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Hypersonic Research Facilities Study(U)

Volume IV Part 1

Phase III Final Studies

Flight Research Facilities

Prepared Under Contract No. NAS2-5458
by
Advanced Engineering
MCDONNELL AIRCRAFT COMPANY
for
OART - ADVANCED CONCEPTS AND MISSIONS DIVISION
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Moffett Field, California 94035

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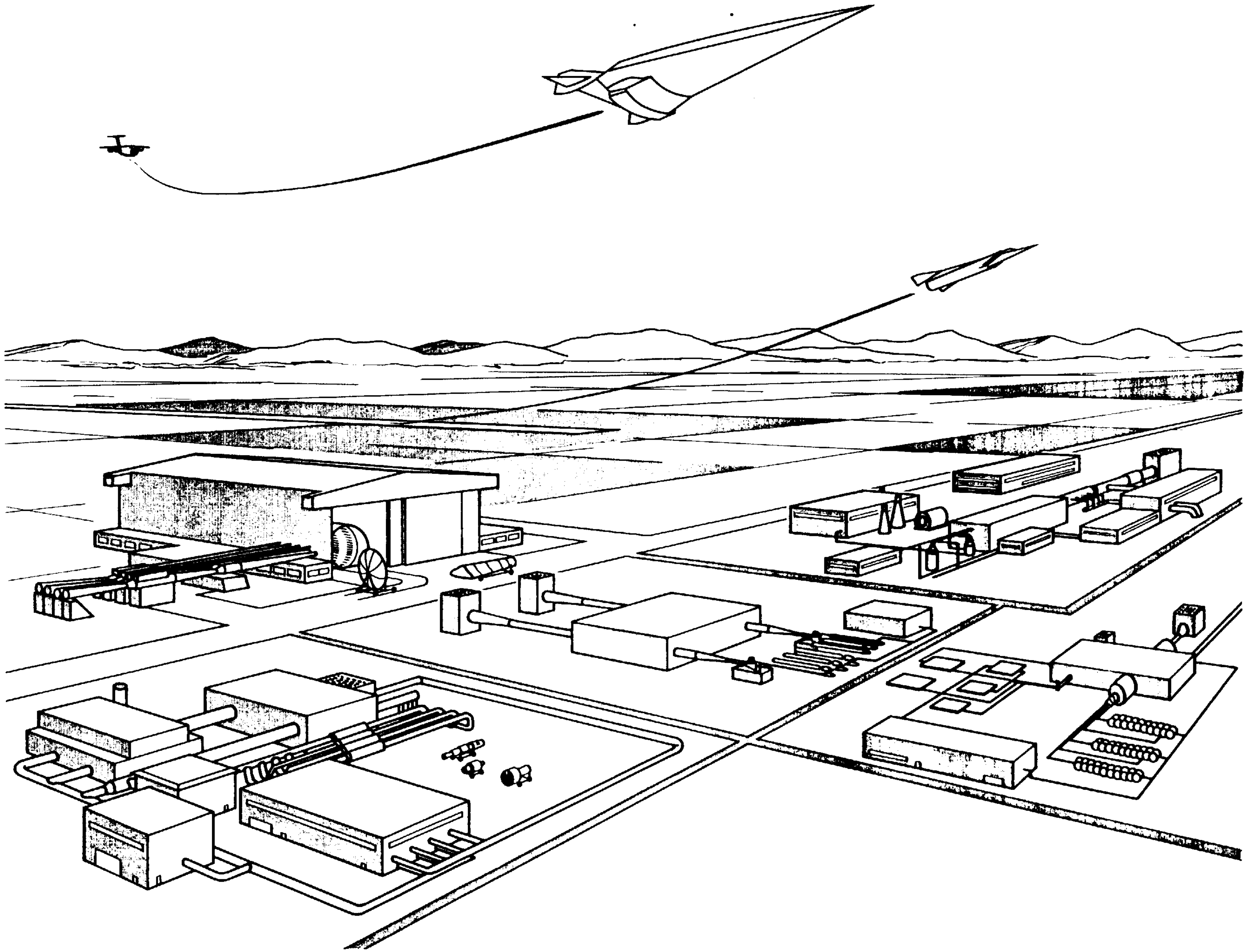

FOREWORD

This report summarizes the results of Phase III of the Hypersonic Research Facilities Study performed from 2 January 1970 through 26 June 1970 under National Aeronautics and Space Administration Contract NAS2-5458 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 support of the following engine companies in the flight vehicle synthesis is gratefully acknowledged: AiResearch Manufacturing Division of the Garrett Corporation, The General Electric Company, The Marquardt Company, and Pratt and Whitney Aircraft.

