880	604	876				
Nose Landing Gear	187	188	173	175	177	188	140	111	152	136	155				
Main Landing Gear	367	390	337	340	354	349	232	256	308	268	320				
Control Actuators	210	207	209	211	217	206	182	177	222	177	204				
Engine Section	-	-	-	-	-	-	-	-	-	-	-				
Propulsion Group	-	-	-	-	-	-	-	-	-	-	-				
Engine(s)	910	1038	728	874	1019	808	789	698	728	708	874				
Engine(s)	-	-	-	-	-	-	-	-	-	1393	1747				
Air Induction	-	-	-	-	-	-	-	-	-	-	-				
Fuel System	649	717	507	504	513	467	415	413	521	406	499				
APU	338	338	345	345	336	297		338	390	304	426				
Engine Controls	27	27	36	36	36	27	23	25	32	41	41				
Lube Cooling	-	-	-	-	-	-	-	-	-	-	-				
Instruments	79	79	79	79	79	79	79	-	-	79	79				
Hydraulics	127	220	196	199	204	195	179	168	207	163	186				
Electrical	136	136	136	136	136	136	136	204	204	136	136				
Electronics	324	324	324	324	324	324	324	490	490	324	324				
Furnishings	181	181	181	181	181	181	181	-	-	181	181				
ECG	113	113	113	113	113	113	113	136	136	113	113				
Contingency	238	280	237	241	251	238	221	194	235	207	271				
Weight Empty	12147	14005	12077	12091	12553	11914	11088	9712	11741	11281	13563				
Crew and Equipment	109	109	109	109	109	109	109	-	-	109	109				
Payload	454	454	454	454	454	454	454	454	454	454	454				
Vent Fuel/Oxidizer	150	197	138	133	140	122	113	109	236	98	127				
Trapped Fuel	-	-	-	-	-	-	-	-	-	-	-				
Pressurant	45	50	63	63	66	48	54	40	48	45	57				
APU Fuel	154	168	159	159	164	154	148	204	136	138	146				
OVE	13059	14981	13000	13009	13487	12991	11906	10519	12615	12125	14456				
Fuel	5048	6609	5556	5352	5534	4082	3774	3334	5375	3855	5820				
Oxidizer	30288	39658	27670	26808	27624	24494	22694	19981	27057	19405	25288				
NOGW	48395	61250	46226	45169	46644	41177	38434	33834	45047	35385	45564				
Characteristics															
Sp (M)	108	130	112	109	111	103	93	82	110	88	110				
Engine (Type)	111B-129	111B-129	111B-10	111B-10	111B-10	111B-129	111B-129	111B-129	111B-10	111B-129	111B-10				
Engine (Type)	111B-129	111B-129	111B-10	111B-10	111B-10	111B-129	111B-129	111B-129	111B-10	111B-129	111B-10				
Fuel	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX				
Oxidizer	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX				
Tradeoff Study	2268 Kg Landing Prop.	4536 Kg Landing Prop.	Hardware Engines (TOGW) 7/4 * 1.117	T/W (TOGW) 1.375	T/W (TOGW) 1.55	Δ111B-129	Δ111B-129	Base (Unmanned)	Hardware Engines (Unmanned)	Base	Hardware Engines				

(U) FIGURE 4-156
GROUP WEIGHTS COMPARISON - lb
M = 12.0 HTO Vehicles

CONFIGURATION	B257 - MANNED, HTO, BLUNDED WING BODY, TURBOJET/CSJ												B260 - MANNED, HTO, ALL BODY, ALL ROCKET					
TRADEOFF STUDY	F	F1-1	F2-1	F3-1	F3-2	F5-1	F5-2	F5-3	F5-4	F9-1	F9-2	F9-3	A	A1-1	A1-2	A6	A/VTO	B
Wing/Body	34000	23579	34404	40806	37404	30236	37085	35216	39038	28232	35931	29208	19926	14986	12713	33822	19926	22082
Rotating Tips	3650	2272	3540	4550	4000	3250	3910	4300	4800	2960	3700	2940	1580	972	802	3445	1580	2045
Vertical Tail(s)	4630	2885	4470	5650	5050	4130	5120	5420	6000	3440	4700	3700	2580	1595	1310	5695	2580	3310
Rose Landing Gear	370	254	430	560	525	340	505	385	515	480	475	320	460	375	332	1035	460	489
Main Landing Gear	1850	1274	1970	2400	2310	1670	2170	1930	2240	1670	2070	1510	2760	1994	1780	5435	2760	2860
Control Actuation	1080	854	1135	1200	1120	1030	1095	1130	1370	870	1090	965	620	482	472	958	620	632
Engine Section																		
Propulsion Group																		
Engine(s)	13625	10900	16350	16350	16350	13625	16350	16350	16350	10900	16350	13625	2580	1850	1485	3800	2580	4050
Engine(s)	4420	2755	4350	5550	5030	3920	4990	4440	4850	3080	4650	3540	-	-	-	-	-	-
Air Induction	9400	7500	11300	11300	11300	9400	12000	11500	11300	7500	11300	9400	-	-	-	-	-	-
Fuel System	1060	618	965	1220	1110	920	1030	1020	1070	1400	1010	865	1280	862	658	2665	1280	1690
APU	810	640	790	915	855	760	830	805	840	650	820	950	740	611	564	1142	740	788
Engine Controls	150	150	180	180	180	150	180	180	180	120	180	150	55	50	50	110	55	70
Lube Cooling	200	200	240	240	240	200	240	240	240	160	240	200	-	-	-	-	-	-
Instruments	175	175	175	175	175	175	175	175	175	175	175	175	175	175	175	175	175	175
Hydraulics	1170	925	1280	1310	1230	1100	1190	1220	1480	940	1180	1030	565	490	428	1032	565	694
Electrical	300	300	300	300	300	300	300	300	300	300	300	300	300	300	300	300	300	300
Electronics	715	715	715	715	715	715	715	715	715	715	715	715	715	715	715	715	715	715
Furnishings	400	400	400	400	400	400	400	400	400	400	400	400	400	400	400	400	400	400
ECS	450	450	450	450	450	450	450	450	450	450	450	450	250	250	250	600	250	250
Contingency	1600	1160	1702	1924	1806	1484	1810	1760	1882	1319	1749	1437	714	533	456	1087	714	848
Weight Empty	80055	58004	85106	96195	90550	74255	90545	87936	94195	65761	87485	71880	35700	26640	22898	62375	35700	42398
Crew & Equipment	240	240	240	240	240	240	240	240	240	240	240	240	240	240	240	240	240	240
Payload	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000	1000
Vent Fuel/Oxidizer	700	344	645	900	765	590	690	674	730	750	660	540	460	283	190	1280	460	610
Trapped Fuel	-	-	-	-	-	-	-	-	-	179	-	-	-	-	-	-	-	-
Pressurant	235	120	244	260	255	225	245	230	250	175	240	210	170	147	71	183	170	184
APU Fuel	370	292	365	405	390	350	380	390	385	295	375	330	330	290	251	472	330	368
OWE	82600	60000	87600	99000	93200	76660	93100	90470	96800	68400	90000	74200	37900	28600	24650	65550	37900	44800
Fuel	35230	17200	32247	45202	38398	29460	34950	33700	36775	19900/17010	33415	26977	15620	9450	6325	42700	15620	22200
Oxidizer	-	-	-	-	-	-	-	-	-	-	-	-	93730	56650	37925	256300	93730	122050
TOGW	117830	77200	119847	144202	131598	106120	128050	124170	133575	105310	123415	101177	147250	94700	68900	364550	147250	189050
Characteristics																		
Sp(ft ²)	2860	1780	2760	3440	3100	2530	2900	2840	3000	1980	2920	2280	1405	1025	810	2785	1405	1705
Engine(Type) F-100-GE-100	(5)	(4)	(6)	(6)	(6)	(5)	(6)	(6)	(6)	(4)	(6)	(5)	1)LR-129(1)	1)LR-129(1)	1)LR-129(1)	1)LR-129(1)	1)LR-129(1)	1)LR-129(1)
Engine(Type)	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	1)LR-129(1)	1)LR-129(1)	1)LR-129(1)	1)LR-129(1)	1)LR-129(1)	1)LR-129(1)
Fuel	LH ₂	LH ₂	LH ₂	LH ₂	LH ₂	LH ₂	LH ₂	LH ₂	LH ₂	LH ₂ /JP	LH ₂	LH ₂	LH ₂	LH ₂	LH ₂	LH ₂	LH ₂	LH ₂
Oxidizer	-	-	-	-	-	-	-	-	-	-	-	-	LOX	LOX	LOX	LOX	LOX	LOX
Tradeoff Study	Base M=12.0	M=10.0	T/W (TOGW) Cruise =1.17	10 min Cruise	7.5 min Cruise	3.5g Struct & 3.5g TPS	5.0g Struct & 5.0g TPS	7.3g Struct & 3.5g TPS	7.3g Struct For & 5.0g in Boost	Normal Boiling Fuel	Slush LH ₂		Base M=12.0	M=10.0	M=8.0	Single Base Operation(2.8 min cruise time)	Vertical Hardware Takeoff Engines	

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(U) FIGURE 4-156 (Continued)
GROUP WEIGHTS COMPARISON
M = 12.0 HTO Vehicles
(Presented in International Standard Units- kg)

B257 - MANNED, HTO, BLENDED WING BODY, TURBOJET/CM													B260 - MANNED, HTO, ALL BODY, ALL ROCKET					
CONFIGURATION	F	F1-1	F2-1	F3-1	F3-2	F5-1	F5-2	F5-3	F5-4	F9-1	F9-2	F9-3	A	A1-1	A1-2	A6	A/VTO	B
TRADEOFF STUDY																		
Wing/Body	15422	10696	15606	18510	16967	13715	16822	15974	17708	12806	16398	13249	9039	6798	5767	13341	9039	10470
Rotating Tip	1656	1031	1606	2064	1814	1474	1774	1951	2177	1343	1678	1334	717	441	364	1561	717	928
Vertical Tail(s)	2100	1309	2020	2563	2291	1874	2322	2459	2722	1561	2132	1678	1170	724	598	2583	1170	1502
Nostr Landing Gear	168	115	195	254	238	154	229	175	234	218	216	149	209	170	151	470	209	222
Main Landing Gear	839	577	894	1089	1048	758	984	875	1016	758	939	685	1252	905	807	2465	1252	1297
Control Actuation	490	387	515	544	508	467	497	513	622	495	495	438	281	219	214	435	281	287
Engine Section																		
Propulsion Group													1170	839	674	1724	1170	1837
Engine(s)	6180	4944	7416	7416	7416	6180	7416	7416	7416	4944	7416	6180						
Engine(s)	2005	1250	1973	2518	2282	1778	2264	2014	2200	1397	2109	1606						
Air Induction	4264	3402	5126	5126	5126	4264	5443	5217	5126	3402	5126	4264	561	391	298	1209	561	767
Fuel System	481	280	438	553	503	418	467	463	485	635	458	393	336	277	256	518	336	357
APU	368	290	358	415	388	345	377	365	381	295	372	431	25	23	23	50	25	32
Engine Controls	68	68	82	82	82	68	82	82	82	54	82	68						
Lube Cooling	91	91	109	109	109	91	109	109	109	73	109	91	79	79	79	79	79	79
Instruments	79	79	79	79	79	79	79	79	79	79	79	79	257	222	194	468	257	315
Hydraulics	531	420	562	594	558	499	540	553	671	426	535	467	136	136	136	136	136	136
Electrical	136	136	136	136	136	136	136	136	136	136	136	136	324	324	324	324	324	324
Electronics	324	324	324	324	324	324	324	324	324	324	324	324	181	181	181	181	181	181
Furnishings	181	181	181	181	181	181	181	181	181	181	181	181	113	113	113	272	113	113
ECS	204	204	204	204	204	204	204	204	204	204	204	204	324	242	207	475	324	385
Contingency	726	526	772	873	819	673	821	798	854	598	794	652						
Weight Empty	36313	26310	38604	43634	41073	33682	41071	36888	42727	29829	39683	32605	16194	12084	10386	28293	16194	19232
Crew & Equipment	109	109	109	109	109	109	109	109	109	109	109	109	109	109	109	109	109	109
Payload	454	454	454	454	454	454	454	454	454	454	454	454	454	454	454	454	454	454
Vent Fuel/Oxidizer	317	156	292	408	347	267	313	305	331	340	299	245	209	128	86	580	209	276
Trapped Fuel																		
Pressurant	107	54	111	118	116	102	111	104	113	79	109	95	77	67	32	83	77	83
APU Fuel	168	133	165	183	177	159	172	177	175	134	170	149	149	131	114	214	149	167
OWE	37468	27216	39735	44906	42276	34773	42230	41037	43909	31026	40824	33657	17192	12973	11181	29733	17192	20321
Fuel	15980	7802	14627	20504	17417	13363	15853	15286	16681	9027/7716	15157	12237	7085	4287	2869	19369	7085	10070
Oxidizer													42516	25696	17203	116258	42516	55362
TOGW	53448	35018	54362	65410	59693	48136	58083	56323	60590	47769	55981	45894	66793	42956	31253	165360	66793	85753
Characteristics																		
Sp (M2)	266	165	256	320	288	235	269	264	279	184	271	212	131	95	75	259	131	158
Engine (Type) F-100-GE-100	(5)	(4)	(6)	(6)	(6)	(5)	(6)	(6)	(6)	(4)	(6)	(5)	1)LR-129(1)	1)LR-129(1)	1)LR-129(1)	1)LR-129(1)	1)LR-129	(1)J28
Engine (Type)	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	(1)CSJ	LH2	LH2	LH2	LH2	LH2	LH2
Fuel	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LH2	LOX	LOX	LOX	LOX	LOX	LOX
Oxidizer																		
Tradeoff Study	Base M=12.0	M=10.0	T/V (TOGW) = 1.17	10 min Cruise	7.5 min Cruise	3.5g Struct & 3.5g TPS	5.0g Struct & 5.0g TPS	7.33g Struct & 3.5g TPS	7.33g Struct & 5.0g TPS	JP Used For Boost	Normal Boiling Point Fuel	Slush LH2	Base M=12.0	M=10.0	M=8.0	Single Base Operation (2.8 min cruise time)	Vertical Takeoff Engines	Hardware

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(U) During the course of the analysis, the basic Mach 6.0 vehicle was revised to reflect improved weight estimates for ramjet engines, air induction systems, and associated engine provisions. These revised baselines are noted in Figure 4-153. Since the results of the parametric variations would still result in the same trade-off parameter selection, regardless of the baseline vehicle, only the baseline vehicle weights were revised. However, in developing the weights for the near optimum vehicles shown in Section 4.6, it was necessary to derive their weights based on the revised Mach 6.0 vehicles.

(U) Figures 4-154 and 4-155 provide group weight statements for the B233 manned M 12.0 air launched rocket boost/cruise vehicle, the B284 unmanned M = 12 air launched rocket boost/cruise vehicle, and the B232 manned M 12.0 air launched rocket boost, scramjet cruise vehicle. The basic ground rules used for weight estimation of the vehicles presented on Figures 4-154 through 4-155 are as follows:

	Configuration B233	Configuration B284	Configuration B232
Cruise Speed	M = 12.0	M = 12.0	M = 12.0
Cruise Time	5 min	5 min	5 min
T/W (TOGW)	1.5	1.5	1.5
Base Operation	Dual	Dual	Dual
Boost Engine	Rubberized LR-129	Rubberized LR-129	Rubberized LR-129
Cruise Engine	Rubberized LR-129	Rubberized LR-129	SJ
Structure Load Factor	5.0 g	5.0 g	5.0 g
TPS Load Factor	3.5 g	3.5 g	3.5 g
Payload	1000 lb (454 kg)	1000 lb (454 kg)	1000 lb (454 kg)
Payload ρ (lb/ft ³)	25 (400 kg/m ²)	25 (400 kg/m ²)	25 (400 kg/m ²)
Landing Propellant	0	0	0

(U) Figure 4-156 compares the basic M 12.0 B257 fixed size GE-F100 turbojet boost and convertible scramjet boost/cruise horizontal takeoff vehicle with various tradeoff study configurations. Included on these figures are the M 12.0 horizontal takeoff rocket boost/cruise vehicles. The ground rules for the base vehicles B257 and B260 are as follows:

	Configuration B257	Configuration B260
Cruise Speed	M = 12.0	M = 12.0
Trajectory	HTO	HTO
Cruise Time	5 min	5 min
Boost Engine(s)	GE-F100 (to M = 3.0)	LR-129 Rocket
Boost Engine	CSJ (M = 3.0+)	-
Structure Load Factor	5.0 g	5.0 g
TPS Load Factor	3.5 g	3.5 g
Boost Propellant	LH ₂	LH ₂ /LOX
T/W (TOGW)	.961	1.5
Base Operation	Dual	Dual
LH ₂ Temperature	Subcooled (30°R)	Subcooled (30°R)

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(U) Because of the large number of configurations investigated during Phase II, a detailed weight analysis was not conducted for the seven selected configurations discussed in Section 5.3. However, this wealth of data did allow us to develop a reasonable weight estimate by adjusting the weight of the basic configuration to account for the design changes incorporated. As an example, the B233 "near optimum" weight derivation is presented below:

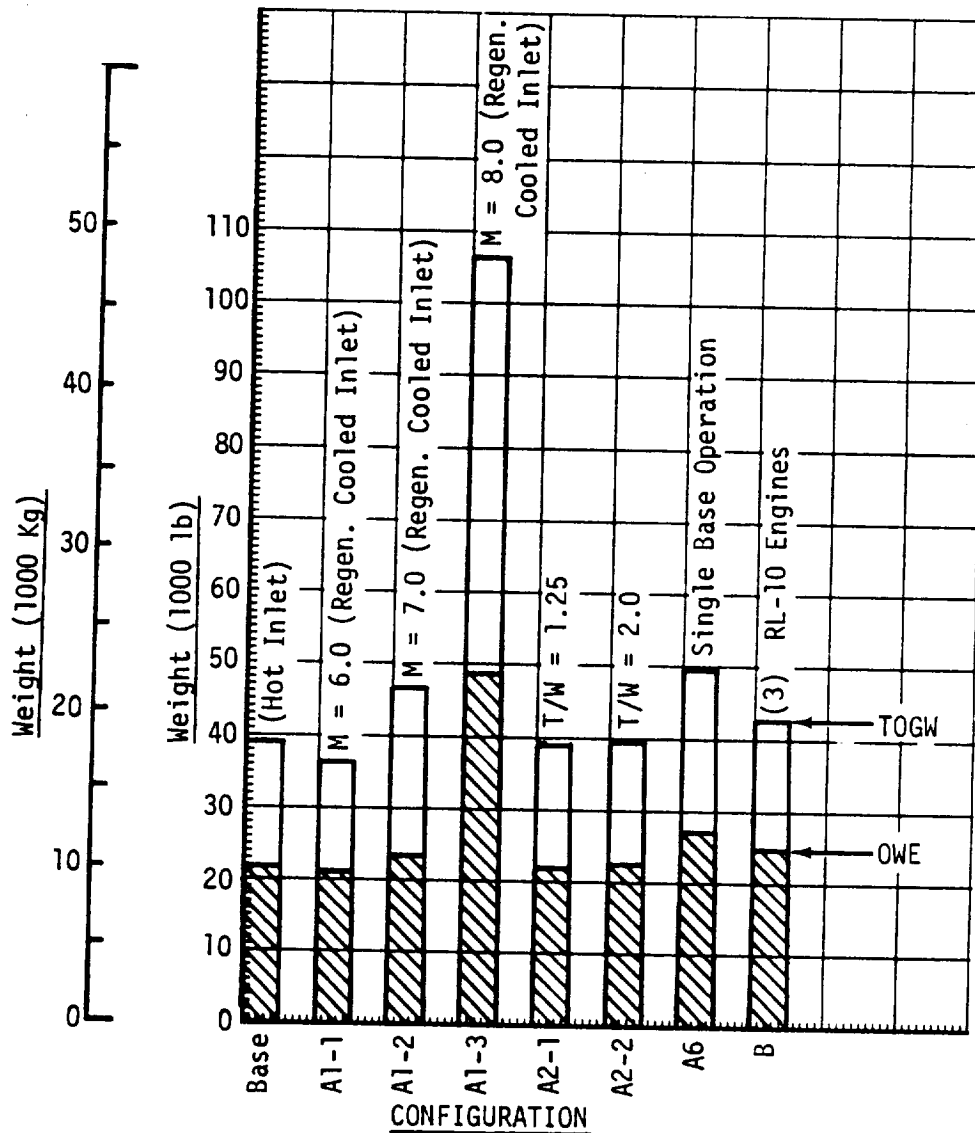
	<u>lb</u>	<u>kg</u>
B233A OWE	24,600	11,159
Δ Weight (five) RL-10 vs (one) LR-129 (Based on B233B tradeoff study)	4,060	1,841
Δ Weight of Active vs Passive Cooling System (Based on B233A7-1 tradeoff study)	-4,300	-1,950
Δ Weight of 1500 lbm vs 1000 lbm payload (Based on B233A4-2 tradeoff study)	900	408
Δ Weight-resizing	<u>400</u>	<u>181</u>
B233 "near optimum" OWE	25,660	11,639

The previous discussion illustrates that the operating weight empty was reasonably derived without extensive configuration redrawing and performance parametric rework. Once the operating weight empty was determined, the average propellant fraction determined in the B233 tradeoff studies (configuration B233A7-1 and configuration B233B) was applied and the takeoff gross weight was derived.

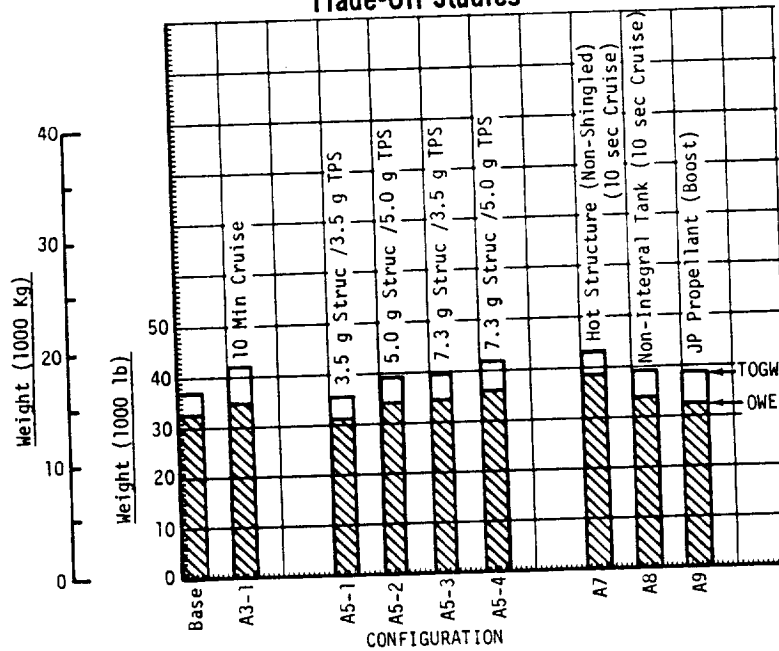
$$TOGW = \frac{25,660}{.295} = 87,000 \text{ lb (39,463 kg)}$$

(U) Figures 4-157 through 4-164 graphically illustrate the effect on weight of the tradeoff studies performed on the seven base configurations. The lower (cross-hatched) bar represents the operating weight empty. The upper bar represents the propellant weight.

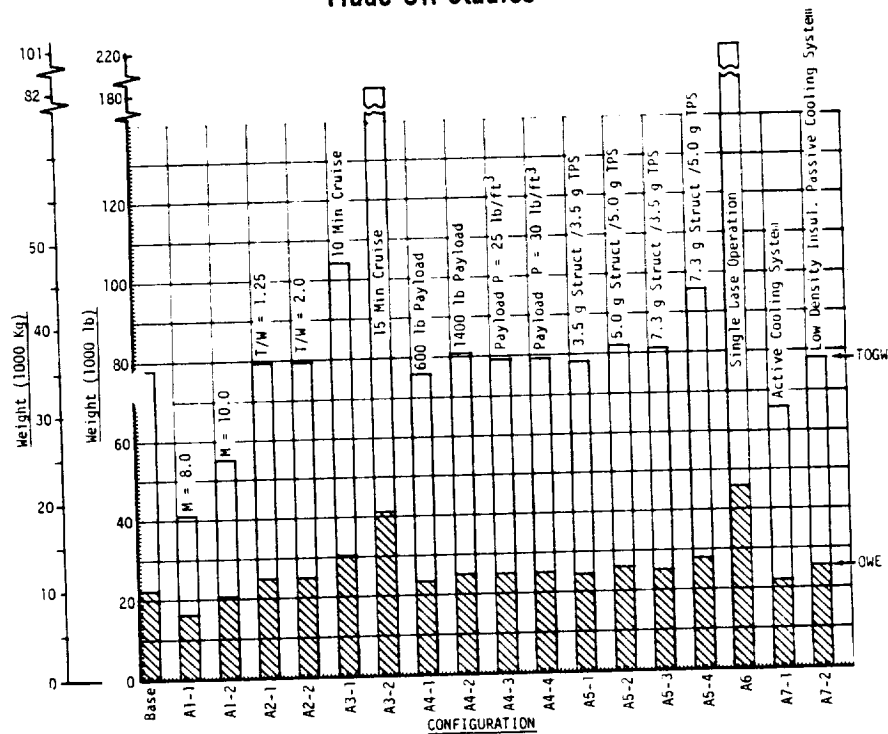
(U) FIGURE 4-157
B207 M - 6 HTO - ROCKET BOOST AND RAMJET CRUISE
Trade-Off Studies



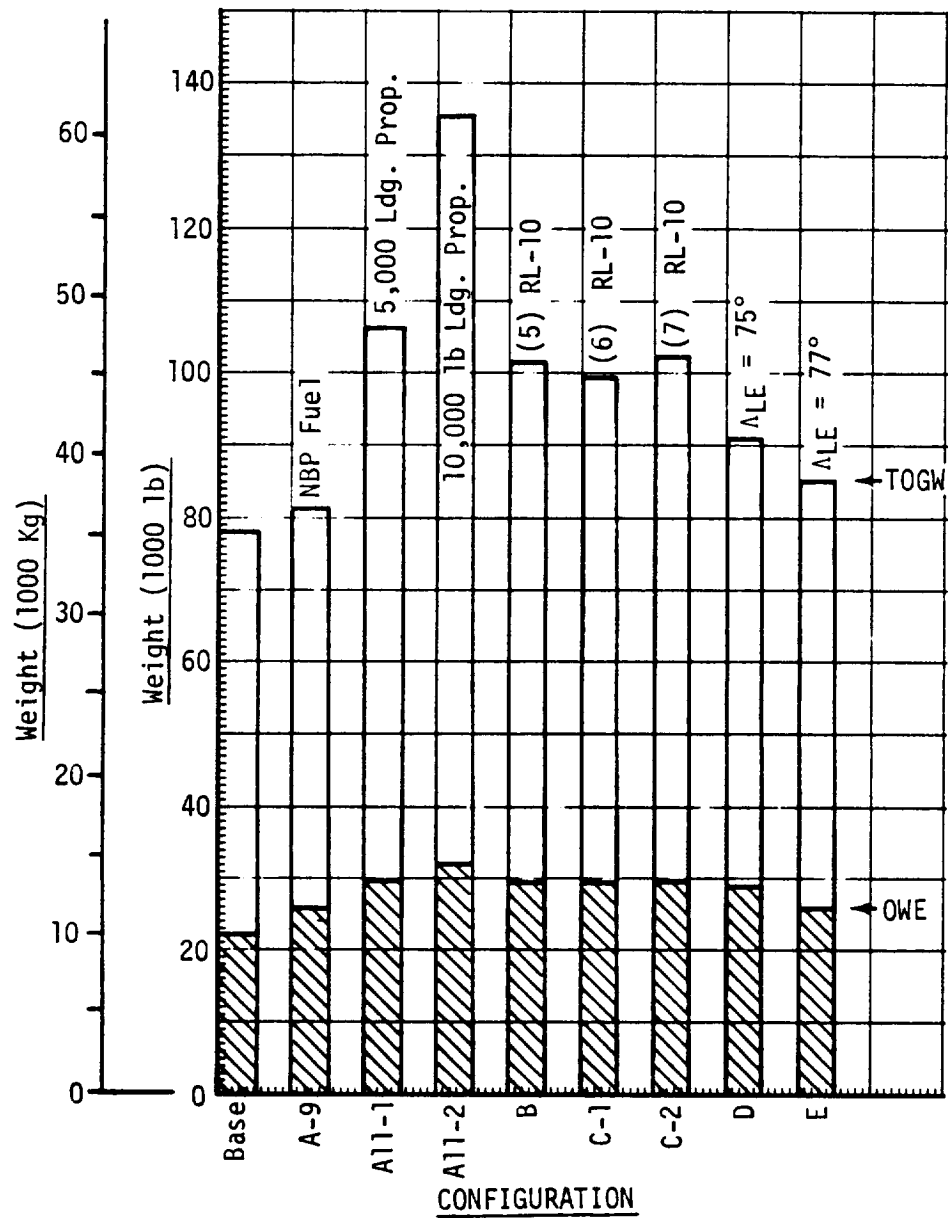
(U) FIGURE 4-158
B212 M = 6.0 HTO - (2) GE F-100 TJ BOOST AND RAMJET CRUISE
Trade-Off Studies