The basic task of Phase III was to refine the two attractive flight research vehicles retained from Phase II and examine the feasibility of providing growth capability to adapt to varying research goals in these vehicles. The Phase III study has been conducted in accordance with the requirements and instructions of NASA RFP A-15109 (HY-81), McDonnell Technical Proposal Report G970, and OART correspondence received during the Phase III period.

This is Volume IV, Part 1 of the overall HYFAC Report, which is organized as follows:

		NASA CONTRACTOR REPORT NUMBER
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

ACKNOWLEDGEMENTS

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

The following contributed significantly to the contents of this volume:

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T. Smith	Weights
R. Bay	Flight Test
R. Woodward	Flight Vehicle Design
L. Hair	Propulsion
L. Pagel	Thermodynamics
D. Peters	Flight Vehicle Costs
J. Sinnett	Propellants
S. Littlepage	Aerodynamics and Performance
F. Slagle	Flight Vehicle Costs
K. Weber	Structures and Materials
J. Borsheim	Avionics

SUMMARY

(U) The Phase III analyses of Flight Research Facilities performed as a portion of the Hypersonic Research Facilities (HYFAC) Study are presented herein. Two attractive flight research vehicle concepts, geared to the development needs of future (1980-2000) operational hypersonic aircraft systems, are defined. The inherent research capability of each concept is further expanded by incorporating provisions within the basic vehicle design to accommodate the testing of several additional research options. The design feasibility of this approach to achieving increased research flexibility is examined. Near term propulsion systems are employed as the basic power plants to shorten acquisition time, reduce development costs, and provide high confidence in attaining the performance capability desired. Performance is determined for each basic vehicle and for the vehicles when incorporating several optional research packages. Total program cost estimates are developed for each vehicle and for the various research options. This study provides the necessary framework for the formulation of an attractive and needed flight research program which is a key element in the systematic development of an overall advanced research plan.

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LIST OF ABBREVIATIONS

Abbreviation	Definition
ARM	Armament
BLD	Boundary layer diverter
Btu	British thermal unit
c.g.	Center of gravity
cm	Centimeters
CSJ	Convertible scramjet
DCPR	Defense Contractor Progress Report
Deg.	Degree
°F	Degrees Fahrenheit
ft	Feet
G	Autogenous
GE	General Electric Company
HT	Hydrotreated
HTO	Horizontal takeoff
HYFAC	Hypersonic Research Facilities Study
H_2	Hydrogen
in	Inch
Kip	One thousand pounds
KCAS	Knots calibrated airspeed
KSI	Kips per square inch
°K	Degrees Kelvin (absolute)
kg	Kilogram
LE	Leading edge
LH_2	Liquid hydrogen

LIST OF ABBREVIATIONS (Continued)

Abbreviation	Definition
LO_2	Liquid oxygen
lb	Pounds, force
lbm	Pounds, mass
M	Mach
MAC	Mean aerodynamic chord
Max.	Maximum
MCAIR	McDonnell Aircraft Company
MIN	Minimum
m	Meter
min.	Minute (60 seconds)
Mos	Months
$\overline{M}$	Million
N	Newtons
No.	Number
nm	Nautical mile
OWE	Operational weight empty
P&WA	Pratt and Whitney Aircraft
PSF	Pounds per foot squared
psi	Pounds per inch squared
Rad	Radian
R&D	Research and development
RDT&E	Research, development, test, and evaluation
RJ	Ramjet
SJ	Scramjet
STG	Staging

LIST OF ABBREVIATIONS (Continued)

Abbreviation	Definition
sec	Seconds
TE	Trailing edge
TJ	Turbojet
TOGW	Takeoff gross weight
TPS	Thermal protection system
TRJ	Turboramjet
VTO	Vertical takeoff

LIST OF SYMBOLS

Symbol	Definition
A	Area
a	Acceleration
b	Wing span
C_{A_q}	Axial force derivative due to pitching velocity
C_{A_α}	Derivative of axial force with respect to angle of attack
C_D	Drag coefficient, untrimmed except as noted
C_{D_O}	Zero lift drag coefficient, untrimmed except as noted
C_L	Lift coefficient, untrimmed except as noted
$C_{L\alpha}$	Lift curve slope, untrimmed except as noted
$C_{L_{OPT}}$	Lift coefficient for $(L/D)_{max}$
$C_{L_{TD}}$	Lift coefficient at touchdown
C_{ℓ_p}	Rolling moment derivative with respect to roll rate
C_{ℓ_r}	Rolling moment dynamic derivative due to yawing velocity
$C_{\ell_{\delta_a}}$	Rolling moment derivative due to aileron deflection
C_{ℓ_β}	Rolling moment derivative due to sideslip angle
C_m	Pitching moment
C_{m_α}	Derivative of pitching moment coefficient with respect to angle of attack
C_{m_q}	Pitching moment derivative due to pitching velocity
C_{N_q}	Normal force derivative due to pitching velocity
C_{N_α}	Derivative of normal force coefficient with respect to angle of attack
C_{n_r}	Yawing moment dynamic derivative due to yawing velocity
C_{n_β}	Yawing moment derivative due to sideslip angle
$C_{n_{\delta_r}}$	Yawing moment due to rudder deflection