(U) FIGURE 4-159
B233 M = 12.0 AIR LAUNCHED
Trade-Off Studies

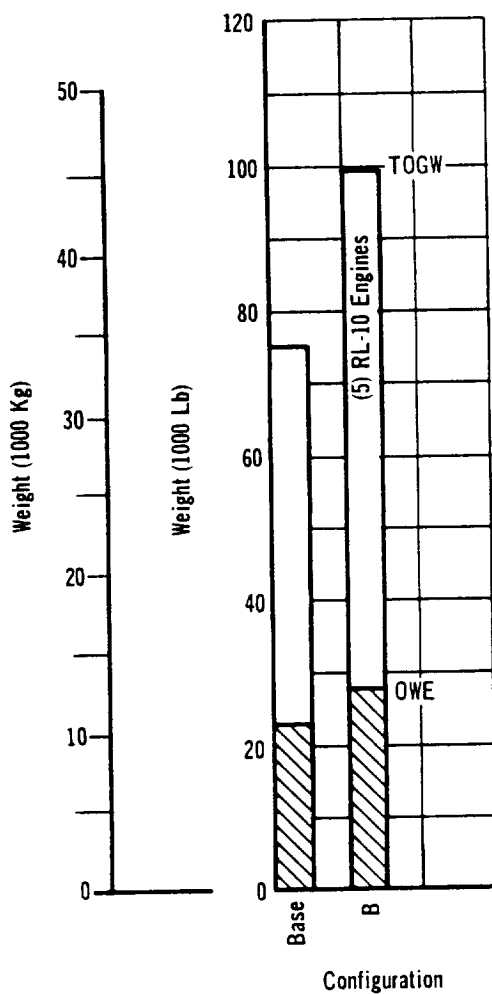


(U) FIGURE 4-160
B233 M = 12.0 AIR LAUNCHED
Trade-Off Studies

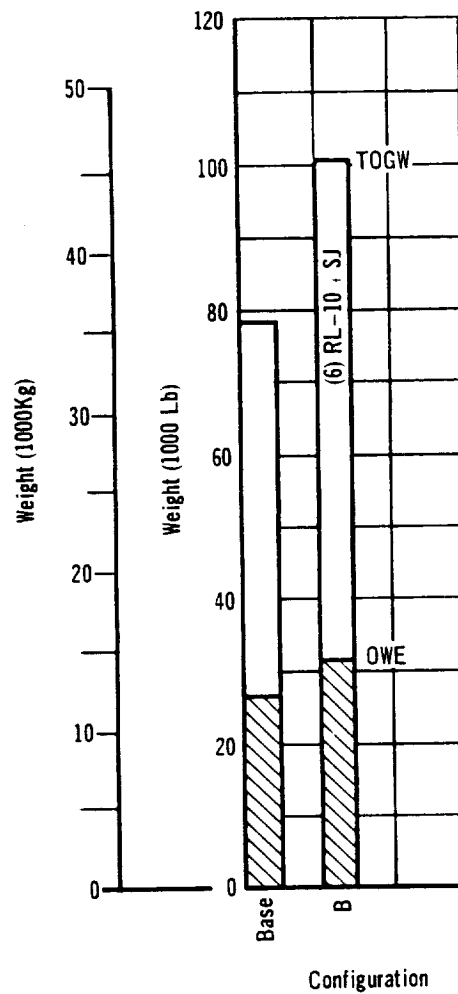


50

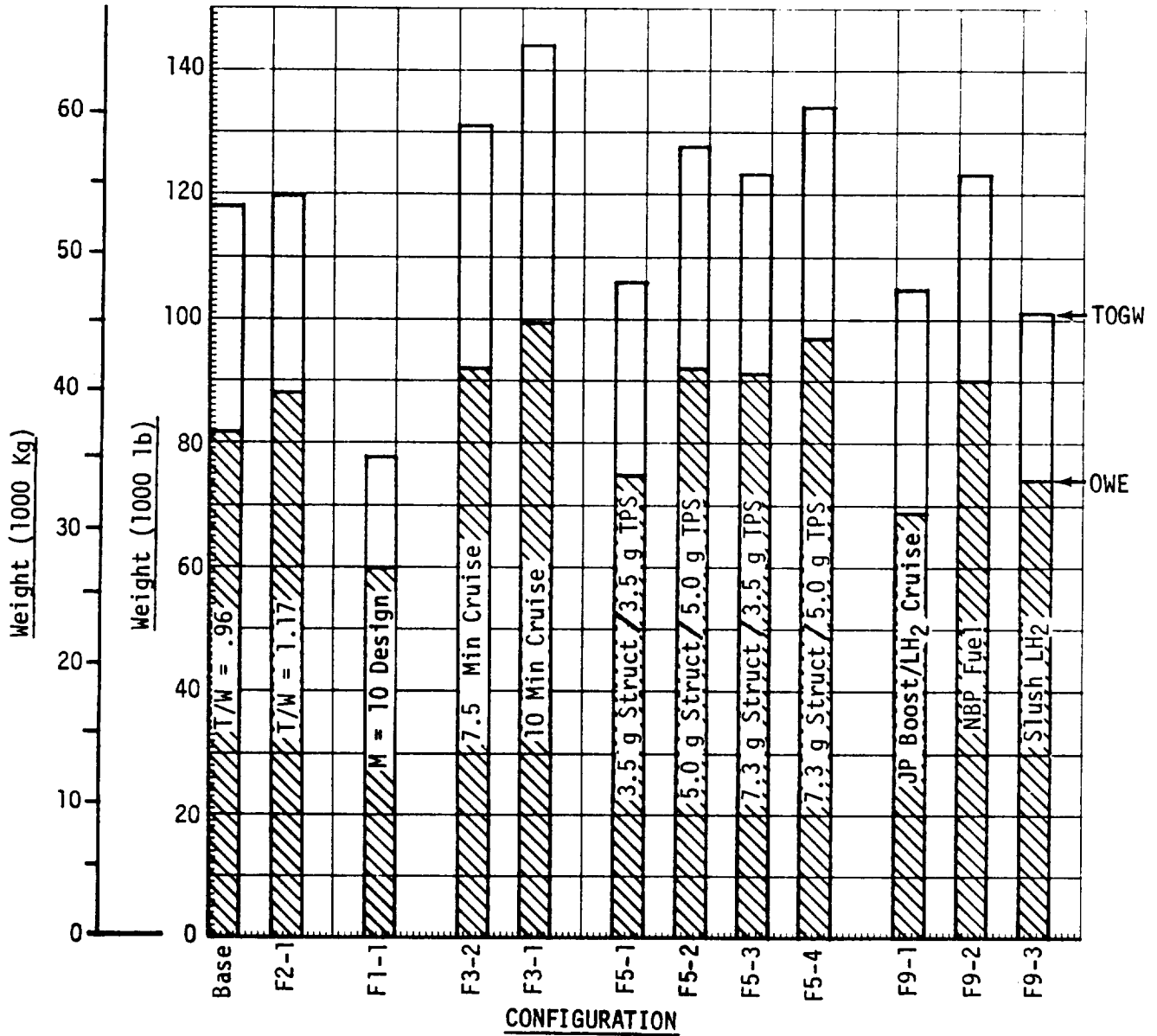
(U) FIGURE 4-161
B284 UNMANNED
M = 12.0 AIR LAUNCHED-
ROCKET BOOST/CRUISE
Trade- Off Studies



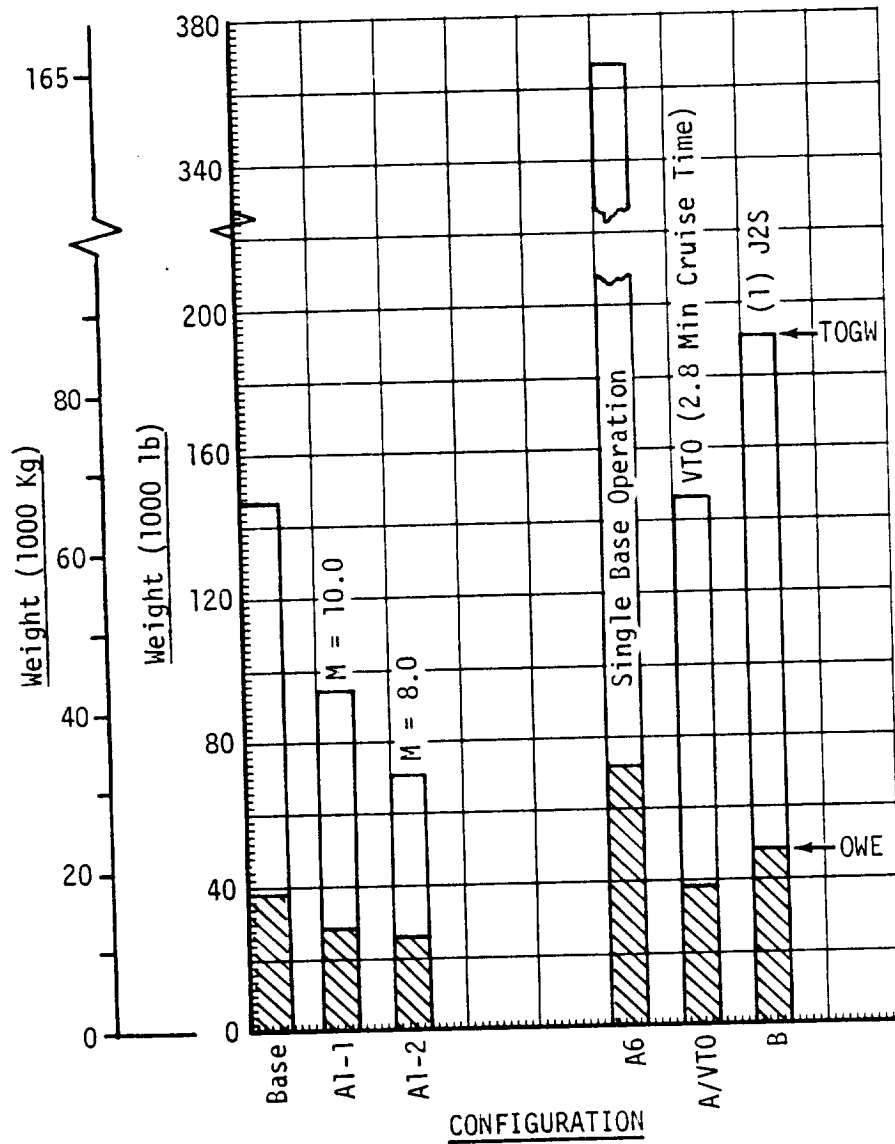
(U) FIGURE 4-162
B232 M= 12.0
AIR LAUNCHED
ROCKET BOOST AND SJ CRUISE
Trade-Off Studies



(U) FIGURE 4-163
B257 M = 12.0 HTO - GE-F100 TJ BOOST AND CSJ CRUISE
Trade-Off Studies



(U) FIGURE 4-164
B260 M = 12.0 HTO - ROCKET BOOST/ROCKET CRUISE
Trade-Off Studies



4.5.3 (U) COSTS - Figures 4-165 and 4-167 through 4-173 show the total system cost summaries in tabular and bar chart form for the Phase II baseline configurations and their parametric variations, while Figures 4-166 and 4-174 show the total system costs for the near optimum configurations. All costs reflect January 1970 economics. Total system costs were derived for seven baseline vehicles, 57 parametric variations of the baseline vehicles, and seven near optimum vehicles.

(U) Detailed total system cost summaries are shown in Figures 4-175 and 4-176 for the baseline flight research vehicles and their parametric variations and the "near-optimum" flight research vehicles.

(U) The weight summary and the airframe and miscellaneous subsystem costs for the baseline configurations and their parametric variations and also for the "near-optimum" configurations are shown in Figure 4-177. The propulsion system parameters and associated costs are shown in Figure 4-178.

(U) Flight research vehicle refurbishment and propellant costs are shown in Figures 4-179 and 4-180.

(U) FIGURE 4-165

TOTAL SYSTEM COST SUMMARY FOR BASELINE CONFIGURATIONS AND THEIR PARAMETRIC VARIATIONS
(Millions of 1970 Dollars)

Cost Categories	Configurations							
	B207, M 6.0-8.0, Air Launched, All Body Config., RKT/Ramjet							
	A	A1-1	A1-2	A1-3	A2-1	A2-2	A6-1	B
	Base Config	Cool Inlet M 6.0	Cool Inlet M 7.0	Cool Inlet M 8.0	TIW 1.25	T W = 2.00	Single Base Operation	Off-the Shelf Engine
RDT&E	332	324	349	467	323	348	359	285
Investment	121	115	134	222	121	122	131	130
Operating	164	157	178	273	164	165	184	178
Total	617	596	661	962	608	635	674	593

Cost Categories	Configurations								
	B212, M 6.0, HTO Launched, Wing Body Config., Turbojet Ramjet								
	A	A3-1	A5-1	A5-2	A5-3	A5-4	A7-1	A8-1	A9-1
	Base Config	10 Minute Test Time	3.5g Structure 3.5g Shingle	5.0g Structure 5.0g Shingle	7.3g Structure 3.5g Shingle	7.3g Structure 5.0g Shingle	Hot Structure	Nonintegral Tank	JP Fuel
RDT&E	304	310	300	308	308	313	312	307	304
Investment	149	163	148	161	162	168	165	152	149
Operating	212	232	212	230	232	239	235	217	212
Total	665	705	660	699	702	720	712	676	665

Cost Categories	Configurations									
	B232, M 12.0, Air Launched All Body Config, RKT SJ		B233, M 8.0-12.0, Air Launched, All Body Config, RKT							
	A	B	A	A1-1	A1-2	A2-1	A2-2	A3-1	A3-2	A4-1
	Base Config	Off-the Shelf Engine	Base Config	M 8.0	M 10.0	TIW 1.25	TIW 2.0	10 Minute Cruise Time	15 Minute Cruise Time	600 Pound Payload
RDT&E	555	476	243	158	196	242	266	300	387	240
Investment	139	157	124	110	115	121	127	151	194	121
Operating	184	205	178	167	171	172	179	204	247	175
Total	878	838	544	435	482	535	572	655	828	536

Cost Categories	Configurations							
	B233, M 8.0-12.0, Air Launched, All Body Config, RKT							
	A4-2	A4-3	A4-4	A5-1	A5-2	A5-3	A5-4	A6-1
	1400 Pound Payload	Payload P 25 lb ft ³ P 400 kg m ³	Payload P 30 lb ft ³ P 481 kg m ³	3.5g Structure 3.5g Shingle	5.0g Structure 5.0g Shingle	7.3g Structure 3.5g Shingle	7.3g Structure 5.0g Shingle	Single Base Operation
RDT&E	247	243	243	239	254	246	257	409
Investment	126	124	123	122	134	127	136	213
Operating	181	177	177	175	190	181	194	279
Total	554	544	543	536	578	554	587	901

(U) FIGURE 4-165
TOTAL SYSTEM COST SUMMARY FOR BASELINE CONFIGURATIONS AND THEIR PARAMETRIC VARIATIONS (CONTINUED)
 (Millions of 1970 Dollars)

Cost Categories	Configurations									
	B233, M 8.0-12.0, Air Launched, All Body Config, RKT									
	A7-1	A7-2	A9	A11-1	A11-2	B	C-1	C-2	D	E
	Active Cooling System	Passive Insulation (Minimum Weight)	NBP Fuel	Powered Landing 5000 Lb Fuel	Powered Landing 10,000 Lb Fuel	5RL-10's T W = 1.117	6RL-10's T W = 1.375	7RL-10's T W = 1.550	75° Leading Edge Sweep	77° Leading Edge Sweep
RD&E	224	247	241	277	314	163	163	167	269	259
Investment	115	128	127	143	161	151	153	159	140	126
Operating	167	183	182	193	213	294	204	209	193	180
Total	506	558	550	613	688	518	520	535	602	565

Cost Categories	Configurations								
	B257, M 10.0-12.0, HTO Launched, Blended Wing Body Config, Turbojet CSJ								
	F	F1-2	F2-1	F3-1	F3-2	F5-1	F5-2	F5-3	F5-4
	Base Config	M - 10.0	TIW 1.17	10 Minute Cruise Time	7.5 Minute Cruise Time	3.5g Structure 3.5g Shingle	5.0g Structure 5.0g Shingle	7.3g Structure 3.5g Shingle	7.3g Structure 5.0g Shingle
RD&E	602	533	601	627	614	588	613	610	622
Investment	281	219	285	325	301	261	303	296	321
Operating	383	297	386	439	408	357	411	402	436
Total	1266	1049	1272	1391	1323	1206	1327	1308	1379

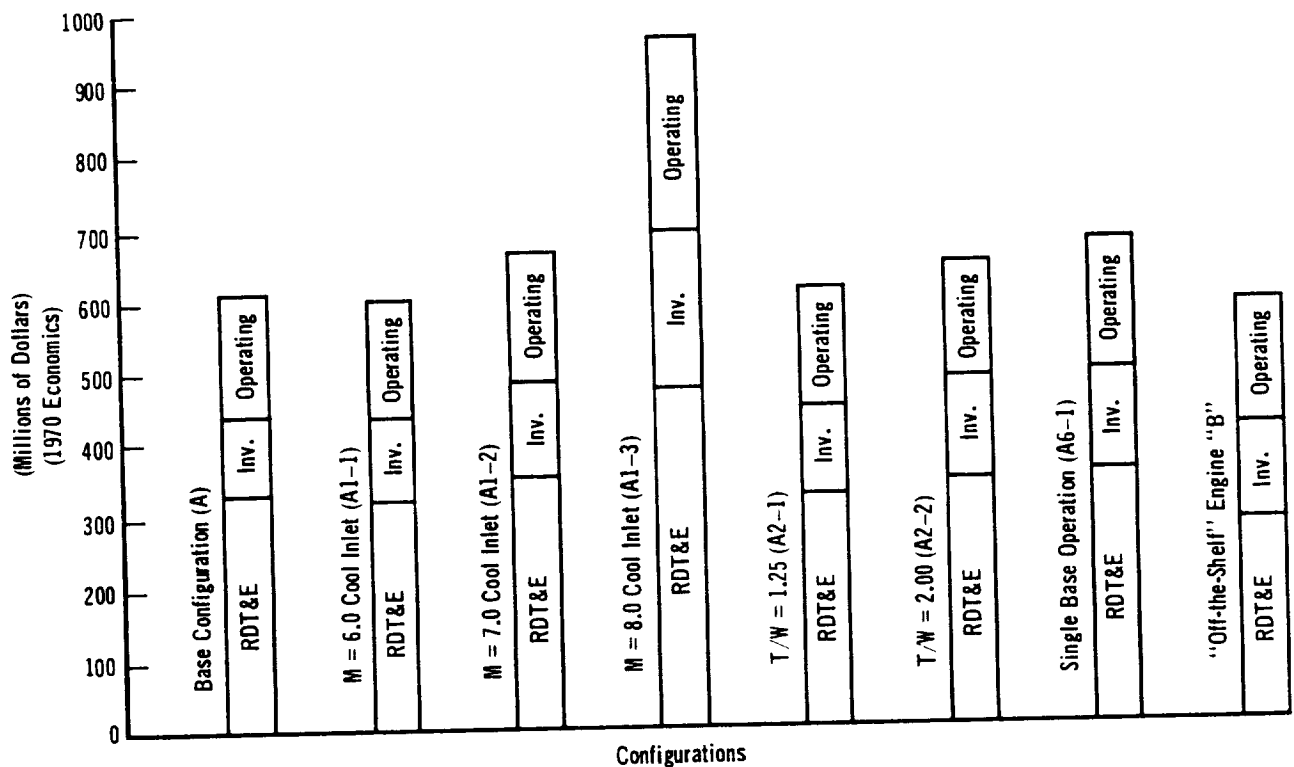
Cost Categories	Configurations									
	B257, M 10.0-12.0, HTO Launched, Blended Wing Body Config, Turbojet CSJ			B260, M 8.0-12.0, V/HTO Launched, All Body Config, Rocket					B284, M 12.0, Air Launched Unmanned, All Body Config, Rocket	
	F9-1	F9-2	F9-3	A	A1-1	A1-2	A6-1	B	A	B
	JP/LH ₂	NBP LH ₂	Slush LH ₂	Base Config	M - 10.0	M - 8.0	Single Base Operation	Off-the Shelf Engines	Base Config	Off-the Shelf Engines
RD&E	570	609	580	344	263	217	537	200	234	162
Investment	234	294	255	160	121	107	229	172	176	217
Operating	327	397	345	238	184	165	347	257	188	218
Total	1131	1300	1180	742	568	489	1113	629	598	597

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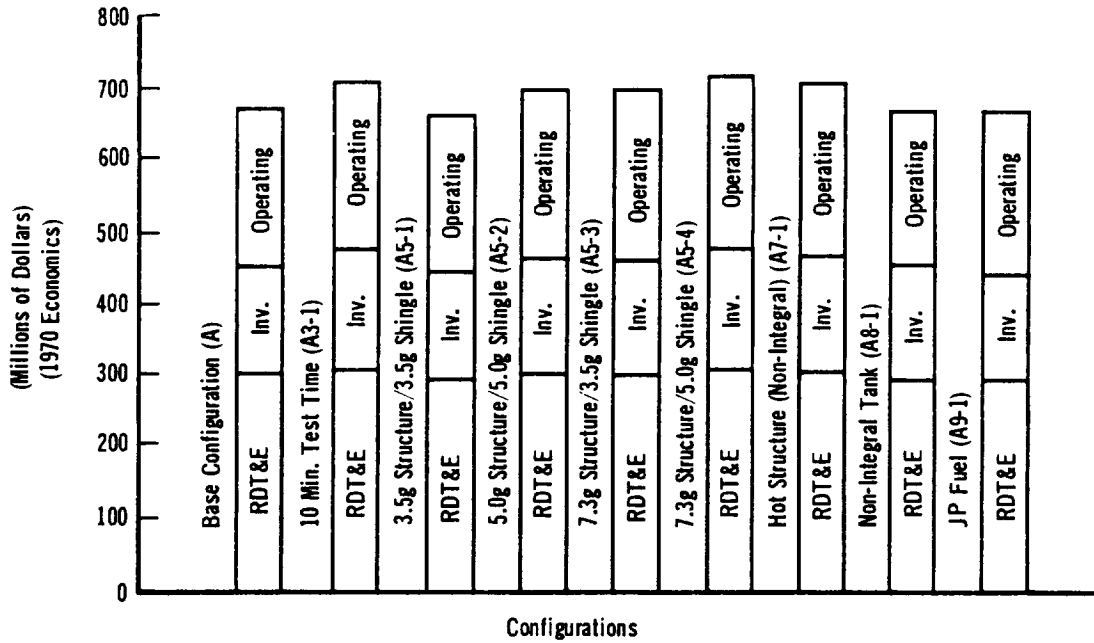
(U) FIGURE 4-166
TOTAL SYSTEM COST SUMMARY FOR THE "NEAR OPTIMUM" CONFIGURATIONS
(Millions of 1970 Dollars)

Cost Categories	"Near Optimum" Configurations						
	B207 M = 6, Air Launched, All Body Config., Rocket/Ramjet	B212 M = 6, HTO Launched, Wing Body Config., Turbojet/Ramjet	B232 M = 12, Air Launched, All Body Config., Rocket/Scramjet	B233 M = 12, Air Launched, All Body Config., All Rocket	B257 M = 12, HTO Launched, Blended Wing Body, Turbojet CSJ	B260 M = 12, HTP Launched, All Body Config., J2S Rocket	B284 M = 12, Air Launched, Unmanned, All Body Config., All Rocket
RDT & E	299	330	465	151	547	196	148
Investment	141	188	155	135	221	172	195
Operating	194	269	200	188	312	256	203
Total	634	787	820	474	1080	624	546

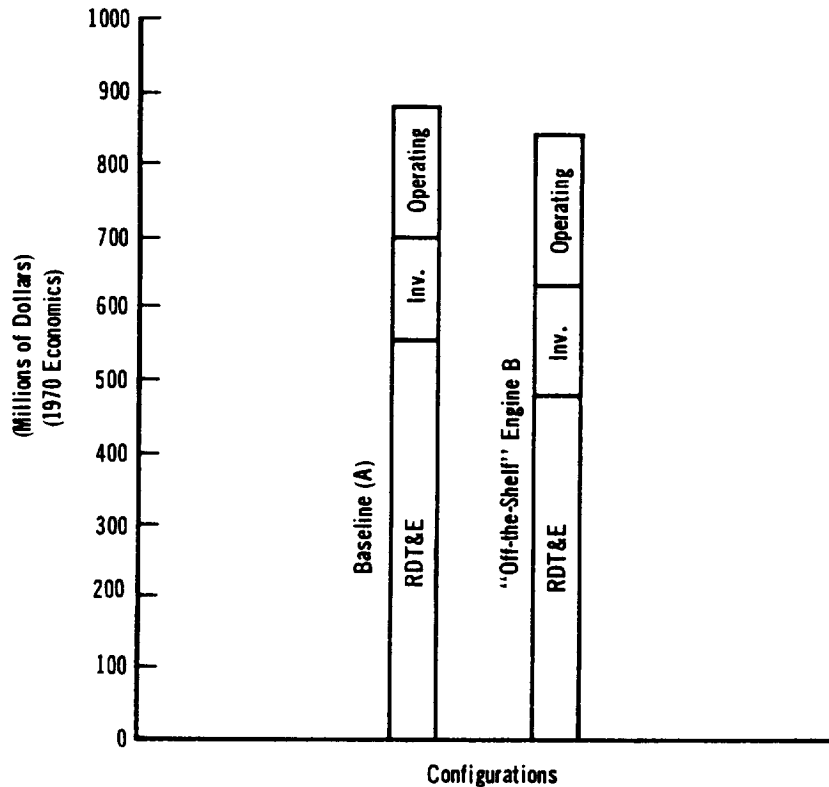
(U) FIGURE 4-167 TOTAL SYSTEM COST BAR CHART FOR B207 CONFIGURATIONS
M = 6.0-8.0, Air Launched, All Body Configuration, Rocket/Ramjet



(U) FIGURE 4-168 TOTAL SYSTEM COST BAR CHART FOR B212 CONFIGURATIONS
M = 6.0, HTO Launched, Wing Body Configuration, Turbojet/Ramjet

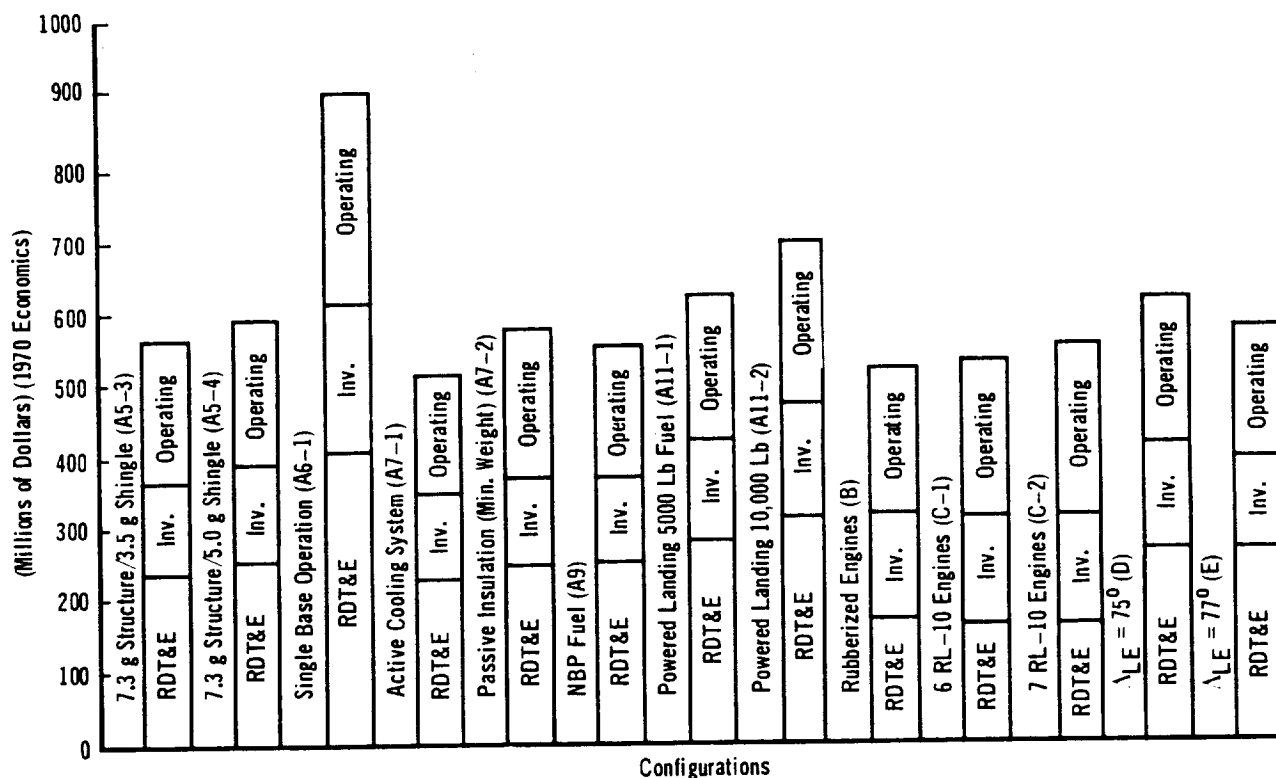
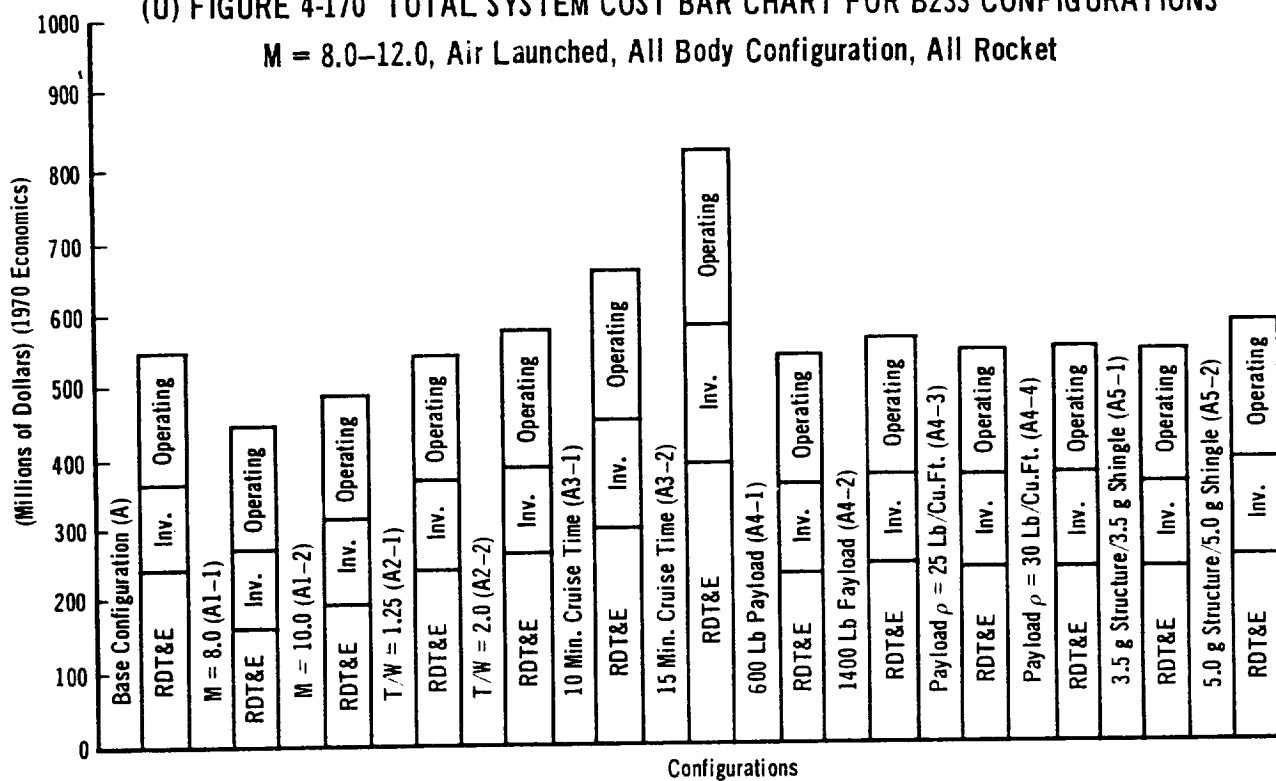