LIST OF SYMBOLS (Continued)

Symbol	Definition
C_R	Wing root chord
C_T	Wing tip chord
C_{Y_r}	Side force dynamic stability derivative due to yawing velocity
C_{Y_β}	Side force derivative due to sideslip
$C_{Y_{\delta_r}}$	Side force derivative due to rudder deflection
F_{tu}	Ultimate tensile strength
F_{ty}	Yield tensile strength
g	Acceleration due to gravity
I_{sp}	Specific impulse
I_{xx}	Mass moment of inertia about the longitudinal axis
I_{xz}	Mass product of inertia about the longitudinal and vertical axes
I_{yy}	Mass moment of inertia about the lateral axis
I_{zz}	Mass moment of inertia about the vertical axis
K	10^3
L'	Induced drag factor, untrimmed
L/D	Lift to drag ratio, untrimmed except as noted
$\dot{m}$	Mass flow
N_z	Flight path normal load factor
O/F	Oxidizer to fuel weight flow ratio
P	Pressure
P_{OSC}/P_{AV}	A measure of the ratio of the oscillatory component of roll rate to the average component of roll rate for a step differential stabilator command
q	Dynamic pressure

LIST OF SYMBOLS (Continued)

Symbol	Definition
$\dot{q}$	Aerodynamic heating rate
S	Area (aerodynamic reference area equals S_P)
T/W	Thrust to weight ratio
$\dot{w}$	Weight flow
X	Length measured from nose aft
α	Angle of attack
	Ratio of wing span to vehicle length
β	Sideslip angle
	Ratio of mean aerodynamic chord to vehicle length
Δ	Increment between two values
$\Delta\beta_{max}$	Maximum sideslip excursion occurring within the greater of two seconds or one-half period of the dutch roll oscillation for a step differential stabilator command
δ	Deflection
ε	Nozzle expansion ratio
ζ_d	Damping ratio of the dutch roll mode
ζ_{sp}	Damping ratio of the longitudinal, short period mode
ϕ	Fuel equivalence ratio (ratio of actual fuel flow to stoichiometric fuel flow)
	Bank angle
ψ_β	Phase angle of the dutch roll oscillation in sideslip
ω_{n_d}	Undamped natural frequency of the dutch roll mode
$\omega_{n_{sp}}$	Undamped natural frequency of the longitudinal, short period mode

LIST OF SUBSCRIPTS

Subscript	Definition
ACT	Actual body length
c	Capture, a fixed reference area on a given vehicle
H	Horizontal stabilator
max	Maximum
0	Evaluated at zero lift
	Isentropic reservoir conditions
	Freestream
p	Planform, or associated with pressure forces
Ref	Reference
S	Projected area
SB	Speedbrake
w	Wetted area
π	Associated with maximum vehicle cross section
∞	Freestream
2	Conditions at engine face

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1. INTRODUCTION

(U) The Phase III results of an investigation of Flight Research Facilities, performed as a part of the Hypersonic Research Facilities (HYFAC) Study, are presented herein. The primary objectives of the HYFAC Study are to assess the research requirements associated with the development of future (1980-2000) operational hypersonic aircraft and, based on these requirements, provide the NASA with descriptions of a number of desirable facilities with which to accomplish the necessary research. Flight research aircraft and new ground research facilities were evaluated to provide an assessment of their capabilities, performance, cost, and acquisition time. The capabilities of existing ground facilities were also assessed for comparison.

(U) The HYFAC Study was performed in three phases. Phase I involved the screening of a broad array of possible research facility concepts. Those concepts appearing most attractive were retained for future trade studies during Phase II. At the conclusion of Phase II, the field of candidate facilities was narrowed once again. Only the most promising air and ground facilities were retained for further refinement during Phase III. Two of these facilities were flight research vehicles, namely a Mach 6 aircraft and a Mach 12 aircraft. It is the analysis of these flight research vehicle concepts that is presented herein (Volume IV, Part 1).

(U) Parallel Phase III efforts pertaining to ground research facilities are presented in Volume IV, Part 2. An analysis of the anticipated research requirements and the potential of each Phase III air and ground facility in satisfying them is presented in Volume IV, Part 3.

(U) The major emphasis of the Phase III refinement studies of flight research facilities concentrated on four elements.