(U) FIGURE 4-169 TOTAL SYSTEM COST BAR CHART FOR B232 CONFIGURATIONS
M = 12.0, Air Launched, All Body Configuration, Rocket/SJ

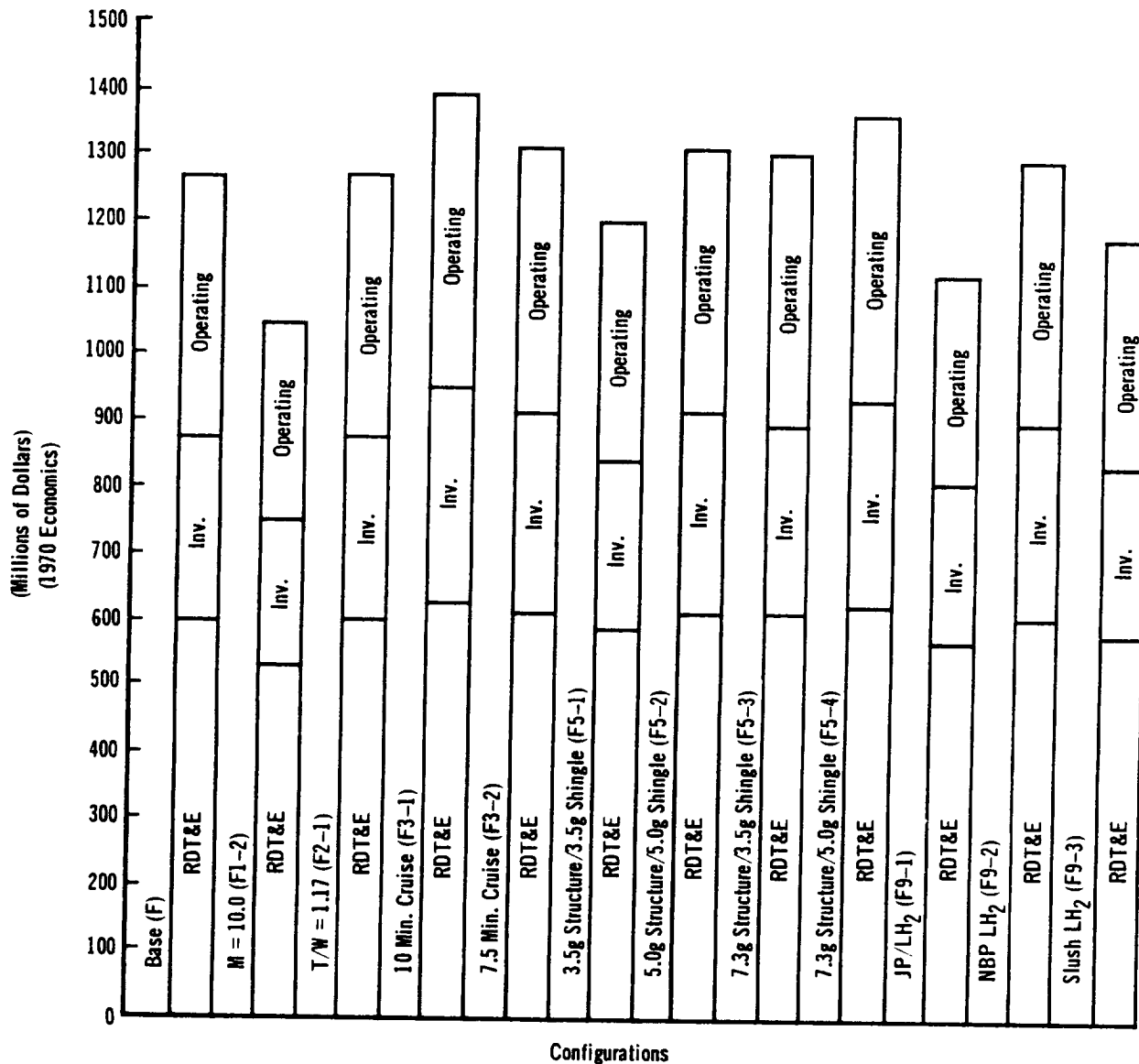


(U) FIGURE 4-170 TOTAL SYSTEM COST BAR CHART FOR B233 CONFIGURATIONS

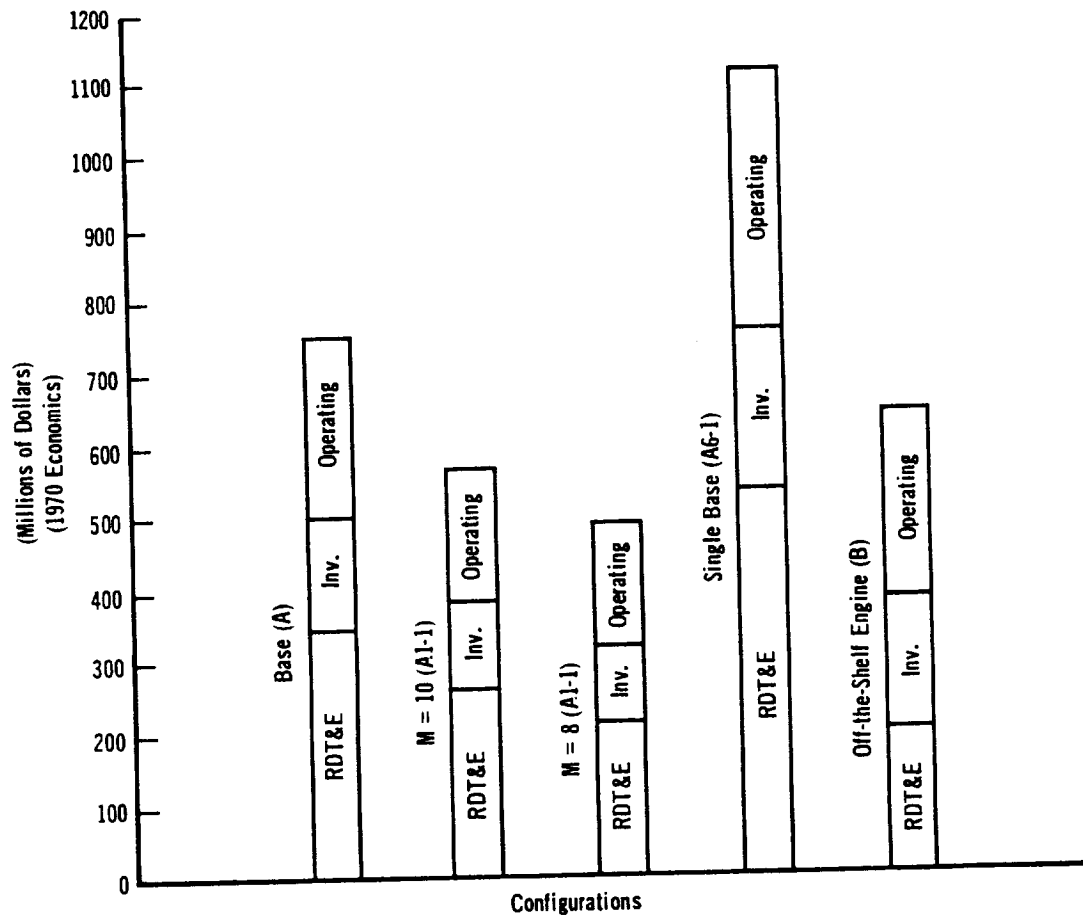
M = 8.0-12.0, Air Launched, All Body Configuration, All Rocket



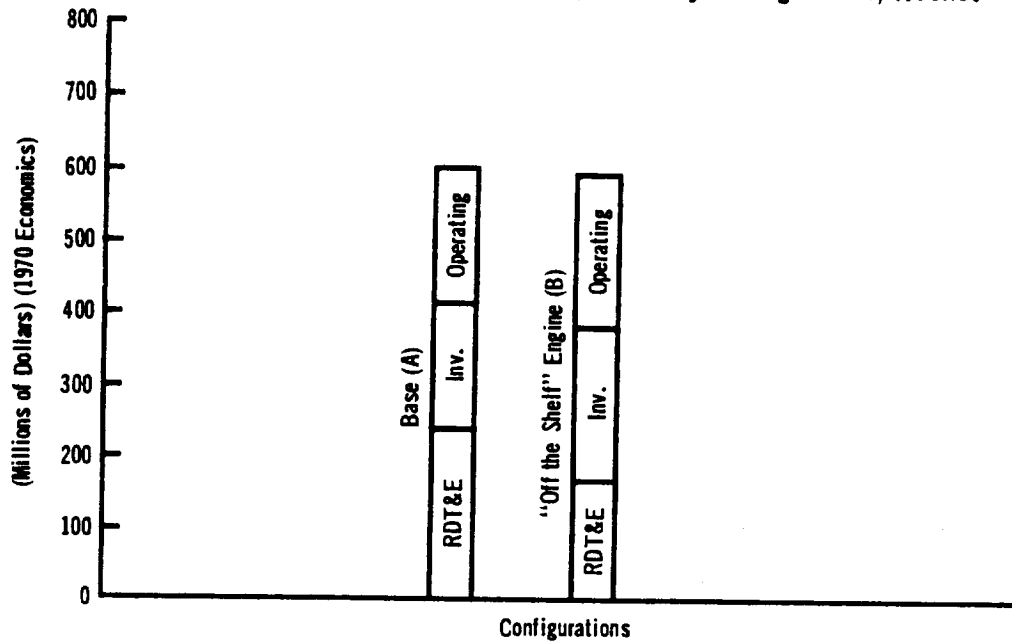
(U) FIGURE 4-171 TOTAL SYSTEM COST BAR CHART FOR B257 CONFIGURATIONS
M = 10.0-12.0, HTO Launched, Blended Wing Body Configuration, Turbojet/CSJ



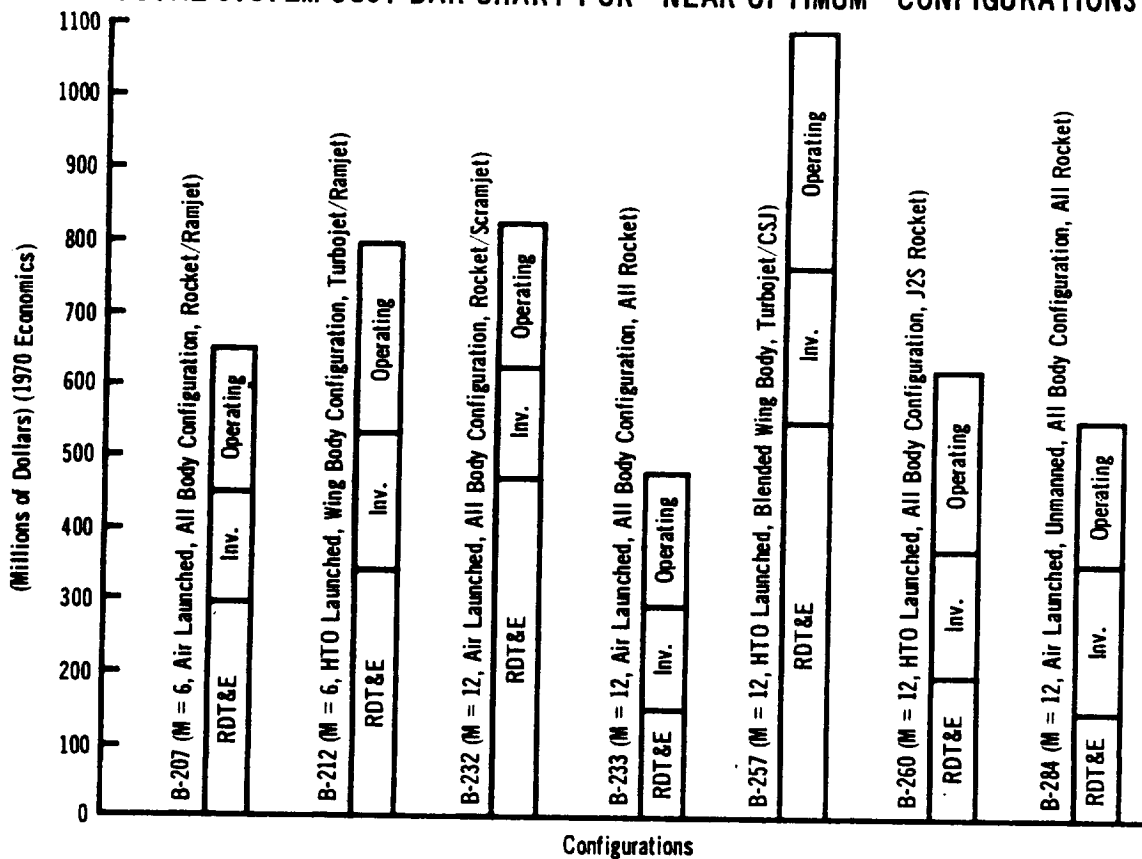
(U) FIGURE 4-172 TOTAL SYSTEM COST BAR CHART FOR B260 CONFIGURATIONS
M = 8.0-12.0, HTO Launched, All Body Configuration, Rocket



(U) FIGURE 4-173 TOTAL SYSTEM COST BAR CHART FOR B284 CONFIGURATIONS
M = 12.0, Air Launched, Unmanned, All Body Configuration, Rocket



(U) FIGURE 4-174
TOTAL SYSTEM COST BAR CHART FOR "NEAR OPTIMUM" CONFIGURATIONS



(U) FIGURE 4-175
DETAILED TOTAL SYSTEM COSTS FOR THE BASELINE CONFIGURATIONS AND
THEIR PARAMETRIC VARIATIONS
(MILLIONS OF 1970 DOLLARS)

Cost Categories and Cost Elements		Configurations													
		B207, M - 6.0-8.0, Air Launched, All Body Configuration Rocket Ramjet								B212 M 6.0, HTO, Wing Body Configuration, Turbojet Ramjet					
		A	A1-1	A1-2	A1-3	A2-1	A2-2	A6-1	B	A	A3-1	A5-1	A5-2	A5-3	A5-4
		Base Configuration Hot Inlet	M = 6 Cool Inlet	M - 7 Cool Inlet	M - 8 Cool Inlet	T W = 1.25	T W 2.00	Single Base Operation	Off-Shell Engine	Base Configuration	10 Min. Test Time	3.5g Structure 3.5g Shingle	5.0g Structure 5.0g Shingle	7.3g Structure 3.5g Shingle	7.3g Structure 5.0g Shingle
I. RDT&E Costs	No. of Flights	200	200	200	200	200	200	200	200	200	200	200	200	200	200
1. Airframe Design and Development															
A. Airframe Design		33.089	30.289	38.013	77.488	33.159	33.130	41.217	34.381	39.302	42.759	37.085	41.750	42.059	45.098
B. Miscellaneous Subsystem Design and Development		4.716	4.660	4.937	6.370	4.742	4.716	5.203	4.982	5.526	6.029	5.021	5.591	5.609	5.683
C. Development Testing		2.904	2.796	3.010	3.545	2.906	2.907	3.072	2.999	3.424	3.605	3.814	3.611	3.631	3.698
D. Test Hardware		4.487	4.202	5.020	8.163	4.498	4.498	5.283	4.870	6.742	7.108	6.332	7.023	7.104	7.401
E. Pre-Delivery Flight Test		3.415	3.414	3.416	3.366	3.415	3.415	3.415	3.414	3.411	3.416	3.416	3.420	3.421	3.418
Sub Total		48.611	45.361	54.396	98.932	48.720	48.666	58.190	50.646	58.405	62.917	55.668	61.395	61.824	65.298
2. Tooling		8.706	8.053	9.336	14.326	8.724	8.723	10.067	9.298	12.349	13.140	11.746	13.039	13.171	13.775
3. Avionics Development		4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669
4. Propulsion Development		244.200	241.020	254.000	314.160	235.720	244.200	251.000	194.000	195.000	195.000	195.000	195.000	195.000	195.000
5. Support Equipment Design and System Integration		17.540	16.887	18.203	27.331	17.536	17.602	19.392	18.137	21.601	22.494	20.990	22.142	22.240	22.856
6. Ground Test Facilities		8.000	8.000	8.000	8.000	8.000	8.000	8.000	8.000	11.500	11.500	11.500	11.500	11.500	11.500
Total		331.726	323.990	348.604	467.418	323.369	331.860	359.318	284.750	303.524	309.720	299.573	307.745	308.404	313.098
II. Investment Costs															
1. Flight Vehicles															
A. Airframe		21.719	20.105	24.347	39.804	21.678	21.773	25.616	23.587	30.839	34.585	30.894	34.231	34.638	36.107
B. Miscellaneous Subsystems		1.168	1.152	1.227	1.622	1.165	1.178	1.300	1.241	1.390	1.529	1.252	1.408	1.413	1.433
C. Propulsion		2.641	2.632	2.838	4.980	2.580	2.664	2.806	3.450	3.326	3.326	3.326	3.326	3.326	3.326
D. Avionics		1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120
Unit Cost (1) Vehicle		26.648	25.209	29.532	47.526	26.643	26.725	30.842	29.398	36.675	40.560	36.592	40.085	40.497	41.966
Unit Cost (3.5 Vehicles)		79.944	75.627	88.596	142.578	79.929	80.175	92.526	88.194	110.025	121.680	109.776	120.255	121.491	125.958
2. Support Costs															
A. Age		11.992	11.344	13.289	21.387	11.989	12.026	6.939	13.229	16.504	18.252	16.466	18.038	18.224	18.894
B. Training Equipment		0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750
C. Initial Stocks (Engines and AGE Spares)		19.686	19.558	21.195	36.999	19.259	19.851	20.336	18.573	18.280	18.455	18.277	18.434	18.452	18.519
D. Initial Training		1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000
E. Initial Transportation		1.717	1.644	1.890	3.097	1.709	1.724	1.799	1.830	2.182	2.377	2.177	2.352	2.373	2.447
Sub Total		35.145	34.296	38.174	63.233	34.707	35.351	30.824	35.382	38.716	40.834	38.670	40.574	40.799	41.610
3. Launch Platform Cost		5.765	5.425	6.881	15.830	5.853	5.764	7.400	6.437	-	-	-	-	-	-
Total		120.854	115.348	133.601	221.641	120.489	121.290	130.750	130.013	148.741	162.514	148.446	160.829	162.290	167.568
III. Operating Costs (5 Year)															
1. Range User Costs		5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566
2. Escort Aircraft and Logistics		2.220	2.220	2.220	2.220	2.220	2.220	2.220	2.220	2.220	2.220	2.220	2.220	2.220	2.220
3. Vehicle Refurbishment Cost		133.240	126.045	147.660	237.630	133.215	133.625	154.210	146.990	183.375	202.800	182.960	200.425	202.485	209.930
4. Propellant Cost		0.140	0.131	0.198	0.652	0.143	0.138	0.253	0.159	0.277	0.180	0.180	0.208	0.209	0.229
5. AGE Maintenance (Labor Only)		1.799	1.702	1.993	3.208	1.798	1.804	1.041	1.984	2.476	2.738	2.470	2.706	2.734	2.834
6. General Purpose Maintenance Support		1.000	1.000	1.000	1.000	1.000	1.000	0.500	1.000	1.000	1.000	1.000	1.000	1.000	1.000
7. Transportation Cost		-	-	-	-	-	-	-	-	-	-	-	-	-	-
8. Pilot Pay and Support Per Pay		17.600	17.400	17.800	20.000	17.600	17.600	18.000	17.800	17.400	17.600	17.400	17.600	17.600	17.600
9. Launch Platform Op Cost		2.540	2.540	2.540	2.540	2.540	2.540	2.540	2.540	-	-	-	-	-	-
Total		164.105	156.604	178.977	272.816	164.082	164.493	184.330	178.259	212.232	232.201	211.796	229.725	231.814	239.379
Grand Total		616.685	595.942	661.182	961.875	607.940	617.643	674.398	593.022	664.497	704.435	659.815	698.299	702.508	720.045

TP8257-90

MCDONNELL AIRCRAFT

4-221

FOLDOUT FRAME 1

(U) FIGURE 4-175 (Continued)
DETAILED TOTAL SYSTEM COSTS FOR THE BASELINE CONFIGURATIONS
AND THEIR PARAMETRIC VARIATIONS
(Millions of 1970 Dollars)

FOLDOUT FRAME 2

Cost Categories and Cost Elements		Configurations												
		B212, M = 6.0, HTO, Wing Body Configuration, Turbojet Ramjet			B232, M = 12.0, Air Launched, All Body Config, Rocket SJ		B233, M = 8.0-12.0, Air Launched, All Body Configuration, Rocket							
		A7-1	A8-1	A9-1	A	B	A	A1-1	A1-2	A2-1	A2-2	A3-1	A3-2	A4-1
		Hot Structure	Non-Integral Tank	JP Fuel	Base Configuration	Off-Shell Rocket	Base Configuration	M = 8.0	M = 10.0	T.W = 1.25	T.W = 2.0	10 Min. Cruise Time	15 Min. Cruise Time	600 Lb Payload
I. RDT&E Costs	No. of Flights	200	200	200	200	200	200	200	200	200	200	200	200	
1. Airframe Design and Development														
A. Airframe Design	49,283	43,075	41,272	71,032	83,152	75,115	39,889	55,686	74,494	75,401	93,049	126,433	74,209	
B. Misc. Subsystem Design and Development	5,858	5,583	5,581	5,718	6,465	5,766	4,246	5,087	5,813	5,695	6,496	7,775	5,725	
C. Development Testing	2,981	3,373	3,331	2,751	2,945	2,785	2,499	2,516	2,669	2,797	3,037	3,407	2,765	
D. Test Hardware	5,234	6,507	6,344	5,077	5,772	5,136	4,924	4,761	4,934	5,169	6,055	7,620	5,078	
E. Pre-Delivery Flight Test	3,389	3,400	3,399	3,387	3,390	3,389	3,353	3,356	3,390	3,389	3,392	3,396	3,389	
Sub-Total	66,755	61,938	59,927	87,965	101,724	92,191	54,911	71,406	91,300	92,451	112,029	148,631	91,166	
2. Tooling	10,260	11,863	11,517	8,479	9,757	8,692	7,033	7,304	8,068	8,768	10,436	13,326	8,580	
3. Avionics Development	4,669	4,669	4,669	4,669	4,669	4,669	4,669	4,669	4,669	4,669	4,669	4,669	4,669	
4. Propulsion Development	195,000	195,000	195,000	385,000	289,000	119,000	76,532	96,460	119,780	142,000	152,000	195,000	118,000	
5. Support Equipment Design & System Integration	23,909	22,123	21,470	19,232	21,333	18,311	14,955	16,206	18,297	18,523	20,873	25,042	17,996	
6. Ground Test Facilities	11,500	11,500	11,500	50,000	50,000	-	-	-	-	-	-	-	-	
Total	312,093	307,098	304,083	555,345	476,483	242,863	158,100	196,125	242,114	266,411	300,007	386,668	240,411	
II. Investment Costs														
1. Flight Vehicles														
A. Airframe	35,240	31,656	30,840	24,497	27,859	24,803	23,971	24,070	23,773	24,956	29,248	36,833	24,502	
B. Misc. Subsystems	1,484	1,405	1,406	1,441	1,644	1,454	1,043	1,267	1,466	1,435	1,656	2,017	1,443	
C. Propulsion	3,326	3,326	3,326	2,734	3,230	1,250	0,858	1,049	1,260	1,430	1,500	1,850	1,230	
D. Avionics	1,120	1,120	1,120	1,120	1,120	1,120	1,120	1,120	1,120	1,120	1,120	1,120	0,859	
Unit Cost (1) Vehicle	41,170	37,507	36,692	29,792	33,853	28,627	26,992	27,506	27,619	28,941	33,524	41,820	28,034	
Unit Cost (3.5 Vehicle)	123,510	112,521	110,076	89,376	101,559	85,881	80,976	82,518	82,857	86,823	100,572	125,460	84,102	
2. Support Costs														
A. AGE	18,527	16,878	16,511	13,406	15,234	12,873	12,146	12,378	12,429	13,023	15,086	18,819	12,615	
B. Training Equipment	0,750	0,750	0,750	0,750	0,750	0,750	0,750	0,750	0,750	0,750	0,750	0,750	0,750	
C. Initial Stocks (Engines and AGE Spares)	18,483	18,318	18,281	20,478	17,673	10,037	7,221	8,581	10,063	11,312	12,009	14,832	9,872	
D. Initial Training	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	
E. Initial Transportation	2,406	2,223	2,182	1,893	2,040	1,625	1,485	1,540	1,603	1,667	1,910	2,382	1,594	
Sub-Total	41,166	39,169	38,724	37,527	36,697	26,285	22,602	24,249	25,845	27,752	30,755	37,783	25,831	
3. Launch Platform Cost	-	-	-	11,702	18,067 (1)	11,693	6,069	8,249	11,874	11,862	20,069 (1)	30,123 (1)	11,348	
Total	164,676	151,690	148,800	138,605	156,323	123,859	109,647	115,016	120,576	126,437	151,396	193,366	121,281	
III. Operating Costs (5 Years)														
1. Range User Cost	5,566	5,566	5,566	5,566	5,566	5,566	5,566	5,566	5,566	5,566	5,566	5,566	5,566	
2. Escort Aircraft and Logistics	2,220	2,220	2,220	2,220	2,220	2,220	2,220	2,220	2,220	2,220	2,220	2,220	2,220	
3. Vehicle Refurbishment Cost	205,850	187,535	183,460	148,960	169,265	143,025	134,960	137,530	138,095	144,705	167,620	209,000	140,170	
4. Propellant Cost	0,203	0,208	0,171	0,431	0,635	0,399	0,173	0,262	0,410	0,405	0,625	1,069	0,387	
5. AGE Maintenance (Labor Only)	2,779	2,532	2,477	2,011	2,285	1,931	1,822	1,857	1,864	1,953	2,263	2,823	1,892	
6. General Purpose Maintenance Support	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	
7. Transportation Cost	-	-	-	1,586	1,586	1,586	1,586	1,586	1,586	1,586	1,586	1,586	1,586	
8. Pilot Pay and Support Per Pay	17,800	17,600	17,600	19,200	20,000	19,800	17,600	18,400	19,200	19,200	20,400	21,800	19,200	
9. Launch Platform Operating Cost	-	-	-	2,540	2,540	2,540	2,540	2,540	2,540	2,540	2,540	2,540	2,540	
Total	235,418	216,661	212,494	183,514	205,097	178,077	167,467	170,961	172,481	179,175	203,820	247,604	174,561	
Grand Total With Propellant	712,187	675,449	665,377	877,464	837,903	544,799	435,214	482,102	535,171	572,023	655,223	827,638	536,253	

TP8257-91

(1) Major modification of C-SA required, priced at 1.5 million dollars per aircraft.