- Basic Research Capability
- Optional Research Flexibility
- Design Feasibility
- Program Cost

These were considered to be the key elements in developing flight research aircraft that would satisfy the basic premise employed throughout the HYFAC Study. Namely, to provide a broad research program capable of expanding the current technology base as a whole rather than in specific narrowly defined areas typical of singular prototype testing.

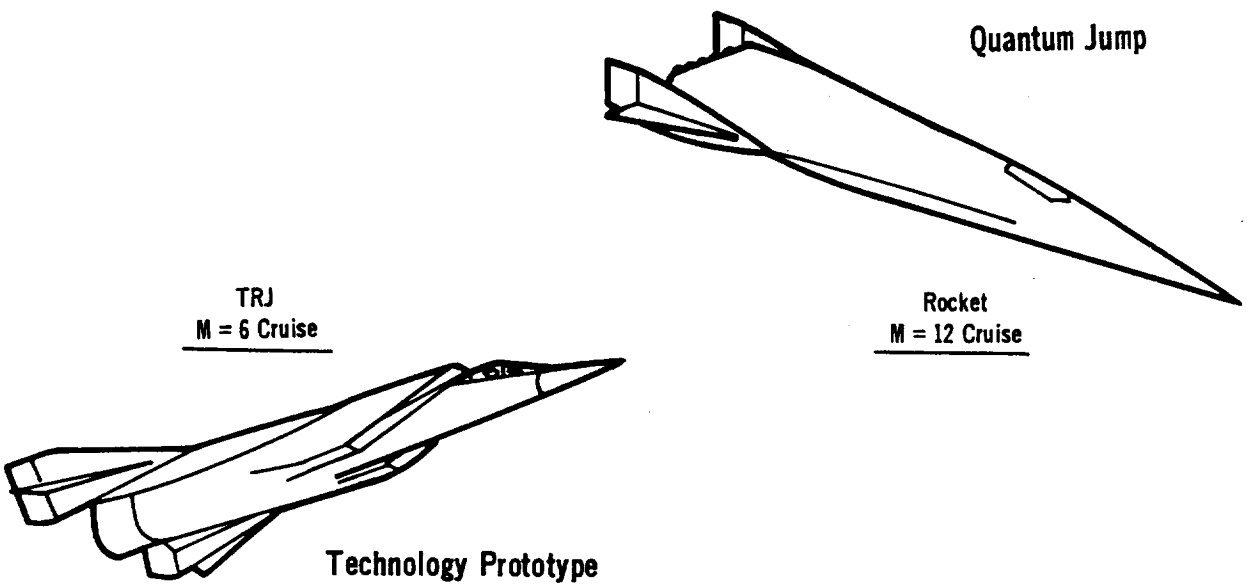
(U) To accomplish this goal, two aircraft were selected; one representing in a broad sense a near term technology prototype (Mach 6), and the second providing a quantum jump in the current hypersonic technology base (Mach 12). The inherent research capabilities of each vehicle were then expanded to provide optional research flexibility by designing to accommodate testing in several additional areas beyond the capabilities of the basic vehicles. To assure design feasibility, installation provisions for the various test options were considered in the basic vehicle design. Furthermore, near term propulsion systems were selected to power the two basic aircraft. This precludes pacing the research program to the development of an advanced engine and also avoids the high costs associated with engine development. At the same time, the option of testing future engines is retained by utilizing the design concepts embodied in the Phase III analysis.

(U) These efforts have resulted in the definition of attractive and desirable flight research facilities capable of fulfilling the aforementioned HYFAC Study goals. If the overall objective of developing hypersonic aircraft for operational use in the 1980-2000 time period is to be met, the need for a flight research program similar to that described herein is imperative.

2. APPROACH TO PHASE III

(U) During Phase II a selected group of flight research vehicles were refined. Parametric trade studies of a number of design and operational options were conducted to determine the performance, research capability, and cost effects. As a result of these studies, two distinctly different, attractive concepts were selected for detailed refinement studies in Phase III, as illustrated in Figure 2-1.

(U) FIGURE 2-1
TWO FEASIBLE RESEARCH VEHICLES



(U) To provide a near term technology prototype aircraft, a Mach 6 airbreathing configuration was selected. This vehicle, a turboramjet powered aircraft, provides capability for technology demonstration of advanced airbreathing propulsion systems, as well as a broad spectrum of research applicable to the defined potential operational systems. The vehicle is designed for steady state cruise at Mach 6 for five (5) minutes, operates in a conventional ground takeoff mode, and is manned. It employs the existing Pratt and Whitney J58 JP fueled turbojet engine together with a LH_2 fueled wraparound ramjet modification. This will provide early research on a near term turboramjet engine. At the same time, the aircraft can be designed to accept an advanced compound airbreathing engine when it becomes available for tests. Therefore, the development of this concept will not be paced by the parallel development of an advanced engine, nor burdened by the associated costs.