MCDONNELL AIRCRAFT

(U) FIGURE 4-175 (Continued)
DETAILED TOTAL SYSTEM COSTS FOR THE BASELINE CONFIGURATIONS
AND THEIR PARAMETRIC VARIATIONS
(Millions of 1970 Dollars)

Cost Categories and Cost Elements		Configurations													
		B233, M = 8.0-12.0, Air Launched, All Body Configuration, Rocket													
		A4-2	A4-3	A4-4	A5-1	A5-2	A5-3	A5-4	A6-1	A7-1	A7-2	A9-1	A11-1	A11-2	B
		1400 Lb Payload	Payload p = 25 Lb Cu Ft.	Payload p = 30 Lb Cu Ft.	3.5g Structure 3.5g Shingle	5.0g Structure 5.0g Shingle	7.3g Structure 3.5g Shingle	7.3g Structure 5.0g Shingle	Single Base Operation	Active Cooling System	Passive Insulation (Min. Weight)	NBP Fuel	Powered Landing (5,000 Lb Fuel)	Powered Landing (10,000 Lb Fuel)	5 RL-10's
I. RDT&E Costs	No. of Flts	200	200	200	200	200	200	200	200	200	200	200	200	200	200
1. Airframe Design and Development															
A. Airframe Design		75.886	74.857	74.732	72.965	78.802	75.228	79.742	118.990	70.212	78.871	77.570	82.484	93.772	85.680
B. Misc. Subsystem Design and Development		5.806	5.766	5.757	5.814	5.940	5.888	5.967	8.539	5.511	5.774	5.823	6.748	7.005	6.318
C. Development Testing		2.800	2.781	2.779	2.743	3.006	2.838	3.062	3.650	2.741	2.837	2.830	2.853	3.036	3.001
D. Test Hardware		5.194	5.131	5.124	5.036	5.631	5.262	5.762	8.916	4.841	5.183	5.296	5.655	6.378	5.917
E. Pre-Delivery Flight Test		3.389	3.389	3.389	3.388	3.414	3.394	3.417	3.394	3.393	3.393	3.390	3.392	3.392	3.393
Sub-Total		93.075	91.924	91.781	89.946	96.793	92.610	97.950	143.489	86.698	96.058	94.909	100.132	113.583	104.309
2. Tooling		8.804	8.680	8.667	8.447	9.745	8.961	10.075	15.180	8.038	9.869	9.969	10.513	10.105	9.669
3. Avionics Development		4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669
4. Propulsion Development		122.000	119.000	118.720	117.660	124.000	120.840	125.000	219.420	108.120	118.720	114.000	143.100	163.240	24.000
5. Support Equipment Design & System Integration		18.626	18.292	18.269	18.039	18.922	18.612	19.148	26.555	16.638	18.133	18.725	20.093	21.789	20.041
6. Ground Test Facilities		-	-	-	-	-	-	-	-	-	-	-	-	-	-
Total		247.174	242.565	242.106	238.761	254.129	245.692	256.842	409.313	224.163	247.449	241.292	277.250	313.794	163.124
II. Investment Costs															
1. Flight Vehicles															
A. Airframe		25.064	24.758	24.722	24.274	27.229	25.394	27.880	43.190	23.060	25.927	25.570	27.203	30.769	28.589
B. Misc. Subsystems		1.466	1.454	1.452	1.467	1.502	1.488	1.510	2.231	1.385	1.457	1.470	1.726	1.799	1.607
C. Propulsion		1.270	1.250	1.251	1.240	1.280	1.272	1.300	2.067	1.155	1.250	1.200	1.442	1.590	2.300
D. Avionics		1.381	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120
Unit Cost (1) Vehicle		29.181	28.582	28.545	28.101	31.131	29.274	31.810	48.608	26.720	29.754	29.360	31.491	35.778	33.616
Unit Cost (3.5 Vehicles)		87.543	85.746	85.635	84.303	93.393	87.822	95.430	145.824	80.160	89.262	88.080	94.473	105.834	100.848
2. Support Costs															
A. AGE		13.131	12.862	12.845	12.645	14.009	13.173	14.314	10.937	12.024	13.389	13.212	14.171	15.875	15.127
B. Training Equipment		0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750
C. Initial Stocks (Engines & AGE Spares)		10.203	10.036	10.042	9.944	10.361	10.221	10.531	15.563	9.287	10.089	9.721	11.511	12.718	13.013
D. Initial Training		1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000
E. Initial Transportation		1.657	1.624	1.622	1.599	1.756	1.662	1.793	2.516	1.517	1.683	1.656	1.800	2.013	1.935
Subtotal		26.741	26.272	26.145	25.938	27.876	26.806	28.260	30.569	24.578	26.911	26.339	29.232	32.356	31.825
3. Launch Platform Cost		12.039	11.656	11.639	11.489	12.383	12.066	12.789	36.318	10.082	11.675	12.137	19.004	23.254	18.287
Total		126.323	123.684	123.419	121.730	133.652	126.694	136.479	212.711	114.820	127.848	126.556	142.709	161.444	150.960
III. Operating Costs (5 Years)															
1. Range User Cost		5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566
2. Escort Aircraft and Logistics		2.220	2.220	2.220	2.220	2.220	2.220	2.220	2.220	2.220	2.220	2.220	2.220	2.220	2.220
3. Vehicle Refurbishment Cost		145.905	142.910	142.725	140.505	155.655	146.370	159.050	243.040	133.600	148.770	146.800	157.455	176.390	168.080
4. Propellant Cost		0.412	0.398	0.399	0.395	0.425	0.413	0.438	1.319	0.348	0.402	0.415	0.584	0.765	0.621
5. AGE Maintenance (Labor Only)		1.370	1.329	1.327	1.297	1.401	1.376	1.447	1.641	1.003	1.008	1.002	1.126	1.281	1.269
6. General Purpose Maintenance Support		1.000	1.000	1.000	1.000	1.000	1.000	1.000	0.500	1.000	1.000	1.000	1.000	1.000	1.000
7. Transportation Cost		1.586	1.586	1.586	1.586	1.586	1.586	1.586	-	1.586	1.586	1.586	1.586	1.586	1.586
8. Pilot Pay and Support Per Pay		19.200	19.200	19.200	19.200	19.400	19.200	19.400	22.600	18.800	19.200	19.400	20.800	20.800	20.800
9. Launch Platform Operating Cost		2.540	2.540	2.540	2.540	2.540	2.540	2.540	2.540	2.540	2.540	2.540	2.540	2.540	2.540
Total		180.399	177.349	177.163	174.909	190.493	180.871	193.947	279.426	167.463	183.292	181.509	193.277	213.248	203.882
Grand Total		553.896	543.598	542.688	535.400	578.274	553.257	587.268	901.450	506.446	558.589	549.357	613.236	688.486	517.966

①. Major modification of C-5A required, priced at 1.5 million dollars per aircraft

FOLDOUT FRAME 2

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FOLDOUT FRAME 1

(U) FIGURE 4-175 (Continued)
DETAILED TOTAL SYSTEM COSTS FOR THE BASELINE CONFIGURATIONS
AND THEIR PARAMETRIC VARIATIONS
(Millions of 1970 Dollars)

Cost Categories and Cost Elements	Configurations												
	B233, M = 8.0-12.0, Air Launched, All Body Configuration, Rocket				B257, M = 10.0-12.0, HTO Launched, Blended Wing Body Configuration, Turbojet CSJ								
	C-1	C-2	D	E	F	F1-2	F2-1	F3-1	F3-2	F5-1	F5-2	F5-3	F5-4
	6 RL-10's	7RL-10's	75° Leading Edge Sweep	77° Leading Edge Sweep	Base Configuration	M = 10.0	T W = 1.17	10 Minute Cruise Time	15 Minute Cruise Time	3.5g Structure 3.5g Shingle	5.0g Structure 5.0g Shingle	7.3g Structure 3.5g Shingle	7.3g Structure 5.0g Shingle
I. RDT&E Costs No. of Flights No. of Flights	200	200	200	200	200	200	200	200	200	200	200	200	200
1. Airframe Design and Development													
A. Airframe Design	85.845	88.282	85.830	84.301	164.063	112.474	161.032	176.363	169.257	154.391	167.858	166.564	172.821
B. Misc. Subsystem Design & Development	6.021	6.458	6.160	5.743	9.107	7.569	8.614	10.321	9.414	8.704	9.327	9.325	10.042
C. Development Testing	2.989	3.007	2.945	2.803	4.580	3.959	4.615	4.968	4.753	4.419	4.788	4.709	4.936
D. Test Hardware	5.845	5.939	5.701	5.201	12.074	8.999	12.172	14.834	12.800	11.244	13.121	12.767	14.023
E. Pre Delivery Flight Test	3.393	3.392	3.390	3.387	3.337	3.391	3.336	3.345	3.341	3.332	3.340	3.331	3.344
Sub-Total	104.093	107.078	104.026	101.435	193.161	136.392	189.769	209.831	199.565	182.090	198.434	196.696	205.166
2. Tooling	10.017	10.189	9.795	8.936	23.690	17.659	24.184	27.444	25.657	22.064	25.895	25.299	27.394
3. Avionics Development	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669	4.669
4. Propulsion Development	24.000	24.000	130.380	125.000	290.000	290.000	290.000	290.000	290.000	290.000	290.000	290.000	290.000
5. Support Equipment Design & System Integration	20.046	20.478	19.858	19.082	37.421	30.992	38.742	41.642	40.185	35.795	40.157	39.484	41.092
6. Ground Test Facilities	-	-	-	-	53.500	53.500	53.500	53.500	53.500	53.500	53.500	53.500	53.500
Total	162.825	166.414	268.728	259.122	602.441	533.212	600.864	627.086	613.576	588.118	612.655	609.648	621.821
II. Investment Costs													
1. Flight Vehicles													
A. Airframe	28.246	28.676	27.540	25.108	59.438	43.761	59.438	68.625	63.410	54.776	64.045	62.366	68.414
B. Misc. Subsystem	1.530	1.646	1.564	1.447	2.399	1.959	2.255	2.697	2.340	2.283	2.461	2.460	2.670
C. Propulsion	2.760	3.220	1.346	1.280	6.731	5.852	6.975	7.632	7.213	6.088	7.104	7.018	7.314
D. Avionics	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120	1.120
Unit Cost (1) Vehicle	33.656	34.662	31.570	28.955	69.102	52.691	69.788	80.074	74.083	64.267	74.730	72.964	79.518
Unit Cost (3.5 Vehicles)	100.968	103.986	94.710	86.865	207.306	158.076	209.364	240.222	222.249	192.801	224.196	218.892	238.554
2. Support Costs													
A. AGE	15.145	15.598	14.207	13.030	31.096	23.711	31.405	36.033	33.337	28.920	33.629	32.834	35.783
B. Training Equipment	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750
C. Initial Stocks (Engines & AGE Spares)	15.315	17.660	10.842	10.263	36.765	31.631	38.016	41.763	39.399	33.332	38.883	38.373	40.148
D. Initial Training	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000
E. Initial Transportation	1.982	2.079	1.788	1.648	4.166	3.244	4.222	4.811	4.465	3.855	4.483	4.386	4.745
Subtotal	34.192	37.087	28.587	26.691	74.126	60.336	75.393	84.357	78.951	67.857	78.745	77.343	82.426
3. Launch Platform Cost	① 17.937	① 18.425	① 16.683	① 12.710	-	-	-	-	-	-	-	-	-
Total	153.097	159.498	139.980	126.266	281.432	218.412	284.757	324.579	301.200	260.658	302.935	296.235	320.980
III. Operating Costs (5 Years)													
1. Range User Cost	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566
2. Escort Aircraft and Logistics	2.270	2.270	2.270	2.270	2.270	2.270	2.270	2.270	2.270	2.270	2.270	2.270	2.270
3. Vehicle Refurbishment Cost	168.280	173.310	157.850	144.775	345.510	263.455	348.940	400.370	370.415	321.335	373.650	364.820	397.590
4. Propellant Cost	0.598	0.618	0.473	0.437	1.464	0.715	1.340	1.878	1.596	1.224	1.452	1.400	1.528
5. AGE Maintenance (Labor Only)	2.272	2.314	2.131	1.955	4.664	3.557	4.711	5.405	5.000	4.338	5.044	4.925	5.367
6. General Purpose Maintenance Support	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000
7. Transportation Cost	1.586	1.586	1.586	1.586	1.586	1.586	1.586	1.586	1.586	1.586	1.586	1.586	1.586
8. Pilot Pay and Support Per Pay	20.000	20.000	19.600	19.400	20.400	19.200	20.400	21.000	20.800	20.000	20.600	20.500	20.800
9. Launch Platform Operating Cost	2.540	2.540	2.540	2.540	-	-	-	-	-	-	-	-	-
Total	204.062	209.154	192.966	179.479	382.410	297.299	385.763	439.025	408.183	357.269	411.118	402.117	435.657
Grand Total	519.984	535.066	601.674	564.867	1,266.283	1,048.923	1,271.384	1,390.690	1,322.959	1,206.045	1,326.708	1,308.000	1,378.458

① Major modification of C-SA required, priced at 1.5 million dollars per aircraft.

TP8757-93

MCDONNELL AIRCRAFT

FOUO COPY NAME

(U) FIGURE 4-175 (CONTINUED)
DETAILED TOTAL SYSTEM COSTS FOR THE BASELINE CONFIGURATIONS
AND THEIR PARAMETRIC VARIATIONS
(Millions of 1970 Dollars)

Cost Categories and Cost Elements	Configurations									
	B257, M 10.0-12.0, HTO, Blended Wing Body Configuration, Turbojet CSJ			B260, M 8.0-12.0, V HTO Launched, All Body Configuration, Rocket				B284, M 12.0, Air Launched, Unmanned, All Body Configuration, Rocket		
	F9-1	F9-2	F9-3	A	A1-1	A1-2	A6-1	B	A	B
	JP LH ₂	NBP LH ₂	Slush LH ₂	Base Configuration	M 10.0	M 8.0	Single Base	Off-Shell Engine	Base Configuration	Off-Shell Engine
I RDT&E Costs	No. of Flights	200	200	200	200	200	200	200	200	200
1. Airframe Design and Development										
A. Airframe Design	141.956	166.469	149.626	110.290	77.814	58.582	163.110	121.963	68.926	82.352
B. Miscellaneous Subsystem Design and Development	8.651	9.312	8.085	7.542	6.152	5.124	11.433	8.010	5.521	6.244
C. Development Testing	4.229	4.690	4.313	3.336	2.842	2.676	4.066	3.491	2.752	3.013
D. Test Hardware	10.279	12.532	10.674	7.428	5.374	4.728	11.322	8.083	5.017	5.907
E. Pre Delivery Flight Test	3.336	3.335	3.398	3.327	3.387	3.387	3.391	3.352	3.292	3.297
Sub Total	168.451	196.338	176.096	131.923	95.569	74.497	193.322	144.939	85.508	100.813
2. Tooling	20.144	24.987	21.013	12.608	9.230	8.088	19.376	13.960	8.383	10.057
3. Avionics Development	4.669	4.669	4.669	4.669	4.699	4.699	4.699	4.669	7.052	7.052
4. Propulsion Development	290.000	290.000	290.000	171.720	133.560	111.300	287.260	10.000	115.540	24.000
5. Support Equipment Design and System Integration	33.478	39.367	35.128	23.632	20.013	18.335	32.651	26.085	17.686	19.543
6. Ground Test Facilities	53.500	53.500	53.500	-	-	-	-	-	-	-
Total	570.242	608.861	580.406	344.552	263.041	216.889	537.278	199.653	234.169	161.465
II Investment Costs										
1. Flight Vehicles										
A. Airframe	49.950	61.242	52.075	35.584	25.905	22.879	54.706	39.109	24.229	28.550
B. Miscellaneous Subsystems	2.270	2.255	2.101	1.905	1.561	1.275	3.062	2.084	1.387	1.581
C. Propulsion	5.003	7.313	6.577	1.747	1.447	1.243	2.916	1.680	1.219	2.300
D. Avionics	1.120	1.120	1.120	1.220	1.220	1.220	1.220	1.220	1.358	1.358
Unit Cost (1) Vehicle	58.343	71.930	61.873	40.456	30.133	26.617	61.904	44.093	28.193	33.796
Unit Cost (3+5 Vehicles)	175.029	215.790	185.619	121.368	90.399	79.851	185.712	132.279	140.965	168.980
2. Support Costs										
A. Age	26.254	32.369	27.843	18.205	13.560	11.978	13.928	19.842	12.687	15.196
B. Training Equipment	0.750	0.750	0.750	0.750	0.750	0.750	0.750	0.750	-	-
C. Initial Stocks (Engines and AGE Spares)	27.640	39.802	35.669	14.050	11.485	9.899	21.805	13.744	9.802	13.005
D. Initial Training	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000	-	-
E. Initial Transportation	3.444	4.363	3.783	2.294	1.731	1.524	3.251	2.238	1.638	2.043
Sub Total	59.382	78.648	69.045	36.136	28.526	25.151	40.734	32.574	24.187	30.244
3. Launch Platform Cost	-	-	-	2.000	2.000	2.000	2.000	2.000	11.189	17.574
Total	234.411	294.438	254.664	159.504	120.925	107.002	228.446	171.853	176.341	216.798
III Operating Costs (5 Year)										
1. Range User Costs	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566	5.566
2. Escort Aircraft and Logistics	2.220	2.220	2.220	2.220	2.220	2.220	2.220	2.200	2.220	2.200
3. Vehicle Refurbishment Cost	291.715	359.650	309.365	202.280	150.665	133.085	309.520	220.465	155.062	183.948
4. Propellant Cost	0.883	1.389	1.121	0.805	0.496	0.332	2.243	1.137	0.386	0.524
5. AGE Maintenance (Labor Only)	3.938	4.855	4.176	2.731	2.034	1.797	2.089	2.976	1.903	2.258
6. General Purpose Maintenance Support	1.000	1.000	1.000	1.000	1.000	1.000	0.500	1.000	1.000	1.000
7. Transportation Cost	1.586	1.586	1.586	1.547	1.547	1.547	-	1.547	1.586	1.586
8. Pilot Pay and Support Per Pay	20.000	20.600	20.000	21.200	19.800	18.800	24.600	22.000	18.000	18.800
9. Launch Platform Op Cost	-	-	-	0.500	0.500	0.500	0.500	0.500	2.540	2.540
Total	376.908	396.866	345.034	237.849	183.828	164.847	347.238	257.411	188.263	218.442
Grand Total	1,131.561	1,300.165	1,180.104	741.905	567.794	488.738	1,112.962	628.917	598.773	596.705

MCDONNELL AIRCRAFT

4-105

FOLDOUT FRAME 1

REPORT MDC A0013 • 2 OCTOBER 1970
VOLUME III • PART 2(U) FIGURE 4-176
DETAILED TOTAL SYSTEM COSTS FOR THE "NEAR OPTIMUM" CONFIGURATIONS
(Millions of 1970 Dollars)

FOLDOUT FRAME 2

Cost Categories and Cost Elements	"Near Optimum" Configurations						
	B207	B212	B232	B233	B257	B260	B284
	M = 6, Air Launched, All Body Configuration, Rocket/Ramjet	M = 6, HTO Launched, Wing Body Configuration, Turbojet/Ramjet	M = 12, Air Launched, All Body Configuration, Rocket SJ	M = 12, Air Launched, All Body Configuration, All Rocket	M = 12, HTO Launched, Blended Wing Body Configuration, Turbojet/CSJ	M = 12, HTO Launched, All Body Configuration, J2S Rocket	M = 12, Air Launched, Unmanned, All Body Configuration, All Rocket
I RDT&E Costs	200	200	200	200	200	200	200
1. Airframe Design and Development							
A. Airframe Design	42.048	50.510	75.212	76.903	121.472	119.885	72.097
B. Miscellaneous Subsystem Design and Development	5.906	8.804	5.750	5.884	8.175	8.198	5.775
C. Development Testing	3.482	4.767	2.850	2.838	4.276	3.626	2.879
D. Test Hardware	5.988	9.345	5.125	5.264	9.719	8.074	5.248
E. Pre-Delivery Flight Test	3.433	3.446	3.390	3.391	3.662	3.340	3.295
Sub-Total	60.857	76.872	92.327	94.280	147.304	143.123	89.294
2. Tooling	10.270	15.603	8.750	8.922	19.072	13.705	8.769
3. Avionics Development	4.669	4.669	4.669	4.669	4.669	4.669	7.376
4. Propulsion Development	194.000	195.000	289.000	24.000	290.000	10.000	24.000
5. Support Equipment Design and System Integration	20.910	26.884	20.201	18.772	32.435	24.825	18.156
6. Ground Test Facilities	8.000	11.500	50.000	-	53.500	-	-
Total	298.706	330.528	464.947	150.643	546.980	196.322	147.595
II Investment Costs							
1. Flight Vehicles							
A. Airframe	26.132	40.989	25.628	25.400	47.262	38.680	25.344
B. Miscellaneous Subsystems	1.474	1.889	1.466	1.486	2.116	2.071	1.451
C. Propulsion	3.314	3.326	4.489	2.300	4.565	1.680	2.300
D. Avionics	1.446	1.446	1.446	1.446	1.446	1.546	1.684
Unit Cost (1) Vehicle	32.366	47.650	33.029	30.632	55.389	43.977	30.779
Unit Cost (3-5 Vehicles)	97.098	142.950	99.087	91.896	166.167	131.931	153.895
2. Support Costs							
A. Age	14.565	21.443	14.863	13.784	24.925	19.790	13.851
B. Training Equipment	0.750	0.750	0.750	0.750	0.750	0.750	-
C. Initial Stocks (Engines and AGE Spares)	18.027	18.774	23.931	12.878	25.318	13.739	12.885
D. Initial Training	1.000	1.000	1.000	1.000	1.000	1.000	-
E. Initial Transportation	1.965	2.729	2.123	1.784	3.250	2.465	1.899
Sub-Total	36.307	44.696	42.667	30.196	55.243	37.744	28.635
3. Launch Platform Cost	6.914	-	13.200	13.050	-	2.000	12.375
Total	141.164	187.646	154.954	135.142	221.410	171.675	194.905
III Operating Costs (5 Year)							
1. Range User Costs	5.566	5.566	5.566	5.566	5.566	5.566	5.566
2. Escort Aircraft and Logistics	2.220	2.220	2.200	2.220	2.220	2.220	2.220
3. Vehicle Refurbishment Cost	161.830	238.250	165.145	153.160	276.945	219.885	169.285
4. Propellant Cost	0.171	0.186	0.553	0.518	0.727	0.999	0.492
5. AGE Maintenance (Labor Only)	2.185	3.216	2.229	2.068	3.739	2.969	2.078
6. General Purpose Maintenance Support	1.000	1.000	1.000	1.000	1.000	1.000	1.000
7. Transportation Cost	-	-	1.586	1.586	1.586	1.547	1.586
8. Pilot Pay and Support Per Pay	18.200	18.400	19.600	19.600	19.800	21.600	18.400
9. Launch Platform Operating Cost	2.540	-	2.540	2.540	-	0.500	2.540
Total	193.712	268.838	200.439	188.258	311.583	256.286	203.167
Grand Total	633.582	787.012	820.340	474.043	1,079.973	624.283	545.667

MCDONNELL AIRCRAFT

FOLDOUT FRAME 1

FOLDOUT FRAME 2

(U) FIGURE 4-177
WEIGHT SUMMARY AND AIRFRAME AND MISCELLANEOUS SUBSYSTEM COSTS FOR BASELINE
CONFIGURATION AND THEIR PARAMETRIC VARIATIONS AND THE "NEAR OPTIMUM" CONFIGURATIONS

Configuration No.		DCPR Weight (Lb) (1)	DCPR Weight (Kilograms)	Empty Weight (Lb)	Empty Weight (Kilograms)	OWE (Lb)	OWE (Kilograms)	TOGW (Lb)	TOGW (Kilograms)	Airframe Costs (2)		Misc. Subsystem Cost (2)	
										RDTE&E (\$3)	Inv. (\$)	RDTE&E (\$4)	Inv. (\$)
B207A (5)	Base Configuration, Hot Inlet	15,900	7,212	21,175	9,605	22,860	10,369	38,430	17,432	35,993	21,719	4,716	1,168
B207A1-1	M = 6.0, Cool Inlet	14,509	6,581	19,761	8,963	21,440	9,725	36,166	16,405	33,085	20,305	4,660	1,152
B207A1-2	M = 7.0, Cool Inlet	17,064	7,740	22,600	10,251	24,350	11,045	45,876	20,809	41,023	24,347	4,937	1,227
B207A1-3	M = 8.0, Cool Inlet	31,994	14,512	46,425	21,058	48,500	21,999	105,530	47,868	81,033	39,804	6,370	1,622
B207A2-1	T/W = 1.25	15,961	7,240	21,176	9,605	22,860	10,369	39,020	17,699	36,065	21,678	4,742	1,165
B207A2-2	T/W = 2.00	15,930	7,226	21,265	9,646	23,000	10,433	38,425	17,429	36,037	21,773	4,716	1,178
B207A6-1	Single Base Operation	19,907	9,030	25,305	11,478	27,100	12,292	49,332	22,377	44,289	25,616	5,203	1,300
B207B	"Off the Shelf" Engine	17,024	7,722	22,465	10,190	24,200	10,977	42,912	19,465	37,380	23,587	4,982	1,241
B212A (6)	Base Configuration	20,957	9,506	30,673	13,913	32,540	14,760	37,223	16,884	42,726	30,839	5,526	1,390
B212A3-1	10 Minute Test Time	23,036	10,449	32,864	14,907	34,850	15,808	41,520	18,833	46,364	34,585	6,029	1,529
B212A5-1	3.5g Structure/3.5g Shingle	19,102	8,665	29,152	13,223	31,000	14,061	35,334	16,027	40,899	30,894	5,021	1,252
B212A5-2	5.0g Structure, 5.0g Shingle	22,319	10,124	32,056	14,540	33,940	15,395	38,949	17,667	45,361	34,231	5,591	1,408
B212A5-3	7.3g Structure/3.5g Shingle	22,552	10,229	32,294	14,648	34,200	15,513	39,233	17,796	45,690	34,638	5,609	1,413
B212A5-4	7.3g Structure, 5.0g Shingle	24,147	10,953	33,909	15,381	35,820	16,248	41,320	18,742	48,796	36,107	5,683	1,433
B212A7-1	Hot Structure	26,879	12,192	36,693	16,644	38,650	17,531	43,543	19,751	52,264	35,240	5,868	1,484
B212A8-1	Non-Integral Tank	21,653	9,822	32,018	14,523	33,900	15,377	38,910	17,649	46,448	31,656	5,583	1,405
B212A9-1	JP Fuel	20,623	9,354	30,322	13,754	32,217	14,613	38,480	17,454	44,603	30,840	5,581	1,406
B232A (7)	Base Configuration	19,013	8,624	24,870	11,281	26,730	12,125	78,010	35,385	73,783	24,497	5,718	1,441
B232B	"Off the Shelf" Engine	22,727	10,309	29,900	13,562	31,870	14,456	100,449	45,563	86,097	27,859	6,465	1,644
B233A (8)	Base Configuration	19,933	9,041	22,725	10,308	24,600	11,158	77,950	35,358	77,900	24,803	5,766	1,454
B233A1-1	M = 8.0	13,743	6,234	15,760	7,149	17,450	7,915	40,460	18,352	42,388	23,971	4,246	1,043
B233A1-2	M = 10.0	15,961	7,240	18,465	8,376	20,170	9,149	54,990	24,943	58,202	24,070	5,087	1,267
B233A2-1	T/W = 1.25	19,477	8,835	22,685	10,290	24,560	11,140	79,160	35,906	77,163	23,773	5,813	1,466
B233A2-2	T/W = 2.0	20,002	9,073	23,201	10,524	25,080	11,376	79,080	35,670	78,198	24,956	5,695	1,435
B233A3-1	10 Min. Cruise Time	25,160	11,412	28,595	12,970	30,700	13,925	113,790	51,614	96,086	29,248	6,496	1,656
B233A3-2	15 Min. Cruise Time	34,819	15,794	39,310	17,831	41,800	18,960	180,820	82,019	129,840	36,833	7,775	2,017
B233A4-1	600 Lb Payload	19,651	8,914	22,423	10,171	23,880	10,832	75,650	34,314	76,974	24,502	5,725	1,443
B233A4-2	1400 Lb Payload	20,228	9,175	23,040	10,451	25,320	11,485	80,260	36,405	78,686	25,064	5,806	1,466
B233A4-3	Payload ρ 25 Lb Cu. Ft (400 kg/m ³)	19,899	9,026	22,687	10,291	24,550	11,136	77,770	35,276	77,638	24,758	5,766	1,454
B233A4-4	Payload ρ 30 Lb Cu. Ft (481 kg/m ³)	19,852	9,005	22,644	10,271	24,500	11,130	77,590	35,194	77,511	24,722	5,757	1,452
B233A5-1	3.5g Structure 3.5g Shingle	19,301	8,755	22,095	10,022	23,980	10,877	76,590	34,741	75,708	24,274	5,814	1,467
B233A5-2	5.0g Structure 5.0g Shingle	21,147	9,592	24,070	10,918	26,000	11,793	82,550	37,444	81,808	27,229	5,940	1,502
B233A5-3	7.3g Structure 3.5g Shingle	20,350	9,231	23,220	10,532	25,290	11,471	80,440	36,487	78,066	25,394	5,888	1,488
B233A5-4	7.3g Structure 5.0g Shingle	21,909	9,938	24,915	11,301	26,860	12,183	85,260	38,673	83,004	27,880	5,967	1,510
B233A6-1	Single Base Operation	38,232	17,342	43,460	19,713	46,190	20,951	222,120	100,752	122,640	43,190	8,539	2,231
*B23A7-1	Active Cooling System	15,780	7,158	18,446	8,367	20,910	9,485	67,210	30,486	72,953	23,060	5,511	1,385

TPB257-86

FOLDOUT FRAME 1

FOLDOUT FRAME 2

(U) FIGURE 4-177 (Continued)
WEIGHT SUMMARY, AIRFRAME AND MISCELLANEOUS SUBSYSTEMS COSTS FOR BASELINE CONFIGURATIONS, THEIR PARAMETRIC VARIATIONS AND "NEAR OPTIMUM" CONFIGURATIONS

Configuration No.		DCPR Weight (Lb) (1)	DCPR Weight (Kilograms)	Empty Weight (Lb)	Empty Weight (Kilograms)	OWE (Lb)	OWE (Kilograms)	TOGW (Lb)	TOGW (Kilograms)	Airframe Costs (2)		Misc. Subsystem Cost (2)	
										RD&E (\$3)	Inv. (\$)	RD&E (\$4)	Inv. (\$)
B233A7-2	Passive Insulation (Min. Weight)	19,505	8,847	22,310	10,120	24,190	10,972	77,830	35,303	81,708	25,927	5,774	1,457
B233A9-1	NBP Fuel	20,709	9,393	23,605	10,707	25,500	11,567	80,911	36,701	80,400	25,570	5,823	1,470
B233A11-1	Powered Landing 5,000 Lb (2,268 kg) Fuel	22,882	10,379	26,780	12,147	28,790	13,059	106,690	48,394	85,337	27,203	6,748	1,726
B233A11-2	Powered Landing 10,000 Lb (4,536 kg) Fuel	26,462	12,003	30,875	14,005	33,030	14,982	135,030	61,249	96,808	30,769	7,005	1,799
B233B	5 RL-10 Engines, T/M = 1.117	23,392	10,608	26,625	12,074	28,660	12,997	101,910	46,216	88,681	28,589	6,318	1,607
B233C-1	6 RL-10 Engines, T/M = 1.375	23,432	10,629	26,655	12,091	28,680	13,009	99,580	45,169	88,834	28,246	6,021	1,530
B233C-2	7 RL-10 Engines, T/M = 1.550	24,136	10,948	27,675	12,553	29,730	13,485	102,830	46,643	91,289	28,676	6,458	1,646
B233D	75° Leading Edge Sweep	23,196	10,522	26,265	11,914	28,220	12,800	91,220	41,377	88,775	27,540	6,160	1,564
B233E	77° Leading Edge Sweep	21,486	9,746	24,445	11,088	26,380	11,966	84,730	38,433	87,104	25,108	5,743	1,447
B257F (9)	Base Configuration	59,664	27,063	80,055	36,312	82,600	37,467	117,830	53,447	168,643	58,852	9,107	2,399
B257F1-2	M = 10.0	42,083	19,089	58,004	26,310	60,000	27,216	77,200	35,017	116,433	43,761	7,569	1,959
B257F2-1	T/W = 1.17	61,996	28,120	85,106	38,603	87,600	39,735	119,847	54,362	165,647	59,438	8,614	2,255
B257F3-1	10 Min. Cruise Time	71,559	32,459	96,195	43,633	99,000	44,906	144,202	65,409	181,331	68,625	10,321	2,697
B257F3-2	7.5 Min. Cruise Time	66,526	30,176	90,550	41,073	93,200	42,275	131,598	59,692	174,010	63,410	9,414	2,340
B257F5-1	3.5g Structure/3.5g Shingle	54,493	24,718	74,255	33,682	76,660	34,772	106,120	48,135	158,810	54,776	8,704	2,283
B257F5-2	5.0g Structure/5.0g Shingle	66,644	30,229	90,545	41,071	93,100	42,229	128,050	58,083	172,846	64,045	9,327	2,461
B257F5-3	7.3g Structure/3.5g Shingle	64,760	29,375	87,936	39,887	90,470	41,037	124,170	56,323	171,273	62,366	9,325	2,460
B257F5-4	7.3g Structure/5.0g Shingle	70,317	31,895	94,195	42,726	96,800	43,908	133,575	60,589	177,757	68,414	10,042	2,670
B257F9-1	JP/LH2	49,759	22,570	65,761	29,829	68,400	31,026	105,310	47,768	146,185	49,950	8,651	2,270
B257F9-2	NBP/LH2	63,992	29,026	87,485	39,683	90,000	40,823	123,415	55,980	171,159	61,242	9,312	2,255
B257F9-3	Slush LH2	52,474	23,802	71,880	32,604	74,200	33,657	101,177	45,893	153,939	52,075	8,085	2,101
B260A (10)	HTO Base Configuration	31,831	14,438	35,700	16,193	37,900	17,191	147,250	66,791	113,626	35,584	7,542	1,905
B260A1-1	M = 10.0	22,629	10,264	26,640	12,084	28,600	12,973	94,700	42,955	80,656	25,905	6,152	1,561
B260A1-2	M = 8.0	18,916	8,580	22,898	10,386	24,650	11,181	68,900	31,253	61,258	22,879	5,124	1,275
B260A6-1	Single Base Operation	54,202	24,586	62,375	28,293	65,550	29,733	364,550	165,357	167,176	54,706	11,433	3,062
B260B	"Off the Shelf" Engine	35,643	16,167	42,398	19,231	44,800	20,321	189,050	85,752	125,454	39,109	8,010	2,084
B284A (11)	Base Configuration	18,421	8,356	21,410	9,711	23,190	10,519	74,590	33,833	71,678	24,229	5,521	1,387
B284B	"Off the Shelf" Engine	22,720	10,306	25,885	11,741	27,810	12,614	99,310	45,046	85,365	28,550	6,244	1,581
B207	M = 6, Air Launched, All Body Configuration, Rocket/Ramjet	22,656	10,276	28,566	12,956	30,825	13,981	51,725	23,460	45,530	26,132	5,906	1,474
B212	M = 6, HTO Launched, Wing Body Configuration, Turbojet/Ramjet	33,011	14,972	44,529	20,196	47,159	21,389	55,629	25,231	55,277	40,989	8,804	1,889
B232	M = 12, Air Launched, All Body Configuration, Rocket/Scramjet	20,793	9,432	26,650	12,088	29,050	13,177	88,000	39,916	78,062	25,628	5,750	1,466
B233	M = 12, Air Launched, All Body Configuration, All Rocket	20,340	9,226	23,210	10,528	25,660	11,639	87,000	39,463	79,741	25,400	5,844	1,486
B257	M = 12, HTO Launched, Blended Wing Body Configuration, Turbojet/CSJ	44,103	20,005	62,030	28,136	64,820	29,402	99,320	45,051	125,748	47,262	8,175	2,116
B260	M = 12, HTO Launched, All Body Configuration, J2S Rocket	34,881	15,822	38,750	17,577	41,200	18,688	167,000	75,750	123,511	38,680	8,198	2,071
B284	M = 12, Air Launched, Unmanned, All Body Configuration, All Rocket	18,861	8,555	21,850	9,911	24,250	10,999	82,500	37,421	74,976	25,344	5,775	1,451

- (1) Defense Contractor's Progress Report Weight
(2) Millions of 1970 Dollars
(3) Includes Airframe Design and Development Tests
(4) Includes Miscellaneous Subsystem Design and Development

- (5) B207, M = 6.0-8.0, Air Launched, All Body Configuration, Rocket/Ramjet
(6) B212, M = 6.0, HTO Launched, Wing Body Configuration, Turbojet/Ramjet
(7) B232, M = 12, Air Launched, All Body Configuration, Rocket SJ
(8) B233, M = 8.0-12.0, Air Launched, All Body Configuration, Rocket

- (9) B257, M = 10.0-12.0, HTO Launched, Blended Wing Body Configuration, Turbojet/CSJ
(10) B260, M = 8.0-12.0, V/HTO Launched, All Body Configuration, Rocket
(11) B284, M = 12.0, Air Launched, Unmanned, All Body Configuration, Rocket

MCDONNELL AIRCRAFT

FOLDOUT FRAME 1

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(U) FIGURE 4-178

PROPULSION SYSTEM PARAMETERS AND COSTS FOR THE BASELINE CONFIGURATIONS
AND THEIR PARAMETRIC VARIATIONS AND THE "NEAR OPTIMUM" CONFIGURATIONS

FOLDOUT FRAME 2

Configuration Number	Configuration Description	Type of Engine				Operating Mach Number		Propellant		RDT&E Costs (Dollars) (1)			Investment Cost (Dollars) (1)			Propellant System Ground Test Facilities Cost (Dollars)	
		Acceleration and Climb		Cruise		Acceleration and Climb	Cruise	Fuel	Oxidizer	Acceleration and Climb Engine	Cruise Engine	Total	Acceleration and Climb Engine	Cruise Engine	Total		
		Number	Type	Number	Type												
B207A (2)	Base Hot Inlet	1	LR-129	1	Square Ramjet	0-6	6	LH ₂	LOX	74,200							
B207A1-1	M = 6.0	1	LR-129	1	Square Ramjet	0-6	6	LH ₂	LOX	71,020	170,000	244,200	0.811	1.830	2,641	8,000	
B207A1-2	M = 7.0	1	LR-129	1	Square Ramjet	0-7	7	LH ₂	LOX	84,000	170,000	241,020	0.795	1.837	2,632	8,000	
B207A1-3	M = 8.0	1	LR-129	1	Square Ramjet	0-8	8	LH ₂	LOX	144,160	170,000	254,000	0.943	1.895	2,838	8,000	
B207A2-1	T W = 1.25	1	LR-129	1	Square Ramjet	0-6	6	LH ₂	LOX	65,720	170,000	314,160	1.452	3.528	4,980	8,000	
B207A2-2	T W = 2.00	1	LR-129	1	Square Ramjet	0-6	6	LH ₂	LOX	90,000	170,000	235,720	0.731	1.849	2,580	8,000	
B207A6-1	Single Base Operation	1	LR-129	1	Square Ramjet	0-6	6	LH ₂	LOX	89,000	170,000	260,000	0.900	1.837	2,737	8,000	
B207B	"Off the Shelf" Engine	3	RL-10	1	Square Ramjet	0-6	6	LH ₂	LOX	24,000	170,000	259,000	0.975	1.831	2,806	8,000	
												194,000	1.380	2.070	3,450	8,000	
B212A (3)	Base	2	F100-GE-100	1	Square Ramjet	0-3	1-6	LH ₂	-	25,000	170,000	195,000	1.166	2.160	3,326	11,500	
B212A3-1	10 Minute Test Time	2	F100-GE-100	1	Square Ramjet	0-3	1-6	LH ₂	-	25,000	170,000	195,000	1.166	2.160	3,326	11,500	
B212A5-1	3.5 g Structure 3.5 g Shingle	2	F100-GE-100	1	Square Ramjet	0-3	1-6	LH ₂	-	25,000	170,000	195,000	1.166	2.160	3,326	11,500	
B212A5-2	5.0 g Structure 5.0 g Shingle	2	F100-GE-100	1	Square Ramjet	0-3	1-6	LH ₂	-	25,000	170,000	195,000	1.166	2.160	3,326	11,500	
B212A5-3	7.3 g Structure 3.5 g Shingle	2	F100-GE-100	1	Square Ramjet	0-3	1-6	LH ₂	-	25,000	170,000	195,000	1.166	2.160	3,326	11,500	
B212A5-4	7.3 g Structure 5.0 g Shingle	2	F100-GE-100	1	Square Ramjet	0-3	1-6	LH ₂	-	25,000	170,000	195,000	1.166	2.160	3,326	11,500	
B212A7-1	Hot Structure (Non-Integral)	2	F100-GE-100	1	Square Ramjet	0-3	1-6	LH ₂	-	25,000	170,000	195,000	1.166	2.160	3,326	11,500	
B212A8-1	Non-Integral Tank	2	F100-GE-100	1	Square Ramjet	0-3	1-6	LH ₂	-	25,000	170,000	195,000	1.166	2.160	3,326	11,500	
B212A9-1	JP Fuel	2	F100-GE-100	1	Square Ramjet	0-3	1-6	LH ₂ & JP	-	25,000	170,000	195,000	1.166	2.160	3,326	11,500	
B232A (4)	Base	1	LR-129	1	SJ	0-12	12	LH ₂	LOX	120,000	265,000	385,000	1.272	1.462	2,734	50,000	
B232B	"Off the Shelf" Engine	6	RL-10	1	SJ	0-12	12	LH ₂	LOX	24,000	265,000	289,000	2.760	0.470	3,230	50,000	
B233A (5)	Base	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	119,000	-	119,000	1.250	-	1,250	-	
B233A1-1	M = 8.0	1	LR-129	Same	Same	0-8	8	LH ₂	LOX	76,532	-	76,532	0.858	-	0,858	-	
B233A1-2	M = 10.0	1	LR-129	Same	Same	0-10	10	LH ₂	LOX	96,460	-	96,460	1.049	-	1,049	-	
B233A2-1	T W = 1.25	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	119,780	-	119,780	1.260	-	1,260	-	
B233A2-2	T W = 2.0	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	142,000	-	142,000	1.430	-	1,430	-	
B233A3-1	10 Minute Cruise Time	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	152,000	-	152,000	1.500	-	1,500	-	
B233A3-2	15 Minute Cruise Time	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	195,000	-	195,000	1.850	-	1,850	-	
B233A4-1	600 Lb Payload	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	118,000	-	118,000	1.230	-	1,230	-	
B233A4-2	1400 Lb Payload	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	122,000	-	122,000	1.270	-	1,270	-	
B233A4-3	Payload p = 25 Lb cu ft	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	119,000	-	119,000	1.250	-	1,250	-	
B233A4-4	Payload p = 30 Lb cu ft	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	118,720	-	118,720	1.251	-	1,251	-	
B233A5-1	3.5 g Structure 3.5 g Shingle	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	117,660	-	117,660	1.240	-	1,240	-	
B233A5-2	5.0 g Structure 5.0 g Shingle	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	124,000	-	124,000	1.280	-	1,280	-	
B233A5-3	7.3 g Structure 3.5 g Shingle	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	120,840	-	120,840	1.272	-	1,272	-	
B233A5-4	7.3 g Structure 5.0 g Shingle	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	125,000	-	125,000	1.300	-	1,300	-	
B233A6-1	Single Base Operation	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	219,420	-	219,420	2.067	-	2,067	-	
B233A7-1	Active Cooling System	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	108,120	-	108,120	1.155	-	1,155	-	
B233A7-2	Passive Insulation (Minimum Weight)	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	118,720	-	118,720	1.250	-	1,250	-	
B233A9	NBP Fuel	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	114,000	-	114,000	1.200	-	1,200	-	
B233A11-1	Powered Landing 5000 Lb Fuel	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	143,100	-	143,100	1.442	-	1,442	-	
B233A11-2	Powered Landing 10,000 Lb Fuel	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	163,240	-	163,240	1.590	-	1,590	-	
B233B	5 RL-10 Engines T W 1.117	5	RL-10	Same	Same	0-12	12	LH ₂	LOX	24,000	-	24,000	2.300	-	2,300	-	
B233C-1	6 RL-10 Engines T W 1.375	6	RL-10	Same	Same	0-12	12	LH ₂	LOX	24,000	-	24,000	2.760	-	2,760	-	
B233C-2	7 RL-10 Engines T W 1.550	7	RL-10	Same	Same	0-12	12	LH ₂	LOX	24,000	-	24,000	3.220	-	3,220	-	
B233D	ALE = 75°	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	130,380	-	130,380	1.246	-	1,246	-	
B233E	ALE = 77°	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	125,380	-	125,380	1.280	-	1,280	-	

TP9257-142

MCDONNELL AIRCRAFT

FOLDOUT FRAME 2

FOLDOUT FRAME 1

(U) FIGURE 4-178 (CONTINUED)
PROPULSION SYSTEM PARAMETERS AND COSTS FOR THE BASELINE CONFIGURATIONS AND THEIR
PARAMETRIC VARIATIONS AND THE "NEAR OPTIMUM" CONFIGURATIONS

Configuration Number	Configuration Description	Type of Engine				Operating Mach Number		Propellant		RDT&E Costs (Dollars) ①			Investment Cost (Dollars) ①			Propellant System Ground Test Facilities Cost (Dollars) ①	
		Acceleration and Climb		Cruise		Acceleration and Climb	Cruise	Fuel	Oxidizer	Acceleration and Climb Engine	Cruise Engine	Total	Acceleration and Climb Engine	Cruise Engine	Total		
		Number	Type	Number	Type												
B257F ⑥	Base	5	F100-GE-100	1	CSJ	0-3	1-12	LH ₂	-	25.000	265.000	290.000	2.915	3.816	6.731	53.500	
B257F1-2	M = 10.0	4	F100-GE-100	1	CSJ	0-3	1-10	LH ₂	-	25.000	265.000	290.000	2.332	3.520	5.852	53.500	
B257F2-1	T/W = 1.17	6	F100-GE-100	1	CSJ	0-3	1-12	LH ₂	-	25.000	265.000	290.000	3.498	3.477	6.975	53.500	
B257F3-1	10 Minute Cruise	6	F100-GE-100	1	CSJ	0-3	1-12	LH ₂	-	25.000	265.000	290.000	3.498	4.134	7.632	53.500	
B257F3-2	7.5 Minute Cruise	6	F100-GE-100	1	CSJ	0-3	1-12	LH ₂	-	25.000	265.000	290.000	3.599	3.816	7.213	53.500	
B257F5-1	3.5 g Structure/3.5 g Shingle	5	F100-GE-100	1	CSJ	0-3	1-12	LH ₂	-	25.000	265.000	290.000	2.915	3.173	6.088	53.500	
B257F5-2	5.0 g Structure/5.0 g Shingle	6	F100-GE-100	1	CSJ	0-3	1-12	LH ₂	-	25.000	265.000	290.000	3.498	3.606	7.104	53.500	
B257F5-3	7.3 g Structure/3.5 g Shingle	6	F100-GE-100	1	CSJ	0-3	1-12	LH ₂	-	25.000	265.000	290.000	3.498	3.520	7.018	53.500	
B257F5-4	7.3 g Structure/5.0 g Shingle	6	F100-GE-100	1	CSJ	0-3	1-12	LH ₂	-	25.000	265.000	290.000	3.498	3.816	7.314	53.500	
B257F9-1	JP LH ₂	4	F100-GE-100	1	CSJ	0-3	1-12	LH ₂ & JP	-	25.000	265.000	290.000	2.332	2.671	5.003	53.500	
B257F9-2	NBP LH ₂	6	F100-GE-100	1	CSJ	0-3	1-12	LH ₂	-	25.000	265.000	290.000	3.498	3.815	7.313	53.500	
B257F9-3	Slush LH ₂	6	F100-GE-100	1	CSJ	0-3	1-12	LH ₂	-	25.000	265.000	290.000	3.498	3.079	6.577	53.500	
B260A ⑦	Base	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	171.720	-	171.720	1.747	-	1.747	-	
B260A1-1	M = 10	1	LR-129	Same	Same	0-10	10	LH ₂	LOX	133.560	-	133.560	1.447	-	1.447	-	
B260A1-2	M = 8	1	LR-129	Same	Same	0-8	8	LH ₂	LOX	111.300	-	111.300	1.243	-	1.243	-	
B260A6-1	Single Base	1	LR-129	Same	Same	0-12	12	LH ₂	LOX	287.260	-	287.260	2.916	-	2.916	-	
B260B	"Off the Shelf" Engine	1	J2S	Same	Same	0-12	12	LH ₂	LOX	10.000	-	10.000	1.680	-	1.680	-	
B284A ⑧	Base	1	RL-129	Same	Same	0-12	12	LH ₂	LOX	115.540	-	115.540	1.219	-	1.219	-	
B284B	"Off the Shelf" Engine	5	RL-10	Same	Same	0-12	12	LH ₂	LOX	24.000	-	24.000	2.300	-	2.300	-	
"Near Optimum" Configurations	B-207	M = 6, Air Launched, All Body Configuration, Rocket/Ramjet	3	RL-10	1	Square Ramjet	0-6	6	LH ₂	LOX	24.000	170.000	194.000	1.380	1.934	3.314	8.000
	B-212	M = 6, HTO Launched, Wing Body Configuration, Turbojet/Ramjet	2	F100-GE-100	1	Square Ramjet	0-3	1-6	LH ₂ & JP	-	25.000	170.000	195.000	1.166	2.166	3.326	11.500
	B-232	M = 12, Air Launched, All Body Configuration, Rocket/Scramjet	5	RL-10	1	Scramjet or CSJ	0-12	12	LH ₂	LOX	24.000	265.000	289.000	2.300	2.189	4.489	50.000
	B-233	M = 12, Air Launched, All Body Configuration, All Rocket	5	RL-10	Same	Same	0-12	12	LH ₂	LOX	24.000	-	24.000	2.300	-	2.300	-
	B-257	M = 12, HTO Launched, Blended Wing Body Configuration, Turbojet/CSJ	4	F100-GE-100	1	CSJ	0-3	1-12	LH ₂	-	25.000	265.000	290.000	2.332	2.233	4.565	53.500
	B-260	M = 12, HTO Launched, All Body Configuration, J2S Rocket	1	J2S	Same	Same	0-12	12	LH ₂	LOX	10.000	-	10.000	1.680	-	1.680	-
	B-284	M = 12, Air Launched, Unmanned, All Body Configuration, All Rocket	5	RL-10	Same	Same	0-12	12	LH ₂	LOX	24.000	-	24.000	2.300	-	2.300	-

① Millions of 1970 Dollars

② B207, M = 6.0-8.0, Air Launched, All Body Configuration, Rocket/Ramjet

③ B212, M = 6.0, HTO Launched, Wing Body Configuration, Turbojet/Ramjet

④ B232, M = 12.0, Air Launched, All Body Configuration, Rocket/SJ

⑤ B233, M = 8.0-12.0, Air Launched, All Body Configuration, Rocket

⑥ B257, M = 10.0-12.0, HTO Launched, Blended Wing Body Configuration, Turbojet/CSJ

⑦ B260, M = 8.0-12.0, HTO Launched, All Body Configuration, Rocket

⑧ B284, M = 12.0, Air Launched, Unmanned, All Body Configuration, Rocket

MCDONNELL AIRCRAFT

FOLDOUT FRAME

(U) FIGURE 4-179
FLIGHT RESEARCH VEHICLE REFURBISHMENT COST SUMMARY
(1970 Economics)

FOLDOUT FRAME 2

Configuration No.	Configuration Description	Vehicle Investment Cost (\$)	Vehicle Refurbishment Cost (\$)			Refurbishment % of Flight Vehicle Invest. Cost Used	No. of Flights	Refurbishment Cost Total (\$)		
			Labor	Material	Total			Labor	Material	Total
B207A ①	Base Configuration, Hot Inlet	26,648,000	399,720	266,480	666,200	2.5	200	79,944,000	53,296,000	133,240,000
B207A1-1	M = 6.0, Cool Inlet	25,209,000	378,135	252,090	630,225	2.5	200	75,627,000	50,418,000	126,045,000
B207A1-2	M = 7.0, Cool Inlet	29,532,000	442,980	295,320	738,300	2.5	200	88,596,000	59,064,000	147,660,000
B207A1-3	M = 8.0, Cool Inlet	47,526,000	712,890	475,260	1,188,150	2.5	200	142,578,000	95,052,000	237,630,000
B207A2-1	T/W = 1.25	26,643,000	399,645	266,430	666,075	2.5	200	79,929,000	53,286,000	133,215,000
B207A2-2	T/W = 2.00	26,808,000	402,120	268,080	670,200	2.5	200	80,424,000	53,616,000	134,040,000
B207A6-1	Single Base Operation	30,842,000	462,630	308,420	771,050	2.5	200	92,526,000	61,684,000	154,210,000
B207B	"Off the Shelf" Engine	29,398,000	440,970	293,980	734,950	2.5	200	88,194,000	58,796,000	146,990,000
B212A ②	Base Configuration	36,675,000	550,125	366,750	916,875	2.5	200	110,025,000	73,350,000	183,375,000
B212A3-1	10 Minute Test Time	40,560,000	608,400	405,600	1,014,000	2.5	200	121,680,000	81,120,000	202,800,000
B212A5-1	3.5g Structure/3.5g Shingle	36,592,000	548,880	365,920	914,800	2.5	200	109,776,000	73,184,000	182,960,000
B212A5-2	5.0g Structure/5.0g Shingle	40,085,000	601,275	400,850	1,002,125	2.5	200	120,255,000	80,170,000	200,425,000
B212A5-3	7.3g Structure/3.5g Shingle	40,497,000	607,455	404,970	1,012,425	2.5	200	121,491,000	80,994,000	202,485,000
B212A5-4	7.3g Structure/5.0g Shingle	41,986,000	629,790	419,860	1,049,650	2.5	200	125,958,000	83,972,000	209,930,000
B212A7-1	Hot Structure	41,170,000	617,550	411,700	1,029,250	2.5	200	123,510,000	82,340,000	205,850,000
B212A8-1	Non-Integral Tank	37,507,000	562,605	375,070	937,675	2.5	200	112,521,000	75,014,000	187,535,000
B212A9-1	JP Fuel	36,692,000	550,380	366,920	917,300	2.5	200	110,076,000	73,384,000	183,460,000
B232A ③	Base Configuration	29,792,000	446,880	297,920	744,800	2.5	200	89,376,000	59,584,000	148,960,000
B232B	"Off the Shelf" Engine	33,853,000	507,795	338,530	846,325	2.5	200	101,559,000	67,706,000	169,265,000
B233A ④	Base Configuration	28,607,000	429,105	286,070	715,175	2.5	200	85,821,000	57,214,000	143,035,000
B233A1-1	M = 8.0	26,992,000	404,880	269,920	674,800	2.5	200	80,976,000	53,984,000	134,960,000
B233A1-2	M = 10.0	27,506,000	412,590	275,060	687,650	2.5	200	82,518,000	55,012,000	137,530,000
B233A2-1	T/W = 1.25	27,619,000	414,285	276,190	690,475	2.5	200	82,857,000	55,238,000	138,095,000
B233A2-2	T/W = 2.0	28,941,000	434,115	289,410	723,525	2.5	200	86,823,000	57,882,000	144,705,000
B233A3-1	10 Minute Cruise Time	33,524,000	502,860	335,240	838,100	2.5	200	100,572,000	67,048,000	167,620,000
B233A3-2	15 Minute Cruise Time	41,820,000	627,300	418,200	1,045,500	2.5	200	125,400,000	83,600,000	209,000,000
B233A4-1	600 Lb Payload	28,034,000	420,510	280,340	700,850	2.5	200	84,102,000	56,068,000	140,170,000
B233A4-2	1400 Lb Payload	29,181,000	437,715	291,810	729,525	2.5	200	87,543,000	58,362,000	145,905,000
B233A4-3	Payload $\rho = 25$ Lb/Cu.Ft. (400 kg/m ³)	28,582,000	428,730	285,820	714,550	2.5	200	85,746,000	57,164,000	142,910,000
B233A4-4	Payload $\rho = 30$ Lb/Cu.Ft. (481 kg/m ³)	28,545,000	428,175	285,450	713,625	2.5	200	85,635,000	57,090,000	142,725,000
B233A5-1	3.5g Structure/3.5g Shingle	28,101,000	421,515	281,010	702,525	2.5	200	84,303,000	56,202,000	140,505,000
B233A5-2	5.0g Structure/5.0g Shingle	31,131,000	466,965	311,310	778,275	2.5	200	93,393,000	62,262,000	155,655,000
B233A5-3	7.3g Structure/3.5g Shingle	29,274,000	439,110	292,740	731,850	2.5	200	87,822,000	58,548,000	146,370,000
B233A5-4	7.3g Structure/5.0g Shingle	31,810,000	477,150	318,100	795,250	2.5	200	95,430,000	63,620,000	159,050,000
B233A6-1	Single Base Operation	48,608,000	729,120	486,080	1,215,200	2.5	200	145,824,000	97,216,000	243,040,000
B233A7-1	Active Cooling System	26,720,000	400,000	267,200	668,000	2.5	200	80,160,000	53,440,000	133,600,000
B233A7-2	Passive Insulation (Minimum Weight)	29,754,000	446,310	297,540	743,850	2.5	200	89,262,000	59,508,000	148,770,000
B233A9-1	NBP Fuel	29,360,000	440,400	293,600	734,000	2.5	200	88,080,000	58,720,000	146,800,000
B233A11-1	Powered Landing 5000 Lb (2268 kg) Fuel	31,491,000	472,365	314,910	787,275	2.5	200	94,473,000	62,982,000	157,455,000
B233A11-2	Powered Landing 10,000 Lb (4536 kg) Fuel	35,278,000	529,170	352,780	881,950	2.5	200	105,834,000	70,556,000	176,390,000
B233B	5 RL-10 Engines, T/W = 1.117	33,616,000	504,240	336,160	840,400	2.5	200	100,848,000	67,232,000	168,080,000
B233C-1	6 RL-10 Engines, T/W = 1.375	33,656,000	504,840	336,560	841,400	2.5	200	100,968,000	67,312,000	168,280,000

MCDONNELL AIRCRAFT

FOLDOUT FRAME 1

FOLDOUT FRAME 2

(U) FIGURE 4-179 (Continued)
FLIGHT RESEARCH VEHICLE REFURBISHMENT COST SUMMARY
(1970 Economics)

Configuration No.	Configuration Description	Vehicle Investment Cost (\$)	Vehicle Refurbishment Cost (\$)			Refurbishment % of Flight Vehicle Invest. Cost Used	No. of Flights	Refurbishment Cost Total (\$)		
			Labor	Material	Total			Labor	Material	Total
B233C-2	7 RL-10 Engines, T W = 1.550	34,662,000	519,930	346,620	866,550	2.5	200	103,986,000	69,324,000	173,310,000
B233D	75 ⁰ Leading Edge Sweep	31,570,000	473,550	315,700	789,250	2.5	200	94,710,000	63,140,000	157,850,000
B233E	77 ⁰ Leading Edge Sweep	28,955,000	434,325	289,550	723,875	2.5	200	86,865,000	57,910,000	144,775,000
B257F (5)	Base Configuration	69,102,000	1,036,500	691,020	1,727,550	2.5	200	207,306,000	138,204,000	345,510,000
B257F1-2	M = 10.0	52,691,000	790,365	526,910	1,317,275	2.5	200	158,073,000	105,382,000	263,455,000
B257F2-1	T W = 1.17	69,788,000	1,046,820	697,880	1,744,700	2.5	200	209,364,000	139,576,000	348,940,000
B257F3-1	10 Minute Cruise Time	80,074,000	1,201,110	800,740	2,001,850	2.5	200	240,222,000	160,148,000	400,370,000
B257F3-2	7.5 Minute Cruise Time	74,083,000	1,111,245	740,830	1,852,075	2.5	200	222,249,000	148,166,000	370,415,000
B257F5-1	3.5g Structure 3.5g Shingle	64,267,000	964,005	642,670	1,606,675	2.5	200	192,801,000	128,534,000	321,335,000
B257F5-2	5.0g Structure 5.0g Shingle	74,730,000	1,097,490	731,660	1,868,250	2.5	200	224,190,000	149,460,000	373,650,000
B257F5-3	7.3g Structure 3.5g Shingle	72,964,000	1,094,460	729,640	1,824,100	2.5	200	218,892,000	145,928,000	364,820,000
B257F5-4	7.3g Structure 5.0g Shingle	79,518,000	1,192,770	795,180	1,987,950	2.5	200	238,554,000	159,036,000	397,590,000
B257F9-1	JP LH2	58,343,000	875,145	583,430	1,458,575	2.5	200	175,029,000	116,606,000	291,715,000
B257F9-2	NBP LH2	71,930,000	1,078,950	719,550	1,798,250	2.5	200	215,790,000	143,860,000	359,650,000
B257F9-3	Slush LH2	61,873,000	928,095	618,730	1,546,825	2.5	200	185,619,000	123,746,000	309,365,000
B260A (6)	HTO Base Configuration	40,456,000	606,840	404,560	1,011,400	2.5	200	121,368,000	80,912,000	202,280,000
B260A1-1	M = 10.0	30,133,000	451,995	301,330	753,325	2.5	200	90,399,000	60,266,000	150,665,000
B260A1-2	M = 8.0	26,517,000	399,255	266,170	665,425	2.5	200	79,851,000	53,234,000	133,085,000
B260A6-1	Single Base Operation	61,904,000	928,560	619,040	1,547,600	2.5	200	185,712,000	123,808,000	309,520,000
B260B	"Off the Shelf" Engine	44,093,000	661,395	440,930	1,102,325	2.5	200	132,279,000	88,186,000	220,465,000
B284A (7)	Base Configuration	28,193,000	465,185	310,123	775,308	2.75	200	93,036,960	62,024,640	155,061,600
B284B	"Off the Shelf" Engine	33,445,000	551,843	367,895	919,738	2.75	200	110,368,560	73,579,040	183,947,600
B-207	M = 6, Air Launched, All Body Configuration, Rocket Ramjet	32,366,000	485,490	323,660	809,150	2.5	200	97,098,000	64,732,000	161,830,000
B-212	M = 6, HTO Launched, Wing Body Configuration, Turbojet Ramjet	47,650,000	714,750	476,500	1,191,250	2.5	200	142,950,000	95,300,000	238,250,000
B-232	M = 12, Air Launched, All Body Configuration, Rocket Scramjet	33,029,000	495,435	330,290	825,725	2.5	200	99,087,000	66,058,000	165,145,000
B-233	M = 12, Air Launched, All Body Configuration, All Rocket	30,632,000	459,480	306,320	765,000	2.5	200	91,896,000	61,264,000	153,160,000
B-257	M = 12, HTO Launched, Blended Wing Body Configuration, Turbojet CSJ	55,389,000	830,835	553,890	1,384,725	2.5	200	166,167,000	110,778,000	276,945,000
B-260	M = 12, HTO Launched, All Body Configuration, J2S Rocket	43,977,000	659,655	439,770	1,099,425	2.5	200	131,931,000	87,954,000	219,885,000
B-284	M = 12, Air Launched, Unmanned, All Body Configuration, All Rocket	30,779,000	507,854	338,569	846,423	2.75	200	101,570,760	67,713,840	169,284,600