(U) To provide a quantum jump in performance, a Mach 12 rocket powered aircraft was selected. This vehicle is manned, airlaunched by a C-5A, and designed to cruise for five minutes at Mach 12. The engines employed are modified P&WA RL10-A-3-9 rockets using LO_2/LH_2 propellants. Five engines are employed for acceleration to cruise speed and cruise is achieved on a single engine throttled to approximately 30% thrust. Like the Mach 6 concept, the use of existing engines will free the research program of engine development costs and preclude pacing the aircraft development to a parallel advanced engine development program. Providing the capability to accommodate future testing of advanced airbreathing engines is an attractive option available with this vehicle.

2.1 OBJECTIVE

(U) The objective of the Phase III studies was to refine the more promising vehicles and determine in greater depth their research capabilities, costs, and development schedules.

(U) Specific emphasis was given to examining approaches to expand the research capability of each vehicle by adapting various research options to the basic vehicle. In this manner, a significant improvement in overall performance capability can be achieved and thus provide a broad degree of research flexibility and versatility.

(U) Particular areas of interest include:

- Engine Performance
- Flight Control
- Thermal Protection
- Armament Installation
- Launch Modes
- Staging Operations

(U) The fundamental design objective was to provide research capability over a wide range of flight conditions using techniques which are feasible within current day technology and can realistically be attained at low cost and high confidence.

2.2 GROUND RULES

(U) General study ground rules applied to all phases of this study are listed below. Other ground rules which apply to specific segments of the study are presented in the appropriate sections of the report.

- All cost estimates are reported in 1970 dollars.

- The assumed state of the art is commensurate with initiation of facility development during the time period from 1970 to 1975. Wherever feasible, proven technology (or technology expected to be proven by the start date) was utilized. Where such design was not feasible, conservative overdesign practices, requiring minimum improvements in the state of the art, were followed.

- Close coordination is assumed between the NASA and the contractors who are building facilities or aircraft, thus minimizing the need for extensive documentation and quality assurance programs.

- Aircraft construction is assumed to conform to experimental shop procedures.

- The development costs for flight research vehicles include all necessary engine and avionics development costs.
- It is assumed that engines need not be developed to the reliability normally required for operational (non-research) use.
- The primary flight safety criterion is that no single component malfunction shall cause a catastrophic situation.
- Reliable rocket or airbreathing engine performance, consistent with that required for JP fueled, single-engine aircraft, is required during takeoff and climb to 25,000 ft (7630 m).
- The vehicle landing characteristics are to be suitable for unpowered landing by a skilled pilot. Adequate fuel reserves are provided to compensate for uncertainties in engine SFC, for meteorological and operational dispersions in fuel consumption, and for powered emergency operations.
- Edwards Air Force Base is considered as the primary operational field for flight research vehicles.
- It is assumed that maximum use will be made of existing or planned tracking and communications facilities.
- The U.S. Standard Atmosphere - 1962 is used throughout the study.