- (1) B-207, M = 6.0-8.0, Air Launched, All Body Configuration, Rocket/Ramjet
(2) B-212, M = 6.0, HTO Launched, Wing Body Configuration, Turbojet/Ramjet
(3) B-232, M = 12.0, Air Launched, All Body Configuration, Rocket/SJ
(4) B-233, M = 8.0-12.0, Air Launched, All Body Configuration, Rocket

- (5) B-257, M = 10.0-12.0, HTO Launched, Blended Wing Body Configuration, Turbojet/CSJ
(6) B-260, M = 8.0-12.0, V/HTO Launched, All Body Configuration, Rocket
(7) B-284, M = 12.0, Air Launched, Unmanned, All Body Configuration, Rocket

MCDONNELL AIRCRAFT

FOLDOUT FRAME 1

(U) FIGURE 4-180
PROPELLANT COST SUMMARY

FOLDOUT FRAME 2

Configuration No.	Configuration Description	Propellant Requirements (Adj.) (1)				No. of Trips	Total Propellant Requirements				Propellant Costs		Total Propellant Cost (\$) (2)		
		Fuel		Oxidizer			Fuel		Oxidizer		(\$/Lb)		Fuel	Oxidizer	Total
		Lb. LH ₂	Kgs	Lb LOX	Kgs		Lb	Kgs	Lb	Kgs	Fuel	Oxidizer			
B207A (3)	Base Configuration Hot Inlet	4,200	1,905	19,155	8,689	200	840,000	381,000	3,831,000	1,737,800	0.1385	0.0061	116,340	23,369	139,709
B207A1-1	M = 6.0, Cool Inlet	3,930	1,783	18,159	8,237	200	786,000	356,600	3,631,800	1,647,400	0.1385	0.0061	108,861	22,154	131,015
B207A1-2	M = 7.0, Cool Inlet	6,009	2,726	26,280	11,920	200	1,201,800	545,200	5,256,000	2,384,000	0.1385	0.0061	166,449	32,062	198,511
B207A1-3	M = 8.0, Cool Inlet	20,670	9,376	64,875	29,427	200	4,134,000	1,875,200	12,975,000	5,885,400	0.1385	0.0061	572,559	79,148	651,707
B207A2-1	T/W = 1.25	4,290	1,946	19,950	9,049	200	858,000	389,200	3,990,000	1,809,800	0.1385	0.0061	118,833	24,339	143,172
B207A2-2	T/W = 2.00	4,148	1,882	18,990	8,614	200	829,600	376,400	3,798,000	1,722,800	0.1385	0.0061	114,900	23,168	138,068
B207A6-1	Single Base Operation	8,019	3,637	25,329	11,489	200	1,603,800	727,400	5,065,800	2,297,800	0.1385	0.0061	222,126	30,901	253,027
B207B	"Off the Shelf" Engine	4,695	2,130	23,373	10,602	200	939,000	426,000	4,674,600	2,120,400	0.1385	0.0061	130,052	28,515	158,567
B212A (4)	Base Configuration	7,024	3,186	-	-	200	1,404,800	637,200	-	-	0.1385	0.0061	194,565	-	194,565
B212A3-1	10 Minute Test Time	10,005	4,538	-	-	200	2,001,000	907,600	-	-	0.1385	0.0061	277,139	-	277,139
B212A5-1	3.5g Structure 3.5g Shingle	6,501	2,949	-	-	200	1,300,200	589,800	-	-	0.1385	0.0061	180,078	-	180,078
B212A5-2	5.0g Structure 5.0g Shingle	7,514	3,408	-	-	200	1,502,800	681,600	-	-	0.1385	0.0061	208,138	-	208,138
B212A5-3	7.3g Structure 3.5g Shingle	7,550	3,425	-	-	200	1,510,000	685,000	-	-	0.1385	0.0061	209,135	-	209,135
B212A5-4	7.3g Structure 5.0g Shingle	8,250	3,742	-	-	200	1,650,000	748,400	-	-	0.1385	0.0061	228,525	-	228,525
B212A7-1	Hot Structure	7,340	3,329	-	-	200	1,468,000	665,800	-	-	0.1385	0.0061	203,318	-	203,318
B212A8-1	Non-Integral Tank	7,515	3,409	-	-	200	1,503,000	681,800	-	-	0.1385	0.0061	208,166	-	208,166
B212A9-1	JP Fuel	5,877	2,666	-	-	200	1,175,400	533,200	-	-	0.1385	-	162,793	-	162,793
		2,580JP	1,170	-	-	200	516,000	234,000	-	-	0.015	-	7,740	-	170,533
B232A (5)	Base Configuration	12,750	5,783	64,170	29,107	200	2,550,000	1,156,600	12,834,000	5,821,400	0.1385	0.0061	353,175	78,287	431,462
B232A	"Off the Shelf" Engine	19,244	8,729	83,625	37,932	200	3,848,800	1,745,800	16,725,000	7,586,400	0.1385	0.0061	533,059	102,023	635,082
B233A (6)	Base Configuration	11,385	5,164	68,640	31,135	200	2,277,000	1,032,800	13,728,000	6,227,000	0.1385	0.0061	315,365	83,741	399,106
B233A1-1	M = 8.0	4,950	2,245	29,565	13,410	200	990,000	449,000	5,913,000	2,682,000	0.1385	0.0061	137,115	36,069	173,184
B233A1-2	M = 10.0	7,485	3,395	44,745	20,296	200	1,497,000	679,000	8,949,000	4,059,200	0.1385	0.0061	207,335	54,589	261,924
B233A2-1	T/W = 1.25	11,730	5,321	70,170	31,829	200	2,346,000	1,064,200	14,034,000	6,365,800	0.1385	0.0061	324,921	85,607	410,528
B233A2-2	T/W = 2.00	11,580	5,253	69,420	31,488	200	2,316,000	1,050,600	13,884,000	6,297,600	0.1385	0.0061	320,766	84,692	405,458
B233A3-1	10 Minute Cruise Time	17,850	8,097	106,785	48,437	200	3,570,000	1,619,400	21,357,000	9,687,400	0.1385	0.0061	494,445	130,278	624,723
B233A3-2	15 Minute Cruise Time	30,750	13,948	177,780	80,640	200	6,150,000	2,789,600	35,556,000	16,128,000	0.1385	0.0061	851,775	216,892	1,068,667
B233A4-1	600 Lb Payload	11,055	5,014	66,603	30,211	200	2,211,000	1,002,800	13,320,600	6,042,200	0.1385	0.0061	306,224	81,256	387,480
B233A4-2	1400 Lb Payload	11,745	5,327	70,664	32,053	200	2,349,000	1,065,400	14,132,800	6,410,600	0.1385	0.0061	325,337	86,210	411,547
B233A4-3	Payload p = 25 Lb/Cu Ft	11,370	5,157	68,460	31,053	200	2,274,000	1,031,400	13,692,000	6,210,600	0.1385	0.0061	314,949	83,521	398,470
B233A4-4	Payload p = 30 Lb/Cu Ft	11,400	5,171	68,235	30,951	200	2,280,000	1,034,200	13,647,000	6,190,200	0.1385	0.0061	315,780	83,247	399,027
B233A5-1	3.5g Structure 3.5g Shingle	11,280	5,117	67,635	30,679	200	2,256,000	1,023,400	13,527,000	6,135,800	0.1385	0.0061	312,456	82,515	394,971
B233A5-2	5.0g Structure 5.0g Shingle	12,150	5,511	72,675	32,965	200	2,430,000	1,102,200	14,535,000	6,593,000	0.1385	0.0061	336,555	88,664	425,219
B233A5-3	7.3g Structure 3.5g Shingle	11,775	5,341	70,950	32,182	200	2,355,000	1,068,200	14,190,000	6,436,400	0.1385	0.0061	326,168	86,559	412,727
B233A5-4	7.3g Structure 5.0g Shingle	12,525	5,681	75,075	34,053	200	2,505,000	1,136,200	15,015,000	6,810,600	0.1385	0.0061	346,943	91,592	438,535
B233A6-1	Single Base Operation	37,650	17,078	226,245	102,623	200	7,530,000	3,415,600	45,249,000	20,524,600	0.1385	0.0061	1,042,905	276,019	1,318,924
B233A7-1	Active Cooling System	9,930	4,504	59,520	26,998	200	1,986,000	900,800	11,904,000	5,399,600	0.1385	0.0061	275,061	72,614	347,675
B233A7-2	Passive Insulation (Min Weight)	11,490	5,212	68,970	31,284	200	2,298,000	1,042,000	13,794,000	6,256,800	0.1385	0.0061	318,273	84,143	402,416
B233A9-1	NBP Fuel	11,850	5,375	71,267	32,326	200	2,370,000	1,075,000	14,253,400	6,465,200	0.1385	0.0061	328,245	86,946	415,191
B233A11-1	Power Lg 5000 Lb Fuel	16,695	7,573	100,155	45,430	200	3,339,000	1,514,600	20,031,000	9,086,000	0.1385	0.0061	262,452	122,189	384,641
B233A11-2	Power Lg 10,000 Lb Fuel	21,855	9,913	131,145	59,486	200	4,371,000	1,982,600	26,229,000	11,897,200	0.1385	0.0061	605,384	159,997	765,381
B233B	5 RL-10 Engines, T/W = 1.117	18,375	8,335	91,500	41,504	200	3,675,000	1,667,000	18,300,000	8,300,800	0.1385	0.0061	508,988	111,630	620,618
B233C-1	6 RL-10 Engines, T/W = 1.375	17,700	8,029	88,650	40,211	200	3,540,000	1,605,000	17,730,000	8,042,200	0.1385	0.0061	490,290	108,153	598,443
B233C-2	7 RL-10 Engines, T/W = 1.550	18,300	8,301	91,350	41,437	200	3,660,000	1,660,200	18,270,000	8,287,200	0.1385	0.0061	506,910	111,447	618,357
B233D	75° Leading Edge Sweep	13,500	6,123	81,000	36,741	200	2,700,000	1,224,500	16,200,000	7,348,200	0.1385	0.0061	373,950	98,820	472,770
B233E	77° Leading Edge Sweep	12,480	5,661	75,045	34,040	200	2,496,000	1,132,200	15,009,000	6,808,000	0.1385	0.0061	345,696	91,555	437,251

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FOLDOUT FRAME 1

(U) FIGURE 4-180 (Continued)
PROPELLANT COST SUMMARY

FOLDOUT FRAME 2

Configuration No.	Configuration Description	Propellant Requirements (Adj.) (1)				No. of Trips	Total Propellant Requirements				Propellant Costs (\$/Lb)		Total Propellant Cost (\$) (2)		
		Fuel		Oxidizer			Fuel		Oxidizer		Fuel	Oxidizer	Fuel	Oxidizer	Total
		Lb LH ₂	Kgs	Lb LOX	Kgs		Lb	Kgs	Lb	Kgs					
B257F (7)	Base Configuration	52,845	23,970	-	-	200	10,569,000	4,794,000	-	-	0.1385	0.0061	1,463,807	-	1,463,807
B257F1-2	M = 10.0	25,800	11,703	-	-	200	5,160,000	2,340,600	-	-	0.1386	0.0061	714,660	-	714,660
B257F2-1	T W = 1.17	48,371	21,941	-	-	200	9,674,200	4,388,200	-	-	0.1385	0.0061	1,339,877	-	1,339,877
B257F3-1	10 Minute Cruise Time	67,803	30,755	-	-	200	13,560,600	6,151,000	-	-	0.1385	0.0061	1,878,143	-	1,878,143
B257F3-2	7.5 Minute Cruise Time	57,597	26,126	-	-	200	11,519,400	5,225,200	-	-	0.1385	0.0061	1,595,437	-	1,595,437
B257F5-1	3.5g Structure 3.5g Shingle	44,190	20,044	-	-	200	8,838,000	4,008,800	-	-	0.1385	0.0061	1,224,063	-	1,224,063
B257F5-2	5.0g Structure 5.0g Shingle	52,425	23,780	-	-	200	10,485,000	4,756,000	-	-	0.1385	0.0061	1,452,173	-	1,452,173
B257F5-3	7.3g Structure 3.5g Shingle	50,550	22,929	-	-	200	10,110,000	4,585,800	-	-	0.1385	0.0061	1,400,235	-	1,400,235
B257F5-4	7.3g Structure 5.0g Shingle	55,163	25,022	-	-	200	11,032,600	5,004,400	-	-	0.1385	0.0061	1,528,015	-	1,528,015
B257F9-1	JP LH ₂	29,850	13,540	-	-	200	5,970,000	2,708,000	-	-	0.1385	-	826,845	-	826,845
B257F9-2	NBP LH ₂	18,711 JP	8,487	-	-	200	3,742,200	1,697,400	-	-	0.015	-	56,133	-	56,133
B257F9-2	NBP LH ₂	50,123	22,735	-	-	200	10,024,600	4,547,000	-	-	0.1385	0.0061	1,388,407	-	1,388,407
B257F9-3	Slush LH ₂	40,466	18,355	-	-	200	8,093,200	3,671,000	-	-	0.1385	0.0061	1,120,908	-	1,120,908
B260A (8)	HTO Base Configuration	22,999	10,432	138,023	62,606	200	4,599,800	2,086,400	27,604,600	12,521,200	0.1385	0.0061	637,072	168,388	805,460
B260A1-1	M = 10.0	14,175	6,430	84,975	38,544	200	2,835,000	1,286,000	16,995,000	7,708,800	0.1385	0.0061	392,648	108,670	496,318
B260A1-2	M = 8.0	9,488	4,304	56,888	25,804	200	1,897,600	860,800	11,377,600	5,160,800	0.1385	0.0061	262,818	69,403	332,221
B260A6-1	Single Base Operation	64,050	29,053	384,450	174,384	200	12,810,000	5,810,600	76,890,000	34,876,800	0.1385	0.0061	1,774,185	469,029	2,243,214
B260B	"Off the Shelf" Engine	33,000	14,969	183,075	83,041	200	6,600,000	2,993,710	36,615,000	16,608,285	0.1385	0.0061	914,100	223,352	1,137,452
B284A (9)	Base Configuration	11,025	5,000	66,075	29,971	200	2,205,000	1,000,000	13,215,000	5,994,200	0.1385	0.0061	305,393	80,612	386,005
B284B	"Off the Shelf" Engine	14,955	6,783	89,595	40,640	200	2,991,000	1,356,600	17,919,000	8,128,000	0.1385	0.0061	414,254	109,306	523,560
B207	M = 6.0, Air Launched All Body Configuration, Rocket Ramjet	5,085	2,307	24,915	11,301	200	1,017,000	461,303	4,983,000	2,260,251	0.1385	0.0061	140,854	30,396	171,250
B212	M = 6.0, HTO Launched, Wing Body Configuration, Turbojet Ramjet	6,000	2,722	-	-	200	1,200,000	544,311	-	-	0.1385	-	166,200	-	166,200
B232	M = 12, Air Launched, All Body Configuration, Rocket Scramjet	6,457 JP	2,929	-	-	200	1,291,400	585,769	-	-	0.015	-	19,365	-	19,365
B233	M = 12, Air Launched, All Body Configuration, All Rocket	16,800	7,620	71,625	32,489	200	3,360,000	1,524,000	14,325,000	6,497,800	0.1385	0.0061	465,360	87,383	552,743
B257	M = 12, HTO Launched, Blended Wing Body Configuration, Turbojet CSJ	15,330	6,594	76,680	34,781	200	3,066,000	1,390,800	15,336,000	6,956,200	0.1385	0.0061	424,641	93,550	518,191
B260	M = 12, HTO Launched, All Body Configuration, J2S Rocket	27,900	12,655	-	-	200	5,580,000	2,531,045	-	-	0.1385	-	772,830	-	772,830
B260	M = 12, HTO Launched, All Body Configuration, J2S Rocket	17,490 JP	7,933	-	-	200	3,498,000	1,586,666	-	-	0.15	-	52,470	-	52,470
B284	M = 12, Air Launched, Unmanned, All Body Configuration, All Rocket	29,025	13,166	159,675	7,243	200	5,805,000	2,633,200	31,935,000	1,448,600	0.1385	0.0061	803,993	194,804	998,797
B284	M = 12, Air Launched, Unmanned, All Body Configuration, All Rocket	14,550	6,600	72,825	33,033	200	2,910,000	1,320,000	14,565,000	6,606,600	0.1385	0.0061	403,035	88,847	491,882

(1) The following utilization factors were used: LH₂ - 1.5, LOX - 1.5 and JP - 1.1.

(2) Millions of 1970 Dollars

(3) B-207, M = 6.0-8.0, Air Launched, All Body Configuration, Rocket Ramjet.

(4) B-212, M = 6.0, HTO Launched, Wing Body Configuration, Turbojet Ramjet.

(5) B-232, M = 12.0, Air Launched, All Body Configuration, Rocket SJ.

(6) B-233, M = 8.0-12.0, Air Launched, All Body Configuration, Rocket.

(7) B-257, M = 10.0-12.0, HTO Launched, Blended Wing Body Configuration, Turbojet CSJ.

(8) B-260, M = 8.0-12.0, V/HTO Launched, All Body Configuration, Rocket.

(9) B-284, M = 12.0, Air Launched, Unmanned, All Body Configuration, Rocket.

FOLDOUT FRAME 1

(U) FIGURE 4-182

FOLDOUT FRAME 2

"NEAR OPTIMUM" VEHICLE SELECTION RATIONALE SUMMARY VEHICLE PARAMETERS

Configuration Number and Description		Mach Number	Accelerator Engine	T/W	Test Time (Min.)	Payload (Lb m/kg)	Load Factor	Basing Concept	T.P.S.	Fuel Tankage	Fuel Type	W/S lb/ft ² /kg/m ²	Fineness Ratio	Landing Edge Sweep	Landing Power
B207 Manned, Air Launched All Body, Rkt RJ	Parameter	6	(3) RL10-A-3-B Rockets	1.48	10	1500/680.4	5g Struct. 3.5g T.P.S.	Dual	Active	Integral	LH ₂ (Subcooled)	77.4/377.90	8.0	80°	None
	Rationale	Mach No. Trade Study Sect. 4.3.4.1 & 4.3.4.4 Best research value for cost	Rocket Trade Study Sect. 4.4.1.3d Off-shell rockets reduce program cost.	T/W Trade Study Sect. 4.4.1.2 Fallout with Least No. Engines for Mission	Test Time Trade Study Sect. 4.3.4.2 Improved Research Value for Cost	Payload Trade Study Sect. 4.3.4.3 Improved Research Value for Cost	Structural Trade Study Sect. 4.4.3.1 Research Aircraft Safety. Cost best at 5g	Basing Trade Study Sect. 4.4.5.1 Single Base Possible but Presents Operational Problems	T.P.S. Trade Study Sect. 4.3.3.3 Lower Weight and Cost	Tankage Trade Study Sect. 4.4.3.2 Lower Weight and Cost	Propellant Trade Study Sect. 4.4.3 Lower Weight and Cost	Design and Performance Fallout	Shape Fallout	Trade Study Sect. Lower Weight and Cost	Landing Assist Study Sect. Too Costly
B212 Manned, Horizontal Takeoff, Wing Body, TJ RJ	Parameter	6	(2) F100-GE-100 Turbojets	0.99	10	1500 680.4	5g Struct. 3.5g T.P.S.	Dual	Active	Integral	LH ₂ (Subcooled) JP (Room Temp)	46.8/228.5	8.5	75°	1 Go-Around
	Rationale	Mach No. Trade Study Sect. 4.3.4.1 & 4.3.4.4 Best research value for cost	No Applicable Trade Study Selected to reduce program costs.	T/W Trade Study Sect. 4.4.1.2 Fallout with Least No. Engines for Mission	Test Time Trade Study Sect. 4.3.4.2 Improved Research value for cost	Payload Trade Study Sect. 4.3.4.3 Improved Research Value for Cost	Structural Trade Study Sect. 4.4.3.1 Research Aircraft Safety. Cost best at 5g	Basing Trade Study Sect. 4.4.5.1 Single Base Capability Increases Vehicle Size and Cost Disproportionately	T.P.S. Trade Study Sect. 4.3.3.3 Lower Weight and Cost	Tankage Trade Study Sect. 4.4.3.2 Lower Weight and Cost	Propellant Trade Study Sect. 4.4.3 Lower Weight and Cost	Design and Performance Fallout	Configuration Fallout	No Applicable Study Established for Specific Vehicle Configuration	Landing Assist Study Sect. 4.4 Fuel required has minimum impact on size & cost. Improved safety.
B232 Manned, Air Launched All Body, Rkt SJ	Parameter	12	(5) RL10-A-3-B Rockets	1.29	5	1500 680.4	5g Struct. 3.5g T.P.S.	Dual	Active	Integral	LH ₂ (Subcooled)	89/434.53	7.1	80°	None
	Rationale	Mach No. Trade Study Sect. 4.3.4.1 and 4.3.4.4 Best research value for cost	Rocket Trade Study Sect. 4.4.1.3d Off-shell rockets reduce program cost	T/W Trade Study Sect. 4.4.1.2 Fallout with Least No. Engines for Mission	Test Time Trade Study Sect. 4.3.4.2 Increased over minimum time not cost effective	Payload Trade Study Sect. 4.3.4.3 Improved Research Value for Cost	Structural Trade Study Sect. 4.4.3.1 Research Aircraft Safety. Cost best at 5g	Basing Trade Study Sect. 4.4.5.1 Pressure suit time most constraining factor on single base.	T.P.S. Trade Study Sect. 4.3.3.3 Lower Weight and Cost	Tankage Trade Study Sect. 4.4.3.2 Lower Weight and Cost	Propellant Trade Study Sect. 4.4.3 Lower Weight and Cost	Design and Performance Fallout	Shape Fallout	Trade Study Sect. Lower Weight and Cost	Landing Assist Study Sect. Too Costly

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FOLDOUT FRAME

FOLDOUT FRAME 2

(U) FIGURE 4-182 (Continued)

"NEAR OPTIMUM" VEHICLE SELECTION RATIONALE SUMMARY VEHICLE PARAMETERS

Configuration Number and Description		Mach Number	Accelerator Engine	T/W	Test Time (Min.)	Payload (Lb m/kg)	Load Factor	Basing Concept	T.P.S.	Fuel Tankage	Fuel Type	W/S lb/ft ² /kg/m ²	Fineness Ratio	Leading Edge Sweep	Landing Power
B233 Manned, Air Launched All Body, All Rocket	Parameter	12	(5) RL10-A-3-8 Rockets	1.30	5	1500/680.4	5g Struct. 3.5g T.P.S.	Dual	Active	Integral	LH ₂ (Subcooled)	86/419.89	7.6	80°	None
	Rationale	Mach No. Trade Study Sect. 4.3.4.1 & 4.3.4.4 Best research value for cost.	Rocket Trade Study Sect. 4.4.1.3d Significant program cost reduction with off-shell rockets for acceleration.	T/W Trade Study Sect. 4.4.1.2 Fallout with Least No. Engines for Mission	Test Time Trade Study Sect. 4.3.4.2 Increased over minimum time not cost effective	Payload Trade Study Sect. 4.3.4.3 Improved Research Value for Cost	Structural Trade Study Sect. 4.4.3.1 Research Aircraft Safety/ Cost best at 5g	Basing Trade Study Sect. 4.4.5.1 Pressure suit time most constraining factor on single base	T.P.S. Trade Study Sect. 4.3.3.3 Lower Weight and Cost	Tankage Trade Study Sect. 4.4.3.2 Lower Weight and Cost	Propellant Trade Study Sect. 4.4.3 Lower Weight and Cost	Design and Performance Fallout	Shape Fallout	Trade Study Sect. 4.4.3 Lower Weight and Cost	Landing Assist Study Sect. Too Costly
B257 Manned, Horizontal Takeoff Blended Wing Body Turbojet/CSJ	Parameter	12	(5) F100-GE-100 Turbojets	1.21	5	1500/680.4	5g Struct. 3.5g T.P.S.	Dual	Active	Integral	LH ₂ (subcooled)	43.4/212.38	7.5	77°	None
	Rationale	Mach No. Trade Study Sect. 4.3.4.1 and 4.3.4.4 Best research value for cost.	No Applicable Trade Study Selected to reduce program costs	T/W Trade Study Sect. 4.3.4.2 Fallout with Least No. Engines for Mission	Test Time Trade Study Sect. 4.3.4.2 Test range requirement excessive at longer cruise time.	Payload Trade Study Sect. 4.3.4.3 Research Value for Cost	Structural Trade Study Sect. 4.4.3.1 Research Aircraft Safety/ Cost best at 5g	Basing Trade Study Sect. 4.4.5.1 Single Base Capability Increases Vehicle Size and Cost Disproportionately	T.P.S. Trade Study Sect. 4.3.3.3 Lower Weight and Cost	Tankage Trade Study Sect. 4.4.3.2 Lower Weight and Cost	Propellant Trade Study Sect. 4.4.3 Lower Weight and Cost	Design and Performance Fallout	Shape Fallout	Trade Study Sect. 4.4.3 Lower Weight and Cost	Landing Assist Study Sect. 4.4 Too Heavy and Costly
B260 Manned, Horizontal/ Vertical Takeoff, All Body/ All Rocket	Parameter	12	(1) J2S Rocket	1.59	5	1500/680.4	5g Struct. 3.5g T.P.S.	Dual	Active	Integral	LH ₂ (Subcooled)	103/502.89	6.9	80°	None
	Rationale	Mach No. Trade Study Sect. 4.3.4.1 and 4.3.4.4 Best research value for cost.	Rocket Trade Study Sect. 4.4.1.3d	T/W Trade Study Sect. 4.4.1.2 Fallout with Least No. Engines for Mission	Test Time Trade Study Sect. 4.3.4.2 Increased over minimum time not cost effective	Payload Trade Study Sect. 4.3.4.3 Improved Research Value for Cost	Structural Trade Study Sect. 4.4.3.1 Research Aircraft Safety/ Cost best at 5g	Basing Trade Study Sect. 4.4.5.1 Single Base Capability Increases Vehicle Size and Cost Disproportionately	T.P.S. Trade Study Sect. 4.3.3.3 Lower Weight and Cost	Tankage Trade Study Sect. 4.4.3.2 Lower Weight and Cost	Propellant Trade Study Sect. 4.4.3 Lower Weight and Cost	Design and Performance Fallout	Shape Fallout	Trade Study Sect. 4.4.3 Lower Weight and Cost	Landing Assist Study Sect. Too Costly
B284 Unmanned, Air Launched All Body, All Rocket	Parameter	12	(5) RL10-A-3-8 Rockets	1.38	5	1500/680.4	5g Struct. 3.5g T.P.S.	Dual	Active	Integral	LH ₂ (Subcooled)	82.5/402.80	7.1	80°	None
	Rationale	Mach No. Trade Study Sect. 4.3.4.1 and 4.3.4.4 Best research value for cost.	Rocket Trade Study Sect. 4.4.1.3d	T/W Trade Study Sect. 4.4.1.2 Fallout with Least No. Engines for Mission	Test Time Trade Study Sect. 4.3.4.2 Increased over minimum time not cost effective	Payload Trade Study Sect. 4.3.4.3 Improved Research Value for Cost	Structural Trade Study Sect. 4.4.3.1 Research Aircraft Safety/ Cost best at 5g	Basing Trade Study Sect. 4.4.5.1 Single base not practical from operational standpoint	T.P.S. Trade Study Sect. 4.3.3.3 Lower Weight and Cost	Tankage Trade Study Sect. 4.4.3.2 Lower Weight and Cost	Propellant Trade Study Sect. 4.4.3 Lower Weight and Cost	Design and Performance Fallout	Shape Fallout	Trade Study Sect. 4.4.3 Lower Weight and Cost	Landing Assist Study Sect. Too Costly

MCDONNELL AIRCRAFT

4.6 "NEAR OPTIMUM" CANDIDATE FLIGHT TEST RESEARCH VEHICLES

(U) Facility research values were presented in Section 3 for the "baseline" facilities spectrum. The tradeoff studies described in Section 4.4 resulted in improved flight vehicle configurations, defined as "near-optimum" systems. Research values are presented in Figure 4-181 for the "near-optimum" facilities and in all cases are increased slightly over the "baseline" facility values presented in Section 3.

(U) Based on the conclusions of Section 4.4, seven "near-optimum" vehicle configurations were evolved which are described in the following paragraphs. All rocket propulsion systems use "off-the-shelf" rockets with subcooled liquid hydrogen (LH₂), and liquid oxygen (LO₂) for propellants. All ramjets and scramjets use subcooled liquid hydrogen fuel. Instrumentation payload was increased to 1500 pounds in all "near-optimum" vehicles. The modest increase in vehicle costs was judged reasonable for an increase in instrumentation capability of 50% value. An increase in test time from five minutes to ten minutes was also justified for the Mach 6 airbreather cruise vehicles (configurations B207 and B212) on the basis of the research value compared to the investment. Although substantially increased research value was exhibited for all flight vehicles as test time was increased to ten minutes, the additional vehicle cost could not be justified for any vehicle except B207 and B212. The maximum Mach number of the "near-optimum" vehicles was not changed from that of the "baseline" vehicles. In all cases, variation of the research value resulted from changes in maximum Mach number. However, the increased weight and cost of the flight vehicle was only found to be effective at the extreme upper end of the Mach range studied.

(U) Dual basing concept was selected for all "near-optimum" vehicles. It was found that the fuel volume required for a single base operation became excessive and created large, costly vehicles for little gain. This study is discussed in Section 4.4.6.1.

(U) Pilot comfort and efficiency is affected by the length of time in a pressure suit. This time has been set as a maximum of four hours in Section 4.4.6 and it is shown that any Mach 12 test flight will cause this time limit to be exceeded if single base operations were achieved.

(U) Powered landings and go-around capabilities create an additional requirement for fuel volume. The ramjet and scramjet engines are inoperable at subsonic speeds and the rocket engines selected are very inefficient at sea level flight conditions. Therefore, the fuel volume for rocket powered vehicles becomes excessive for the powered landing and go-around maneuvers. The one vehicle that does have this capability (B212) uses two JP fueled turbojets for power for these conditions and only a slight increase in vehicle size is required for the one go-around capability. The increase involved is considered acceptable for the added safety.

(U) Figure 4-182 tabulates the effects of the flight vehicle parametric studies and the reasons for selecting the "near-optimum" vehicle configurations.

(U) The following paragraphs present additional discussion for each airplane.

(U) FIGURE 4-181
FACILITY RESEARCH VALUES - "NEAR OPTIMUM" FACILITIES
(Capability of Existing Plus New Facilities)

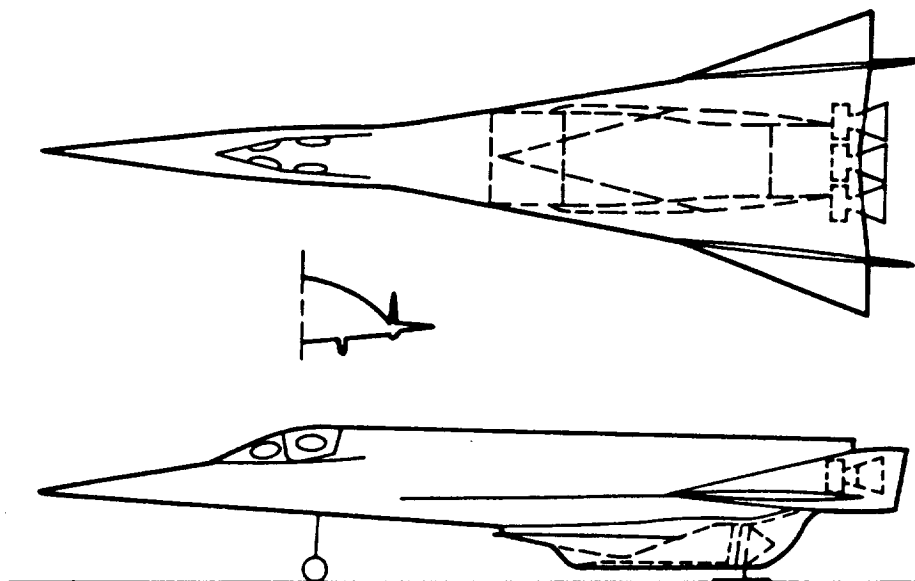
	OPERATIONAL SYSTEM			
	L2	C1	M1	M2
TOTAL	8905.1	8397.7	7947.1	9004.5
EXISTING	3295.0	4425.6	4646.0	3203.3
207	6047.4	7077.3	7087.9	5962.0
212	5990.5	7509.6	7464.3	5798.0
232	6760.4	6792.4	6649.0	7649.9
233	6101.0	6613.5	6610.0	6230.9
257	7674.2	6876.4	6819.6	7050.3
260	6237.5	6712.3	6688.7	6407.7
284	5832.2	6315.1	6310.4	5925.7

PERCENT ACHIEVED OF TOTAL RESEARCH INVOLVED

	OPERATIONAL SYSTEM			
	L2	C1	M1	M2
TOTAL	100.0	100.0	100.0	100.0
EXISTING	37.0	49.7	58.5	35.6
207	67.9	79.5	89.2	66.2
212	67.3	84.4	93.9	64.3
232	75.9	75.3	84.2	85.0
233	65.5	74.3	83.2	69.2
257	85.9	77.3	85.8	78.3
260	70.0	75.4	94.2	71.2
284	65.5	71.0	79.4	65.8

(U) Configuration B207 - This vehicle is a Mach 6, all-body shape using three "off-the-shelf" RL10-A-3-8 rockets for acceleration and a MCAIR two-dimensional ramjet for cruise power. A 1500 pound payload and a 10 minute test time improves the data return and research value of this vehicle. An active thermal protection system and integral fuel tankage result in reduced structure weight. A dual basing concept is required for this vehicle to retain its small size. Go-around and powered landings were deleted for this vehicle. Other selected parameters, weight and cost data are shown in Figure 4-183.

(U) FIGURE 4-183 MACH 6 ALL-BODY
Rocket-Ramjet
Air Launch



CONFIGURATION REFERENCE B207

O.W.E.	30,825 lb
T.O.G.W.	51,725 lb
COST	633.6 Million Dollars

SELECTED PARAMETERS

MACH NO.	6	BASING	DUAL
ENGINE - ACCEL	3 RL10-A-3-8	TPS	ACTIVE
HYP	2d MCAIR RAMJET	TANKAGE	INTEGRAL
THRUST LOADING	1.29	FUEL TYPE	LH ₂ (Subcooled)
TEST TIME	10 Min. @ M = 6	WING LOADING	80.7 psf (393.97 kg/m ²)
PAYLOAD	1500 lb (680.4 kg)	FINENESS RATIO	8.0
LOAD FACTOR	5g (Struct) 3.5(TPS)	L.E. SWEEP	80°
PLANFORM	606 ft ² (56.3 m ²)	LANDING POWER	NONE

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(U) Configuration B212 - This configuration is a Mach 6, wing-body shape with 10 minutes test time and a data package payload of 1500 pounds. The propulsion system consists of two JP fueled turbojets and one LH₂ fueled two-dimensional ramjet.

(U) Two additional configurations, B212B and B212C, were designed using two ramjets and two turbojets with a common inlet to each ramjet-turbojet combination as shown by Figure 4-184. Operation of these inlet systems is shown in Figure 4-185. The drag of these vehicles is significantly higher than the B212 aircraft and is shown in Figure 4-186. Some drag increase is attributed to the large increase in fuselage cross sectional area required to house the engines and inlets. The major increase is due to the large common inlets producing a large external cowl drag.

(U) The "go-around" and powered landing capabilities are acceptable because of the small increase in fuel volume required and it proved a beneficial safety feature for the small increase in vehicle size. The dual basing concept increases this vehicle's potential. Other selected parameters, weights, and cost are presented in Figure 4-187.

(U) A study of vehicle B212A was made to determine the Mach capability of the configuration. Above Mach 6 the acceleration capability is marginal but extends to $M_{max} = 6.7$ @ 103,000 ft. Because of the fixed size of the F-100 engines the vehicle cannot be readily enlarged to accept a significantly larger ramjet engine without becoming divergent due to inadequate transonic acceleration. Therefore, the upper design cruise speed limit of this vehicle is selected as Mach 6.

(U) Configuration B232 - This configuration has a 5 minute test time and a 1500 pound data package payload with an active thermal protection system. The five acceleration engines are "off-the-shelf" RL10-A-3-8 rockets. This manned, Mach 12, air launched, scramjet cruise vehicle requires dual basing operation.

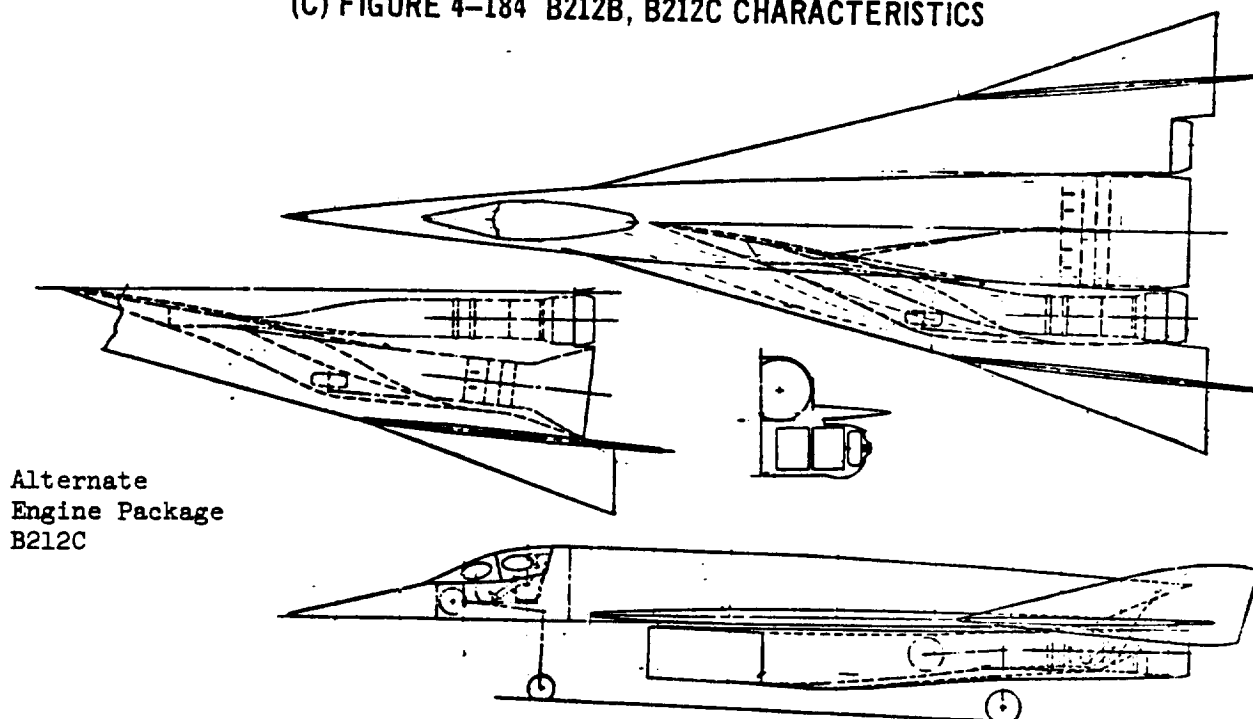
(U) As in the case for all rocket powered air launched vehicles, the addition of "go-around" capability and/or powered landing capability forces the vehicle size and weight beyond the C-5A launch aircraft limits. Other selected parameters, weights, and cost data are shown in Figure 4-188.

(U) Configuration B233 - This configuration is an air launched, Mach 12, manned vehicle using five "off-the-shelf" RL10-A-3-8 rockets for acceleration and cruise. The vehicle is sized for 5 minutes test time at Mach 12. Active thermal protective system and integral tankage reduce the structural weight of the vehicle. Dual basing is required to maintain the small physical size of the vehicle.

(U) The "go-around" capability or powered landing capability is not feasible for the reasons mentioned previously in Paragraph 4.6.3. Other selected parameters, weights and cost data are presented in Figure 4-189.

(U) Configuration B257 - This vehicle, with four F 100-GE-100 JP fueled turbojet acceleration engines and a convertible scramjet cruise engine, provides five minutes of Mach 12 test time. Active thermal protection system, integral tankage, and the dual basing concept are utilized to reduce vehicle size.

(C) FIGURE 4-184 B212B, B212C CHARACTERISTICS

PHYSICAL CHARACTERISTICS

GEOMETRY:

L.E. Sweep	75°
Overall Length ft (m)	75.8 (23.10)
Total Span ft (m)	35. (10.67)
Height ft (m)	132 (4.02)

AREAS:

	ft ²	m ²
Planform	962	(89.37)
Wing	811	(75.34)
Tips, Total	108	(10.03)
Vertical Tail, Total	178	(16.54)

VOLUMES:

	ft ³	m ³
Total	3764	(106.50)
Propellant	1014	(28.70)

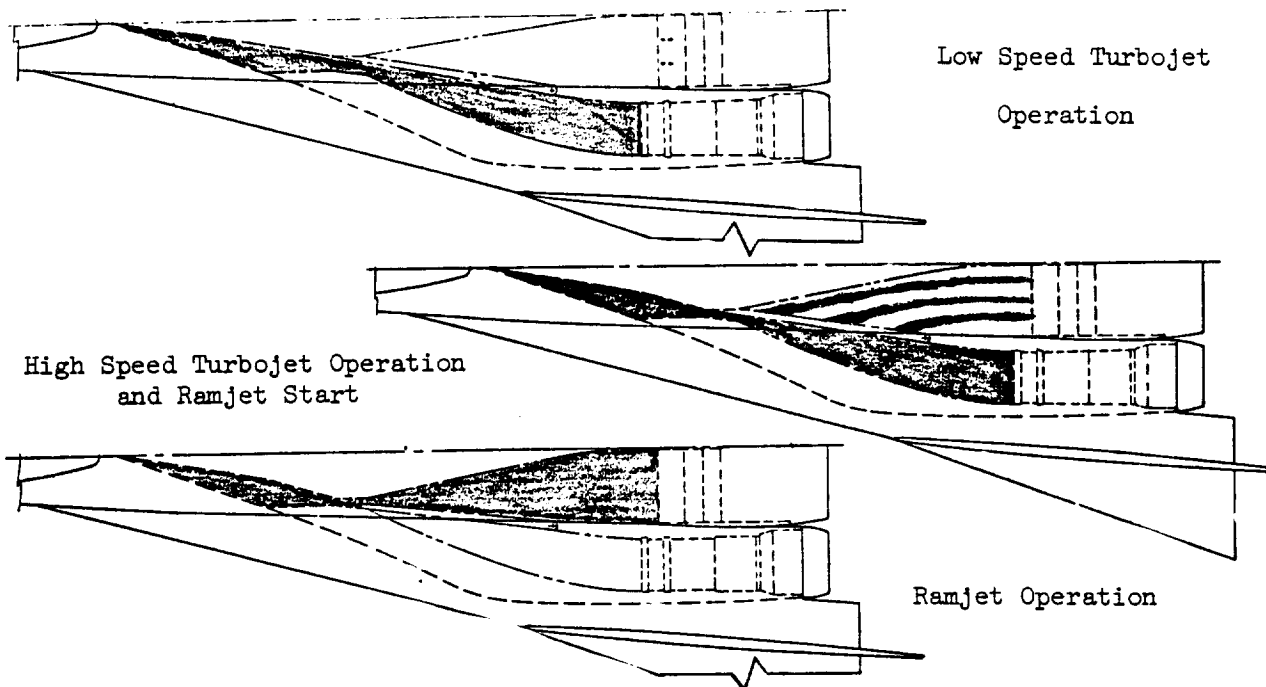
PROPULSION:

	ACCEL.	CRUISE
Engine	(2) F100GE-100	(2) Ramjet
Thrust (SLS, uninst) lb	23,380 (ea)	-
(N)	(103,994) (ea)	-
Thrust (Mach 6 @ lb	-	4665 (ea)
Cruise Alt) (N)	-	(20,750)
A _{capt} Total ft ²	20.4	20.4
(m ²)	(1.90)	(1.90)

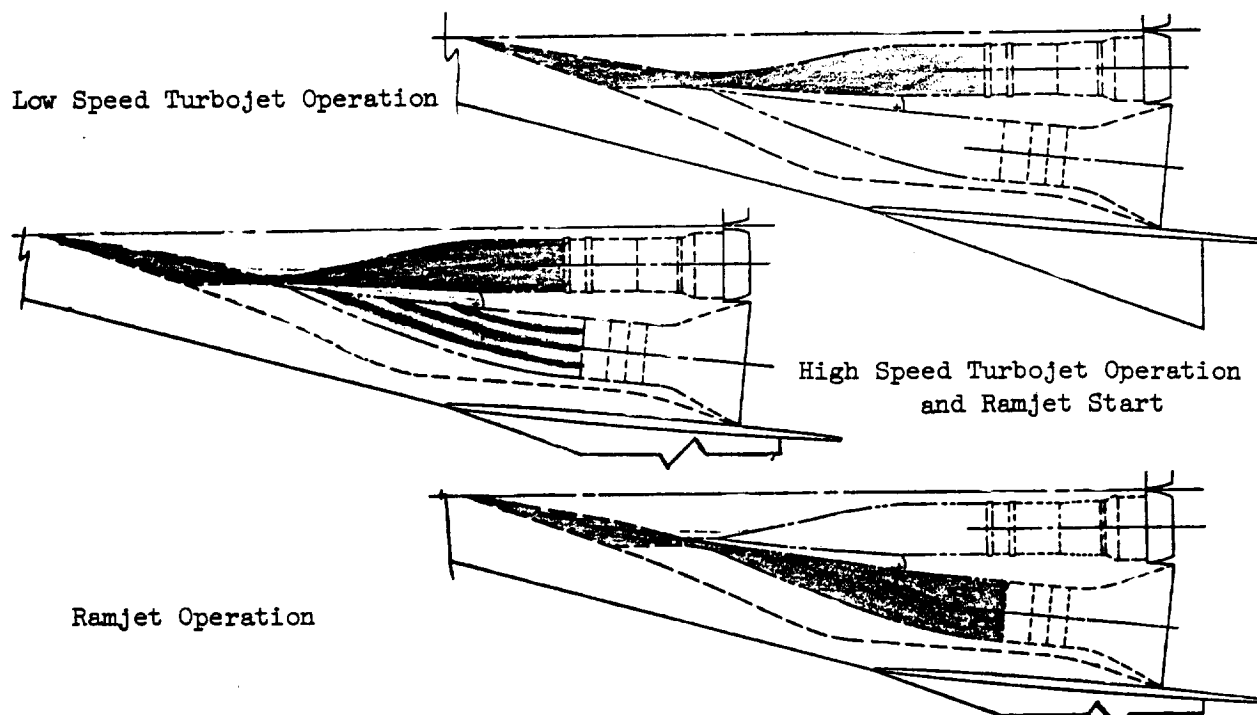
~~CONFIDENTIAL~~

(U) FIGURE 4-185
COMMON INLET OPERATION
B212B AND B212C

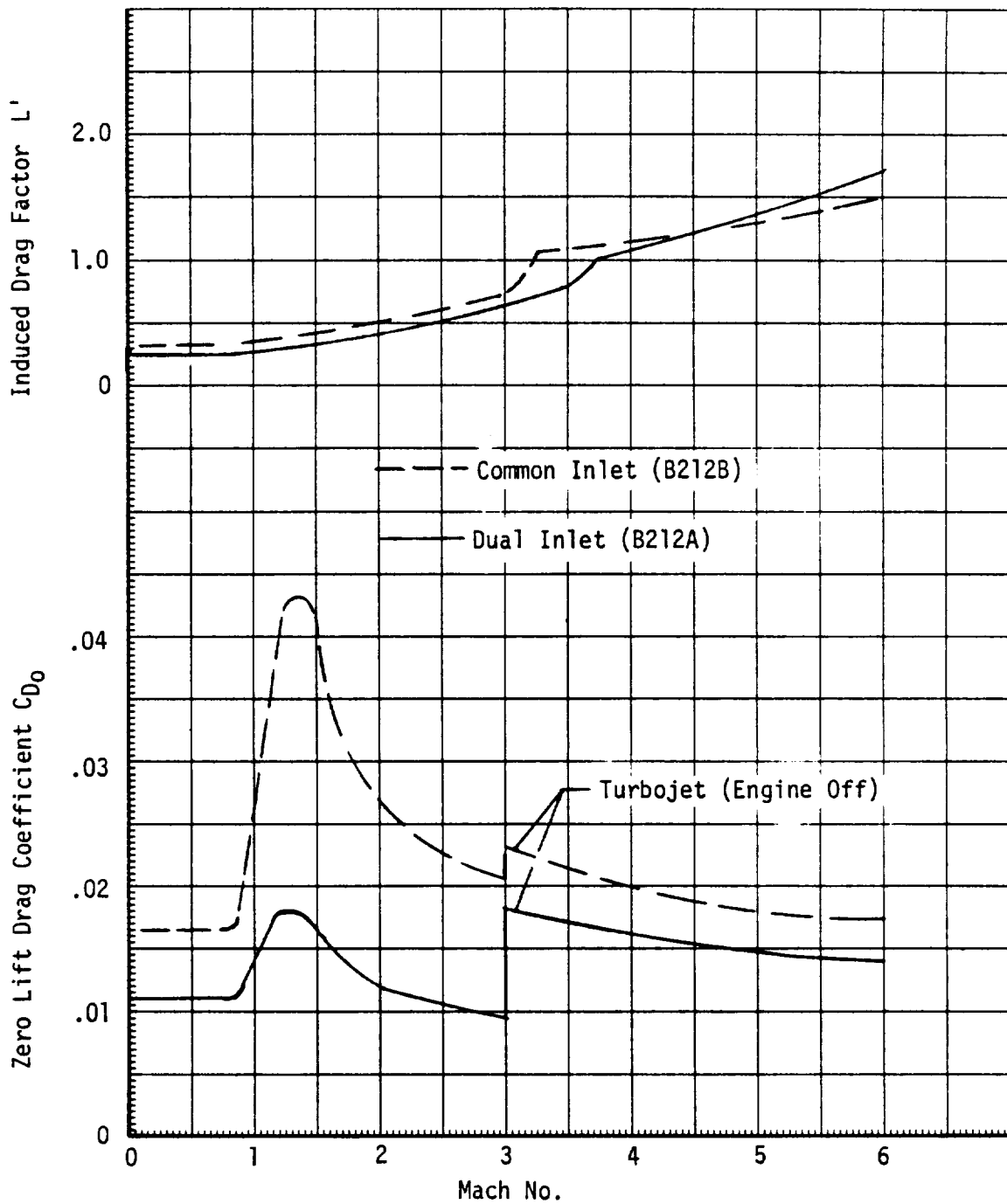
B212B COMMON INLET OPERATION



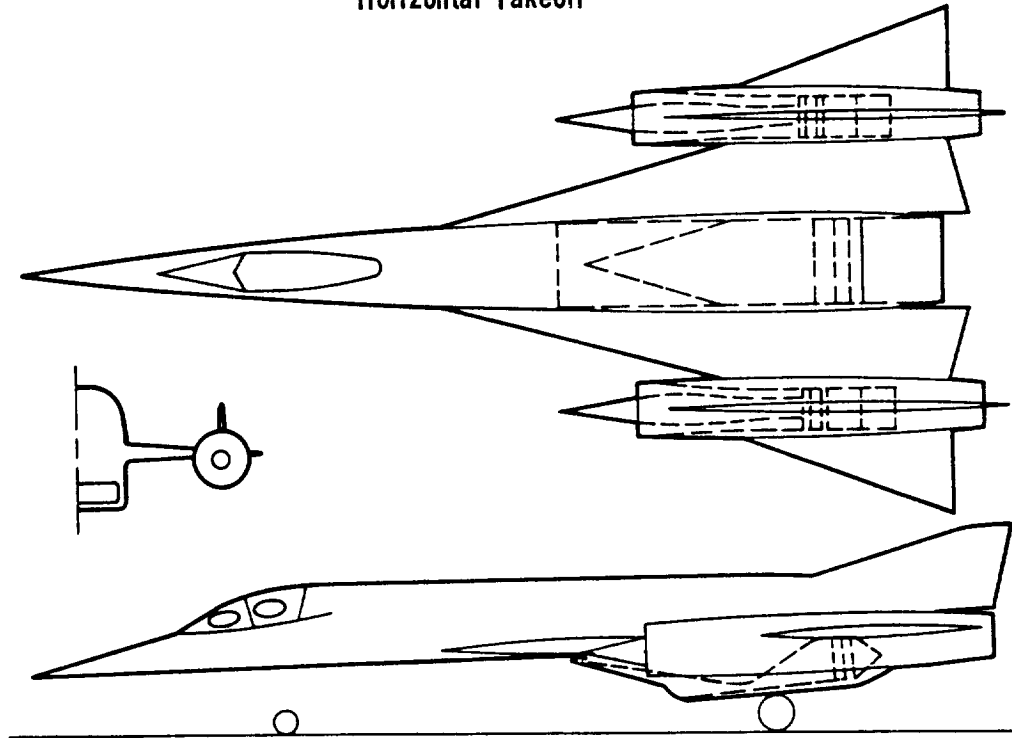
B212C COMMON INLET OPERATION



(U) FIGURE 4-186 EFFECT OF ENGINE INSTALLATION ON
DRAG OF AIRBREATHER WING BODY CONFIGURATION



(U) FIGURE 4-187 MACH 6 WINGBODY
Turbojet-Ramjet
Horizontal Takeoff



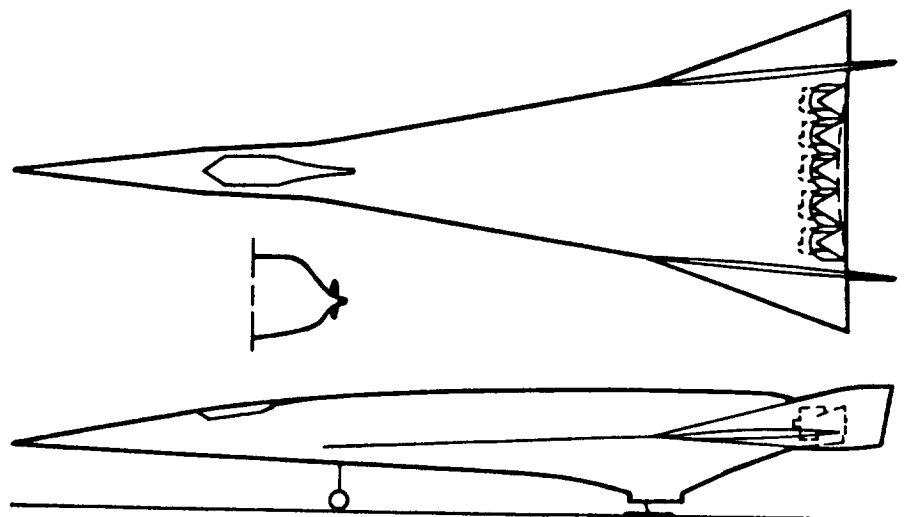
CONFIGURATION REFERENCE B212

O.W.E	47,159 lb
T.O.G.W	55,629 lb
COST	787.0 Million Dollars

SELECTED PARAMETERS

MACH NO.	6	BASING	DUAL
ENGINE - ACCEL	2 F100-GE-100	TPS	ACTIVE
HYP	2d MCAIR Ramjet	TANKAGE	INTEGRAL
THRUST LOADING	.82	FUEL TYPE	LH ₂ (Subcooled) JP (Room Temp.)
TEST TIME	10 Min @ M = 6	WING LOADING	55.4 psf (270.46 kg/m ²)
PAYLOAD	1500 lb (680.4 kg)	FINENESS RATIO	8.5
LOAD FACTOR	5 g (Struct) 3.5 (TPS)	L.E. SWEEP	75°
PLANFORM	1000 ft ² (92.9 m ²)	LANDING POWER	1 Go Around

(U) FIGURE 4-188 MACH 12 ALL BODY,ROCKET-SCRAMJET,AIR LAUNCH



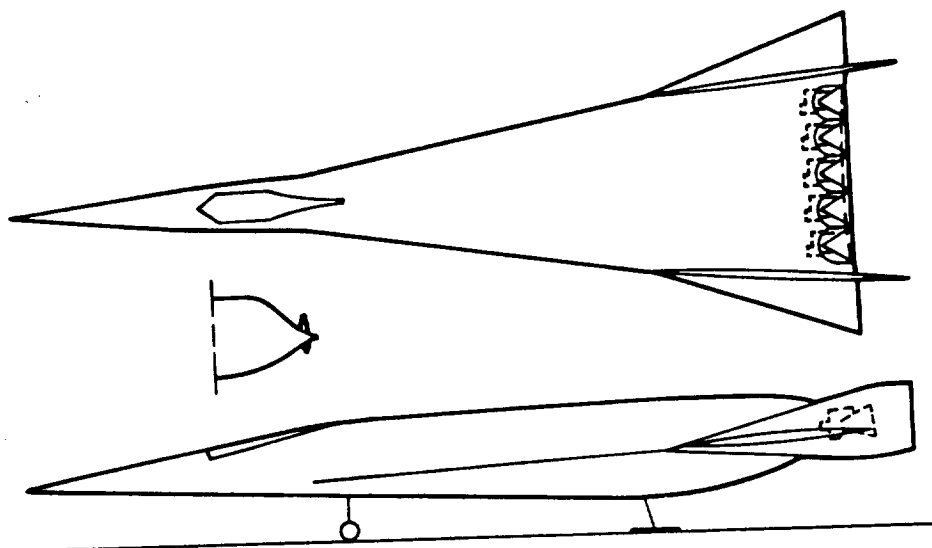
CONFIGURATION REFERENCE B232

O.W.E. 29,050 lb
T.O.G.W. 88,000 lb
COST 817.5 Million Dollars

SELECTED PARAMETERS

MACH NO.	12	BASING	DUAL
ENGINE - ACCEL	5 RL10-A-3-8	TPS	ACTIVE
HYP	SCRAMJET	TANKAGE	INTEGRAL
THRUST LOADING	1.29	FUEL TYPE	LH ₂ (Subcooled)
TEST TIME	5 Min. @ M = 12	WING LOADING	89 PSF
PAYLOAD	1500 lb (680.4 kg)	FINENESS RATIO	(434.53 kg/m ²) 7.1
LOAD FACTOR	5 g (Struct) 3.5(TPS)	L.E. SWEEP	80°
PLANFORM	989 ft ² (91.9 m ²)	LANDING POWER	NONE

(U) FIGURE 4-189 MACH 12 ALL BODY, ALL ROCKET, AIR LAUNCH



CONFIGURATION REFERENCE B233

O.W.E.	25,660 lb
T.O.G.W.	87,000 lb
COST	471.3 Million Dollars

SELECTED PARAMETERS

MACH NO.	12	BASING	DUAL
ENGINE - ACCEL	5 RL10-A-3-8	TPS	ACTIVE
HYP	(1) THROTTLED RL10-A-3-8	TANKAGE	INTEGRAL
THRUST LOADING	1.30	FUEL TANK	LH ₂ (Subcooled)
TEST TIME	5 Min @ M = 12	WING LOADING	86.0 psf (419.89 kg/m ²)
PAYLOAD	1500 lb (680.4 kg)	FINENESS RATIO	7.6
LOAD FACTOR	5 g (Struct) 3.5 (TPS)	L.E. SWEEP	80°
PLANFORM	1010 ft ² (93.9 m ²)	LANDING POWER	NONE

(U) A trade study of the all elliptical body shape versus the blended wing-body shape is discussed in Section 4.4.1. For a comparable mission, this trade showed the all elliptical body exceedingly large and heavy when compared to the blended wing-body shape. This resulted in the selection of the blended wing-body shape as the baseline and "near-optimum" configuration. The fuel trade study in Section 4.4.2.2 shows an advantage in using JP fuel for the turbojet engines; therefore, the "near-optimum" incorporates JP fuel. The turbojet engines are shut down at Mach 3 and the vehicle accelerates on the CSJ to Mach 12 and cruises on the CSJ. Go-around power was deleted based on the trade study in Section 4.4.6, which shows a large increase in fuel requirements and, therefore, cost, weight, and size to achieve this capability. Additional parameters, weights and cost data are shown on Figure 4-190.