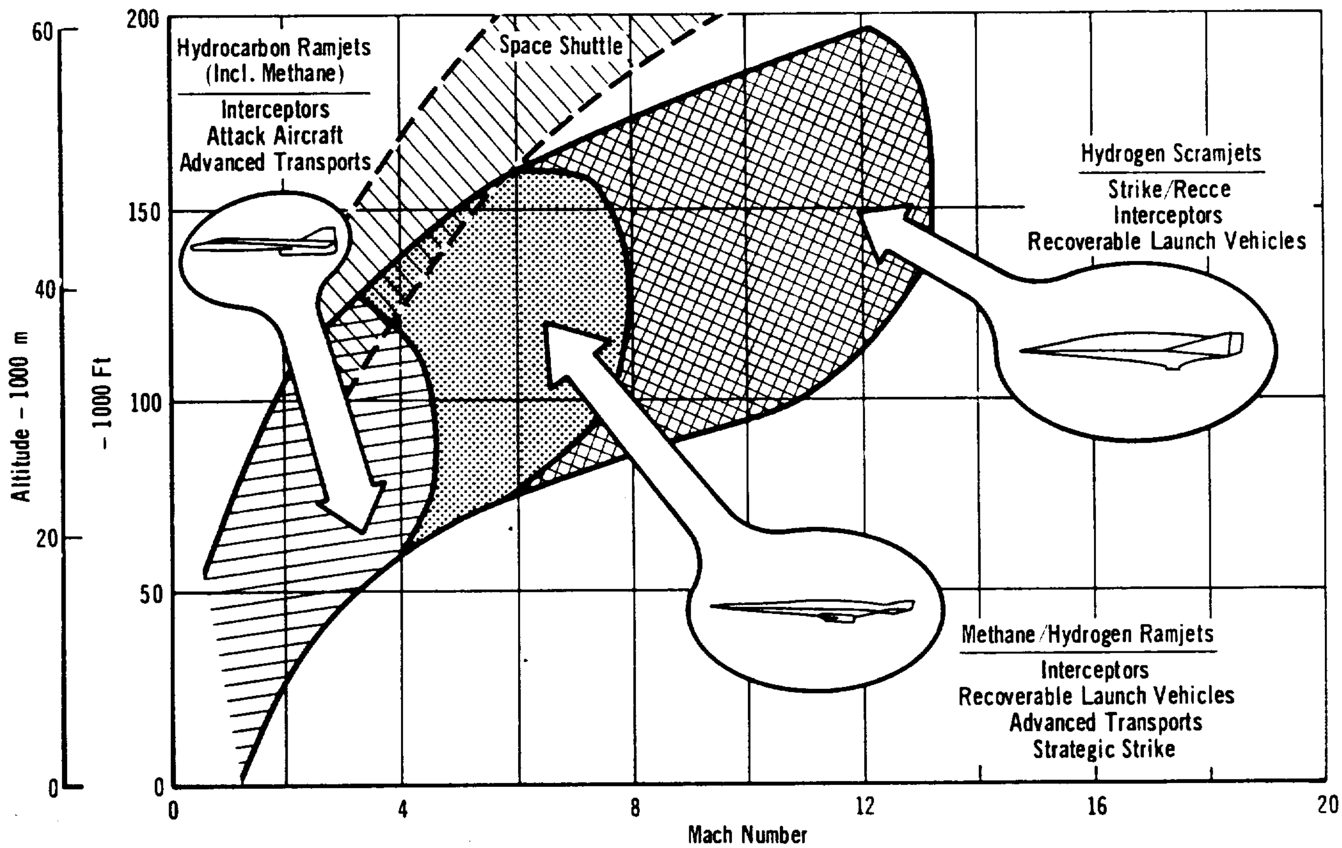
2.3 DESIGN MISSION

(U) For consistency in performance evaluations, a design mission was established for each of the two basic research aircraft concepts under consideration. These design missions are based on providing steady state cruise capability in the flight regimes illustrated in Figure 2-2 for future operational aircraft systems. The selected design missions are shown by superposition in Figure 2-3 and are discussed in the following sections. Alternate mission profiles employed in study of the various research options are discussed in the appropriate sections.

2.3.1 (U) MACH 12 AIRCRAFT - The following design mission profile applies for the Mach 12 research aircraft con ept and is essentially the same as that used in Phase II. The vehicle research instrumentation payload is 1500 lb (681 kg).

- Air launch at .8 Mach number and 35,000 ft (10,680 m)
- Climb and accelerate along the flight path for minimum fuel usage to Mach 12 and the equilibrium altitude for cruise at $(L/D)_{max}$
- Cruise for five minutes at Mach 12 and $(L/D)_{max}$
- Descend unpowered along the equilibrium glide path corresponding to $(L/D)_{max}$
- Land unpowered

(U) FIGURE 2-2
FUTURE AERONAUTICAL SYSTEM APPLICATIONS

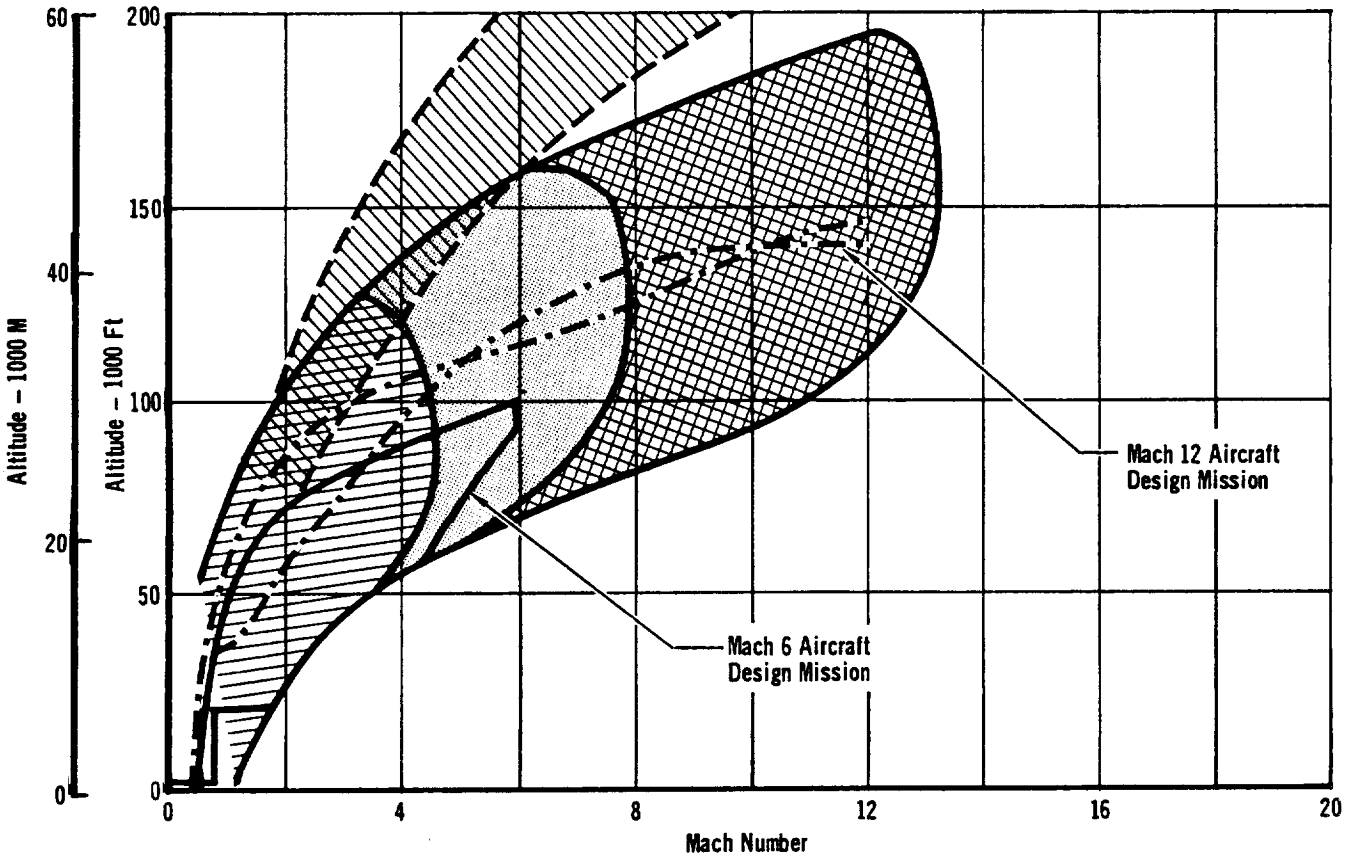


The low dynamic pressure climb profile employed in the rocket aircraft design mission is optimized for minimum fuel usage by the Method of Steepest Descent Trajectory Optimization Computer Program, Reference (1).

2.3.2 (U) MACH 6 AIRCRAFT - The following design mission profile applies to the Mach 6 research aircraft concept and is essentially the same as that used in Phase II. The vehicle research instrumentation payload is 1300 lb (590 kg).

- o Horizontal takeoff
- o Accelerate to .8 Mach number
- o Climb to 20,000 feet (6100 m)
- o Accelerate to design dynamic pressure, 2000 lb/ft^2 (95,800 N/m)^2
- o Climb and accelerate at design dynamic pressure to inlet duct pressure limit, 150 lb/in^2 (103.4 N/cm^2)

(U) FIGURE 2-3
FLIGHT RESEARCH DESIGN MISSIONS



- o Climb and accelerate at inlet duct pressure limit to Mach 6
- o Climb at Mach 6 to the equilibrium altitude for cruise at $(L/D)_{max}$
- o Cruise for five minutes at Mach 6 and $(L/D)_{max}$
- o Descend unpowered along the equilibrium glide path corresponding to $(L/D)_{max}$
- o Land unpowered