(U) Configuration B260 - This Mach 12, horizontal takeoff vehicle has a 5 minute test time at Mach 12. An active thermal protection system, integral fuel tankage, and dual basing are required to reduce vehicle size. The acceleration engine is a J2S "off-the-shelf" rocket with cruise capability through throttling or pulsing the engine.

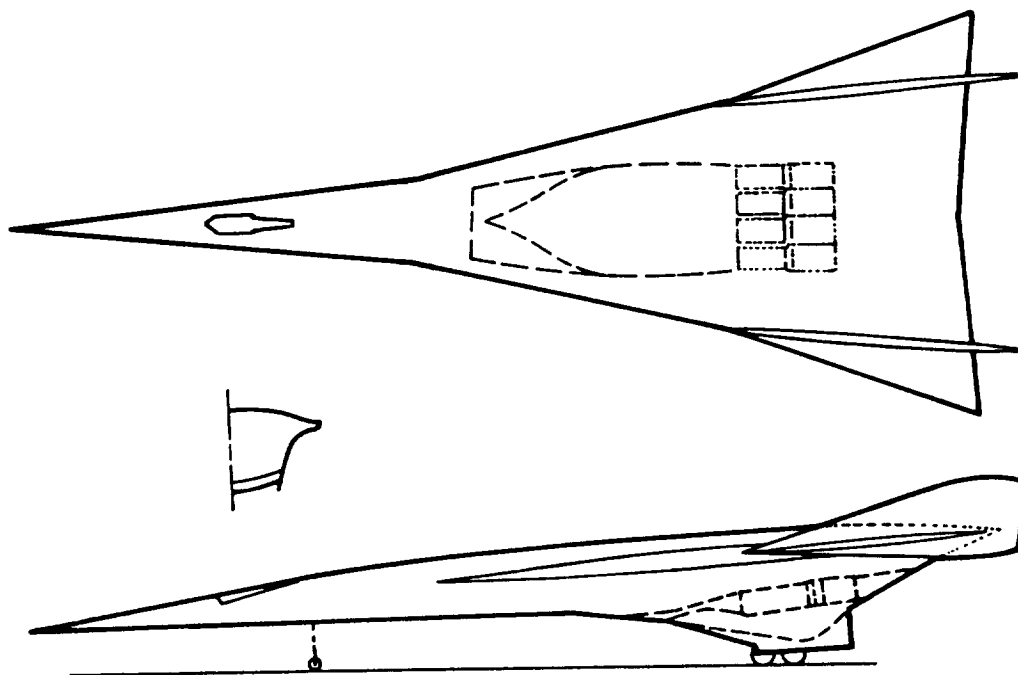
(U) The vertical takeoff mode for this vehicle was evaluated using the baseline vehicle size (B260A) and resulted in a decrease in cruise capability from five minutes to 2.83 minutes. Crew safety in this mode is a problem to be investigated if this operational mode is selected for further study. Additional parameters, weights and cost data are shown in Figure 4-191.

(U) Configuration B284 - This configuration is essentially an unmanned version of B233. It is powered by five RL10-A-3-8 "off-the-shelf" rockets for acceleration with cruise flight maintained by throttling the center rocket with all other rockets shut down. To minimize weight, an active thermal protection system and integral fuel tankage are incorporated and the vehicle is sized for a five minute test time at Mach 12.

(U) This vehicle, as with all other air launched configurations, is dual based because it incurred a large fuel volume increase and therefore cost, weight, and size when operated from a single base. The basing trade study discusses this effect in Section 4.4.6.

(U) The operational problems of an unmanned dual based Mach 12 test vehicle flying across the continental United States are discussed in Section 4.4.6. Additional parameters, weights, and cost data are shown in Figure 4-192.

(U) FIGURE 4-190 MACH 12 BLENDED WING BODY, TURBOJET-CONVERTIBLE SCRAMJET,
HORIZONTAL TAKEOFF



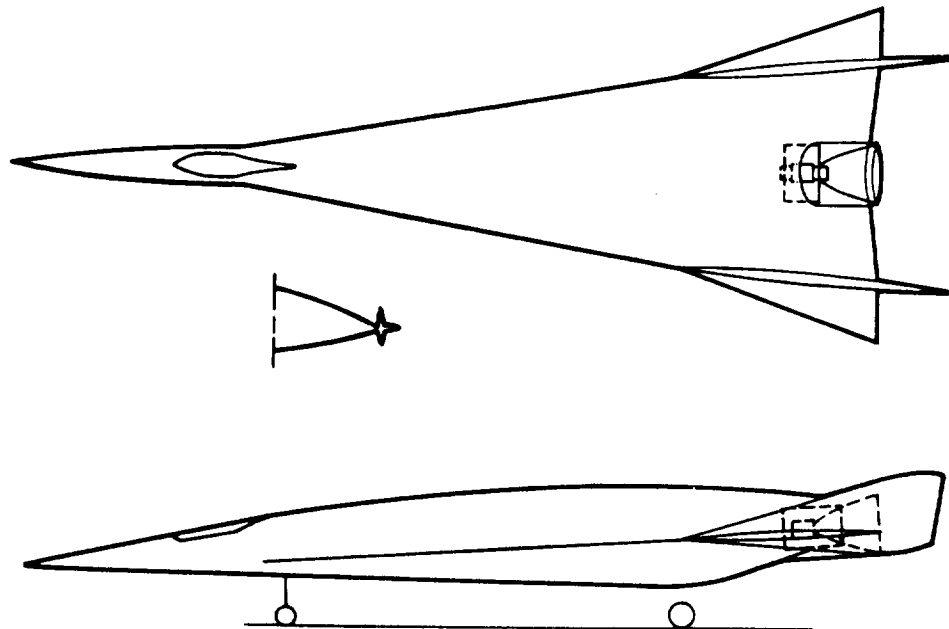
CONFIGURATION REFERENCE B257

O.W.E	64,820 lb
T.O.G.W	99,320 lb
COST	1,077.3 Million Dollars

SELECTED PARAMETERS

MACH NO.	12	BASING	DUAL
ENGINE - ACCEL	4 F100-GE-100	TPS	ACTIVE
HYP	CONV. SCRAMJET	TANKAGE	INTEGRAL
THRUST LOADING	.92	FUEL TYPE	LH ₂ (Subcooled) JP (Room Temp.)
TEST TIME	5 Min @ M = 12	WING LOADING	53.4 psf (260.07 kg/m ²)
PAYLOAD	1500 lb (680.4 kg)	FINENESS RATIO	7.5
LOAD FACTOR	5 g (Struct) 3.5 (TPS)	L.E. SWEEP	77°
PLANFORM	1860 ft ² (172.8 M ²)	LANDING POWER	NONE

(U) FIGURE 4-191 MACH 12 ALL BODY, ALL ROCKET, HORIZONTAL TAKEOFF



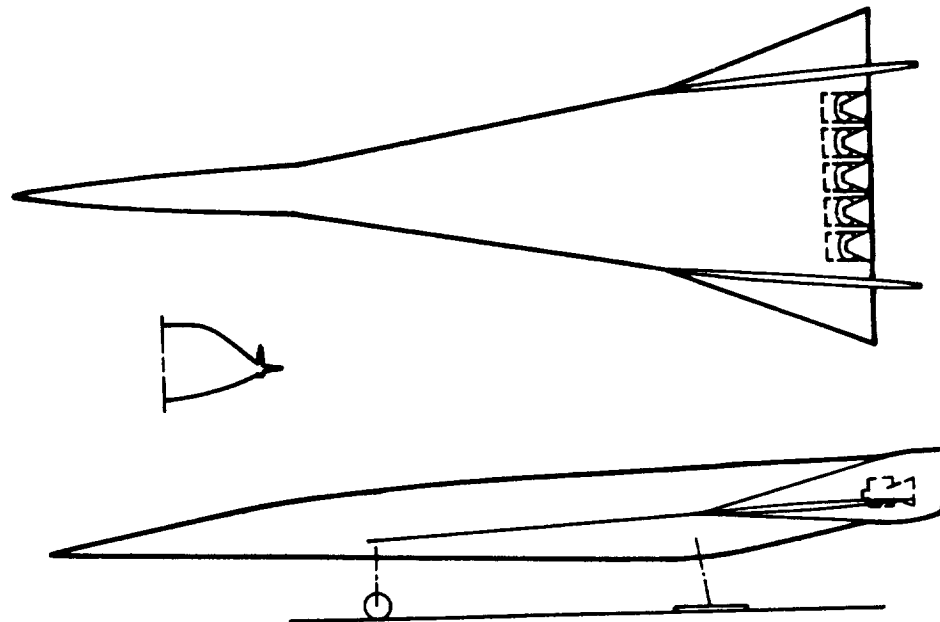
CONFIGURATION REFERENCE B260

O.W.E	41,200 lb
T.O.G.W	167,000 lb
COST	621.5 Million Dollars

SELECTED PARAMETERS

MACH NO.	12	BASING	DUAL
ENGINE - ACCEL	1 J2S	TPS	ACTIVE
HYP	THROTTLED OR PULSED	TANKAGE	INTEGRAL
THRUST LOADING	1.59	FUEL TYPE	LH ₂ (Subcooled)
TEST TIME	5 Min @ M = 12	WING LOADING	103 psf (502.89 kg/m ²)
PAYLOAD	1500 lb (680.4 kg)	FINENESS RATIO	6.9
LOAD FACTOR	5 g (Struct) 3.5 (TPS)	L.E. SWEEP	80°
PLANFORM	1620 ft ² (150.5 m ²)	LANDING POWER	NONE

(U) FIGURE 4-192 MACH 12 ALL BODY -UNMANNED,ALLROCKET,AIR LAUNCH



CONFIGURATION REFERENCE B284

O.W.E	24,250 lb
T.O.G.W.	82,500 lb
COST	542.0 Million Dollars

SELECTED PARAMETERS

MACH NO.	12	BASING	DUAL
ENGINE - ACCEL	5 RL10-A-3-8	TPS	ACTIVE
HYP	(1)THROTTLED RL10-A-3-8	TANKAGE	INTEGRAL
THRUST LOADING	1.38	FUEL TYPE	LH ₂ (Subcooled)
TEST TIME	5 Min @ M = 12	WING LOADING	82.5 psf (402.80 kg/m ²)
PAYLOAD	1500 lb (680.4 kg)	FINENESS RATIO	7.1
LOAD FACTOR	5 g (Struct) 3.5(TPS)	L.E. SWEEP	80°
PLANFORM	1000 ft ² (92.9 m ²)	LANDING POWER	NONE

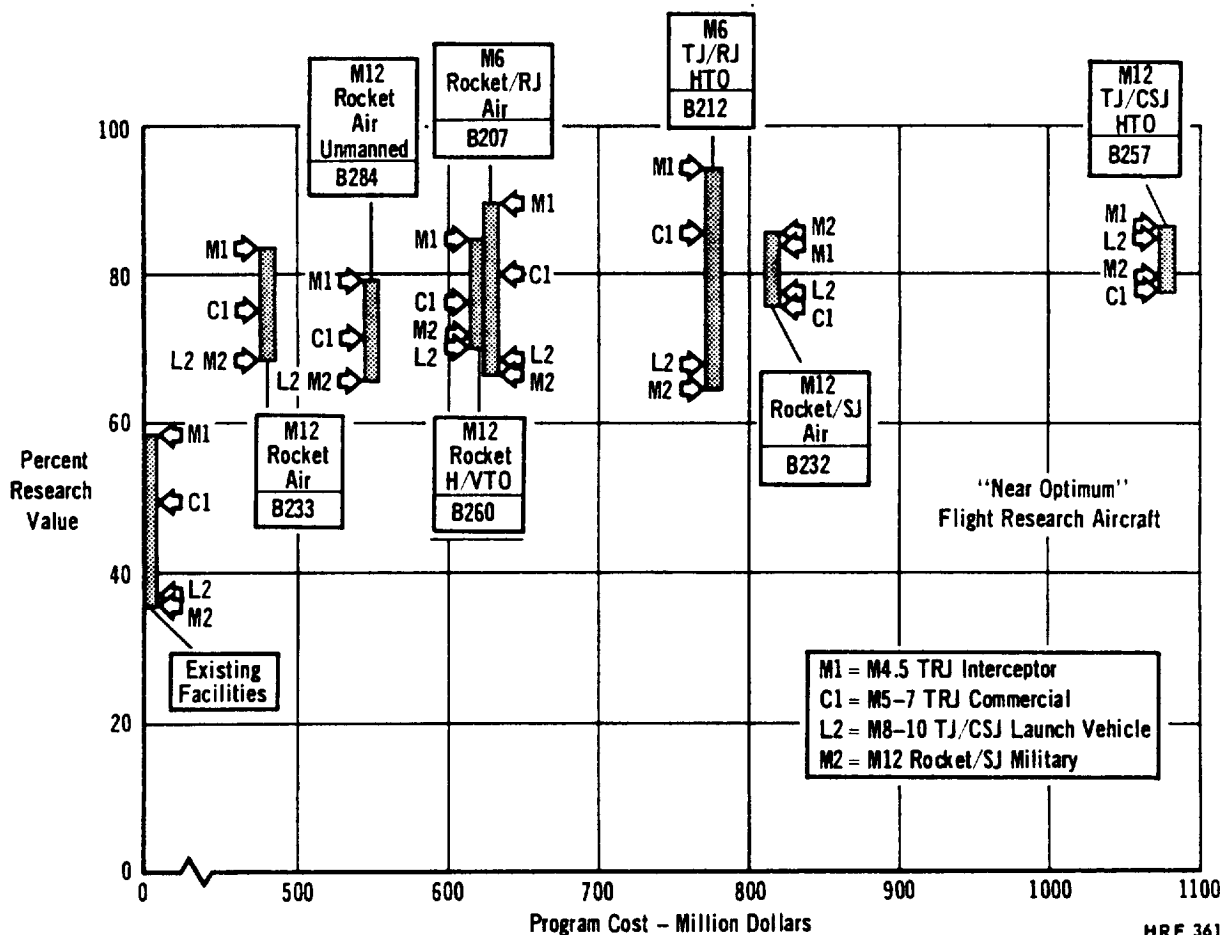
5. FLIGHT RESEARCH VEHICLE SCREENING AND SELECTION

(U) Each of the seven flight research vehicles retained from Phase I has been refined and a "near optimum" configuration developed. The characteristics of these vehicles and the rationale for selecting the combination of design features and operational concepts were presented in Section 4.6. The following sections present comparisons and evaluation of these "near optimum" vehicles. Selection of the most attractive vehicles and recommendations for Phase III refinements are presented in Section 5.3.

5.1 RESEARCH VALUE AND COSTS

(U) Figure 5-1 summarizes the facility research value and program costs for each of the seven "near optimum" vehicles. Research value is presented as a percentage rather than as a quantified value. It must be recognized that the research intrinsic values developed in this study do not represent absolute values, but are on a relative basis. Thus, values are in effect normalized by using percentage of total applicable research accomplished.

(U) FIGURE 5-1
"NEAR-OPTIMUM" FLIGHT RESEARCH VEHICLE VALUE-COST COMPARISONS
Facility Research Values



(U) The research values of existing ground facilities are also presented. Research values are keyed to the operational systems as noted by the horizontal arrows. Four operational systems are used to exemplify the study results. The four operational systems chosen are judged to be representative of the complete spectrum of the nine operational systems used throughout the study. Figure 5-2 summarizes these nine systems. M_1 is chosen since it is the only system in its class. C_1 is representative of L_1 , and covers commercial systems as well as Mach 5 to 7 turboramjet vehicles. L_2 is representative of C_2 and covers launch vehicle systems as well as Mach 8 to 10 turbojet/convertible scramjet systems. M_2 is representative of L_3 , L_4 , and M_3 and covers military systems as well as Mach 12 vehicles employing rocket and scramjet propulsion systems.

(C) FIGURE 5-2 OPERATIONAL SYSTEMS SUMMARY

CODE	TYPE
	RECOVERABLE 1ST STAGE LAUNCH VEHICLE HORIZONTAL TAKEOFF
L_1	Turbojet plus Ramjet, $M = 5$ to 7 staging
L_2	Turbojet plus Convertible Scramjet
L_3	All rocket, Mach 12, staging
L_4	Rocket plus Scramjet, Mach 10 staging
	HYPERSONIC TRANSPORT
C_1	Mach = 6, Turboramjet
C_2	Mach = 10, Turbojet plus Convertible Scramjet
	MILITARY SYSTEMS
M_1	$M = 4.5$ Advanced Manned Interceptor, Turboramjet
M_2	$M = 12$ Strike, Rocket plus Scramjet
M_3	$M = 8$ to 12 Interceptor, Rocket plus Scramjet

(U) Existing facility research value is highest for M_1 (59%) and lowest for M_2 (36%). The band of research value is representative of the current status of the technology base for hypersonic aircraft.

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(U) The band of research value for flight vehicle configuration B233 is narrower than it is for existing facilities, yet the increment between M_1 and C_1 is about the same in both cases. The increment between L_2 , M_2 , and C_1 is significantly lower. This is explained as follows.

(U) The research value of the flight research vehicles includes the value of utilizing ground facilities in the research and development program that is required for development of the research vehicle. This value is somewhat reduced by the fact that there are differences in a number of areas between the flight research vehicles and the operational systems. However, the contribution of existing facilities is still quite high. Thus, the band of research value for configuration B233 is reduced, since the increase in research value provided by the flight research vehicle is greater for L_2 and M_2 than it is for M_1 .

(U) This result is further clarified by considering the incremental increase in research value provided by the flight vehicle over that of existing facilities. The increase for M_1 is from 59 to 84%, or an incremental increase of 25% while the increase for M_2 is from 36 to 69%, or an incremental increase of 33%. Thus, flight vehicle B233 provides a greater increase in capability for the more advanced systems than it does for the nearer term systems.

(U) This identical effect is evident on configuration B284, which is an unmanned version of configuration B233, and on configuration B260, which is a horizontal takeoff version of configuration B233. Similar effects are evident on the other flight vehicles studied.

(U) Another interesting facet of the results is seen by comparing the results for flight research vehicle configuration B232 with those of configuration B233. The research values for operational system M_1 and C_1 are almost identical. However, the contribution of the flight vehicle is much larger for operational system M_2 on configuration B232 than on configuration B233 -- so much so that the position of research value for M_2 is completely interchanged. This reflects the fact that configuration B232 is very nearly a prototype configuration for operation system M_2 , and thus it provides a substantial capability to conduct research for this operational system. This result is further clarified by comparing incremental increases in research value over that provided by existing facilities. As in the previous comparison the increment that the flight vehicle provides for M_1 is 25%. For M_2 , however, the incremental increase is from 36 to 86%, or an incremental increase of 50% compared to 33% for configuration B232. Thus, flight vehicle configuration B233 provides 1.5 times more increase in research value for M_2 than does configuration B232, while it provides the same increase in research value for M_1 .

5.2 VEHICLE EVALUATION AND SCREENING

(U) Configuration B284 is identical to Configuration B233, except that it is unmanned. Both are airlaunched, rocket accelerated and rocket cruise vehicles. Neither vehicle requires the development of new engines to meet the mission performance. Thus, maximum speed and cruise performance are low risk items. Both have excellent speed growth potential and provide good configuration similarity to a variety of potential operational systems. Both vehicles provide a good test range capability for speed build-ups and development of system confidence.

(U) Figure 5-1 shows, however, that Configuration B284 provides about 5% lower research value at an increase in cost of 15%. The decrease in research value results from:

- o No Man-Machine Integration
- o No Man Judgement Ratings
- o No Option for In-Flight Alterations
- o No Pilot Interpretation of Data

(U) The cost increase is due to:

- o Increased system complexity, requiring five flight vehicles rather than three and higher maintenance requirements.
- o Potential program delays and resulting costs
- o No pilot emergency corrective action capability.

(U) Based on these factors,

- CONFIGURATION B284 WAS REJECTED -

(U) Configuration B260 is very similar to Configuration B233 both being manned rocket accelerated and rocket cruise vehicles. Configuration B260, however, incorporates the capability of take off horizontally, as compared to airlaunching for Configuration B233. In addition, the basic vehicle incorporates the capability to accomplish vertical takeoff, with an associated degraded performance. Neither vehicle requires the development of new engines to meet mission performance. Thus, maximum speed and cruise performance are low risk items. Both have excellent speed growth potential. Configuration B260, due to its ability to take off horizontally or vertically, causes the research value for this vehicle to be slightly higher than for Configuration B233. The HTO/VTO capability also results in a closer similarity to the operational system than the airlaunched configuration. The test range capability of Configuration B260 is not as good as for Configuration B233. Speed buildups are feasible from Holloman, Carswell, and Eglin but the test operational flexibility is somewhat reduced.

(U) The overall advantages of Configuration B260, however, are nullified by its much greater costs. As shown by Figure 5-1, Configuration B260 costs about 30% more than Configuration B233. Based on the above comparison,

- CONFIGURATION B260 WAS REJECTED -

(U) Note: The horizontal takeoff/vertical takeoff capability is highly desirable from a research standpoint. Incorporation of this capability in an air-launched vehicle appears quite attractive. It appears feasible to design a basic air-launched vehicle capable of Mach 12 flight to accommodate provisions which would allow usage of the vehicle in both HTO and VTO launch modes. In the alternate launch modes, it would not be necessary to achieve Mach 12 flight, lower speed capability would still satisfy the desired research and provide a direct comparison of operational techniques between the HTO and VTO launch modes.

(U) Configuration B232 is identical to Configuration B233, except that it incorporates an accelerator scramjet providing the capability to either accelerate the vehicle from Mach 8 to 12 or to cruise at any desired speed in this range. Both are air-launched, manned, and rocket accelerated. The following observations apply for Configuration B232:

- o It is a high cost program, requiring scramjet development.
- o Achieving maximum speed performance is low risk.
- o Achieving cruise performance on the scramjet engine represents a low risk item.

Both vehicles have good speed growth potential when utilizing the rocket, and provide good operational system similarity and good test range capability.

(U) Both Configurations 232 and 233 are essentially identical with the only difference being that Configuration 232 incorporates a scramjet engine in addition to the basic rocket engines. Figure 5-1 shows that the research value for Configuration B232 is about 20% higher for operational system M2 than Configuration B233. However the cost penalty of approximately 350 million dollars, makes this configuration less attractive. On the basis of costs,

- CONFIGURATION B232 WAS REJECTED -

(U) Note: While the cost is high for the scramjet configuration, it provides a high capability in an area of research that has been assessed as very desirable. The similarity between Configuration 232 and 233 suggests that an approach of initiating a research program, wherein a basic vehicle employing proven rocket engines might be developed incorporating provisions such that a scramjet engine could be added as a growth option, would be quite desirable.

(U) Configuration B257 is quite different from Configuration B233. It incorporates a turbojet for acceleration, instead of a rocket. Furthermore, it requires acceleration on a convertible scramjet to achieve cruise speed. It does provide horizontal takeoff capability, and is a manned system but the body is a blended body wing-body shape whereas Configuration B233 has an all-body shape.

(U) Observations concerning Configuration B257 are:

- o It is the highest cost vehicle of those studied
- o Development of the convertible scramjet is the most difficult engine development task of those studied.
- o The program represents high risk, since it depends upon development of the convertible scramjet to achieve speeds over M 3 to 3.5

- o Even though the vehicle has turbojets, powered landing is not provided, since the effect on vehicle weight is much too large.
- o The test range capability is poor, due to the long range traversed during acceleration.

Figure 5-1 shows that the research value for the more advanced operational systems is quite high. However, the cost is extremely high. Therefore, on the basis of cost and risk,

- CONFIGURATION B257 WAS REJECTED -

(U) Note: Convertible scramjet testing capability is judged to be of high research value. An approach similar to that possible with a scramjet engine and a basic rocket powered vehicle, discussed in previous paragraphs is quite desirable.

(U) In summary, airlaunched, rocket powered vehicles offer high research value and program flexibility at low risk and cost, and Configuration B233 is the most attractive in the M = 12 class of vehicles studied.

(U) Configuration B207 is similar to Configuration B233 in that the vehicles are both all-body shapes, airlaunched, manned and rocket accelerated. However, they differ widely in their speed class, and in the type of cruise engine they use. From Figure 5-1, it is seen that of all the airbreather configurations studied, Configuration B207 is the lowest cost program and provides a significant research value for all operational systems. This vehicle has an excellent test range compatibility and is suitable for single base operation and a test time of 5 minutes, although increased test time requires a modest increase in the "pilot-in-suit" time. The vehicle has moderate speed growth potential and does not require development of new engines to meet the mission performance. The risk in achieving cruise performance capability is rather low. In general, it is a low risk program.

(U) However, while the aerodynamic similarity to the operational systems is good, none of the operational systems utilize the propulsion combination offered on this vehicle.

(U) To complete the evaluation of Configuration B207 it is necessary to compare it with Configuration B212.

(U) Configuration B212 is a horizontal takeoff, manned, wing-body, turbojet/ramjet vehicle. As shown by Figure 5-1, the vehicle has a good research value, particularly due to the horizontal takeoff capability, at a moderate program cost. Maximum speed performance is dependent upon development of the ramjet engine. However, this represents only a moderate risk. Speed growth potential is rather poor. The vehicle provides a powered landing capability, which adds considerably to the flight operations safety.

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(U) Comparing Configuration B212 research value, risk, and cost with that of Configuration B207, Configuration B207 is more attractive. However, neither Configuration B207 or B212 appear to provide the research potential of Configuration B233. On this basis,

CONFIGURATION B207 WAS REJECTED
- AND -
CONFIGURATION B212 WAS REJECTED

(U) Note: Ramjet and turboramjet testing capability is judged to be of high research value. Incorporation of this capability as part of a follow-on program is quite desirable.

(U) In Phase I, turboramjet systems were evaluated for M = 6 class vehicles and were determined to be attractive. The engine development costs were high, however, (over 500 million dollars) and on this basis turboramjets were not retained for the Phase II study.

(U) Configuration B212 includes engine development costs of 170 million dollars for the ramjets. The turbojets are off-the-shelf and require little development costs for hydrogen conversion. Thus, it appears that while valuable testing could be accomplished on a modification of Configuration B233, it is possible that an attractive M = 6 class horizontal takeoff configuration would be achieved if a turboramjet could be obtained by wrapping a specially developed ramjet around an existing turbojet. Therefore,

A M = 6 CLASS HTO RESEARCH VEHICLE
USING A TURBORAMJET ENGINE,
- WHICH INCORPORATES AN EXISTING TURBOJET, -
IS RETAINED

5.3 RECOMMENDED PHASE III VEHICLES

(U) In Phase I a broad group of flight research vehicle concepts was evaluated and selected vehicles were retained for Phase II refinements. The results of the Phase II refinement studies are presented in the preceding sections of this report. In Phase II, particular choices were made and a number of conclusions were drawn.

(U) A number of observations are available from these Phase I and II study results.

- o The research value for a number of different types of advanced propulsion systems was rated quite high. This implies that a research vehicle should have the capability to test many different propulsion packages.

- o The research value for reusable thermal protection systems was rated quite high. This implies that a research vehicle should have the capability to test a number of different thermal protection system concepts in order to supply comparison data on reusability.

- o The diversity of the defined research clearly indicates that the flight research vehicles should be flexible and adaptable to new and varying research goals. When evaluated for a broad research program capability, unmanned vehicles neither provide this capability nor reduce overall program costs.

- o Program costs for airlaunched vehicles are substantially lower than for ground takeoff vehicles. Furthermore, airlaunched vehicles provide excellent operational flexibility, particularly for speed buildup and envelope expansion.

- o A conservative approach to selecting structural design load factors and to providing space and weight for instrumentation payload is not very costly. In view of the manner in which program requirements usually escalate it appears to be judicious to set a base for design which is a trifle conservative.

- o Increasing the test time for Mach 12 vehicles is extremely costly for rocket cruise vehicles and only slightly less expensive for scramjet cruise vehicles. Mach 6 ramjet cruise vehicles, however, can accommodate increased test time at a reasonable cost.

- o Off-the-shelf rocket engines are available to satisfy the requirements for all rocket accelerated vehicles, reducing initial program risk and cost.

- o Off-the-shelf turbojet engines are also available to satisfy the requirements for all turbojet accelerated vehicles at reduced program cost.

- o The costs of ramjet, scramjet, and convertible scramjet engines are quite high and represent a major cost element for vehicles using such propulsion systems. Furthermore, the risk in developing these engines might compromise the research vehicle program. An off-the-shelf-turbojet with a wraparound ramjet appears feasible, however, and could result in reduced program costs.

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o Propellant costs represent only a minor element in the overall program. The use of LH₂ for rockets and hypersonic engines, rather than storable fuels, appears to be both feasible and economical.

(U) Based on the results of the Phase II parametric studies and comparative evaluations, it is recommended that the following flight research vehicles be retained for Phase II refinement.

(a) Basic System 1 - M = 12, airlaunched, rocket, manned, all body vehicle

Growth or Options to be Studied

- o Incorporate a M=8 to M=12 scramjet
- o Incorporate a M=3.5 to M=12 convertible scramjet
- o Incorporate an integrated two dimensional ramjet
- o Provide a section (complete periphery) of the aircraft suitable for the installation and testing of alternate thermal protection systems. Examine implications of re-skinning the entire vehicle with a different thermal protection system
- o Include provisions for missile launch testing
- o Include provisions for stage separation testing, possibly using a boiler plate second stage
- o Provide horizontal takeoff and vertical takeoff capability at degraded performance
- o Examine capability for testing advanced turbojets, turboramjets, and ramjets
- o Examine suitability of add-on turbojets for takeoff and landings.

(b) Basic System 2 - M=6 horizontal takeoff, turboramjet (1 engine) manned wing-body vehicle

Growth or Options to be Studied

- o Examine advanced technology engine vs near term engine. Make selection and conduct studies on one vehicle
- o Examine the use of rocket power for program initiation. Examine ramjet, scramjet and convertible scramjet test bed capability
- o Provide a section (complete periphery) of the aircraft suitable for the installation and testing of alternate thermal protection systems. Examine implications of re-skinning the entire vehicle with a different thermal protection system
- o Include provision for missile launch testing

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