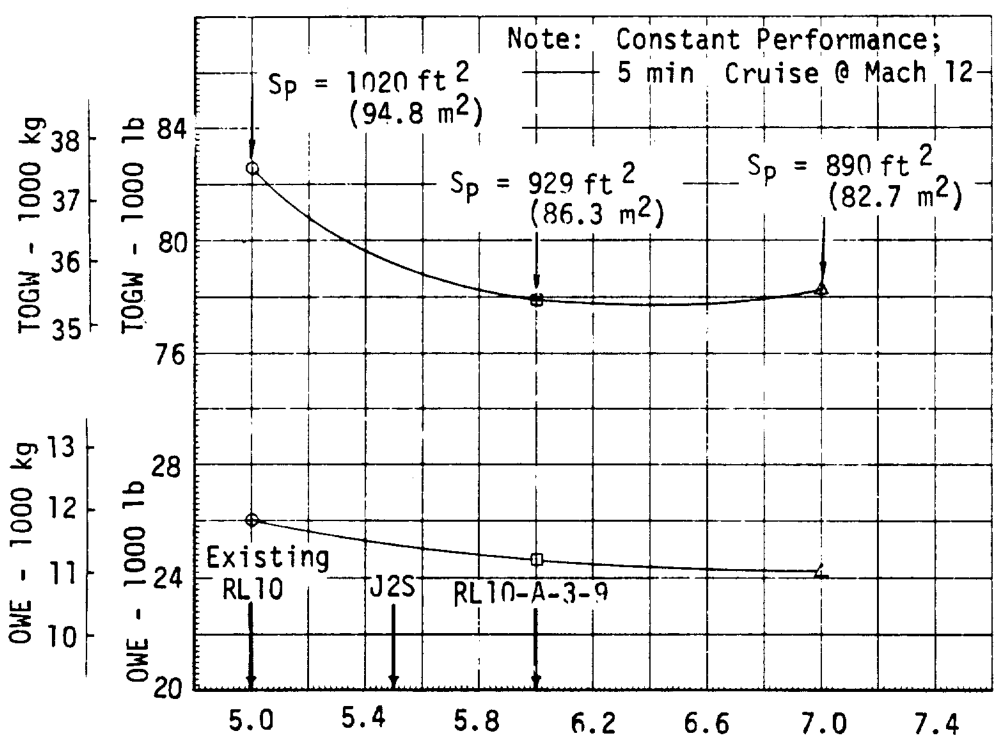
As a result of the Phase II studies, a cruise time of 10 minutes and a payload of 1500 pounds was recommended for the Mach 6 concept. However, with the choice of utilizing an existing fixed size engine, the weight and cost penalties of designing to 10 minutes cruise were judged as not effective, and the cruise time at Mach 6 was reduced to five minutes. This value is identical with the cruise time employed for the Mach 12 design mission. Further, designing the vehicle for a 1500 pound payload would have resulted in marginal performance, thus it was decided to reduce the payload to 1300 pounds. The high dynamic pressure climb profile employed in this design mission is judged as a reasonably optimum profile for an airbreathing accelerator and, therefore, extensive parametric trajectory studies were not deemed necessary.

2.4 FINAL TRADE STUDIES

(U) Prior to initiation of the refinement of the two promising research aircraft, a few trade studies were conducted to assure that the best initial design approach was being used.

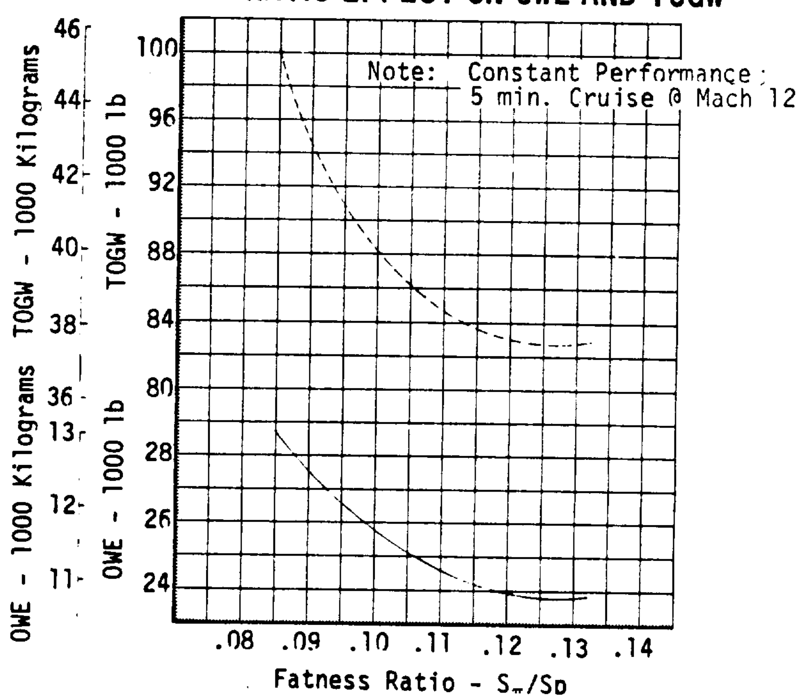
(U) Mach 12 Vehicle - Prior to sizing the aircraft to meet the design mission and initiating Phase III analyses, two interim refinement studies were performed. An oxidizer to fuel (O/F) ratio trade study was conducted to determine the best ratio from the standpoint of minimizing vehicle size and weight. The basis for the study was the Phase II configuration employing the rubberized LR-129 rocket engine. The results are shown in Figure 2-4 based on constant design mission performance. The minimum planform area (Sp) and operational weight empty (OWE) are obtained for an O/F near 7.0 and the minimum takeoff gross weight(TOGW) occurs at an O/F of approximately 6.5. Discussions with Pratt and Whitney indicate that the modified RL10-A-3-9 engines selected for use can be designed to operate efficiently at an O/F of 6.0, rather than the nominal value of 5.0 presently quoted for current versions of the RL10 engines. Therefore, an O/F of 6.0 was selected for the Phase III analysis.

(U) FIGURE 2-4
MACH 12 ROCKET
OXYGEN/FUEL VARIATION



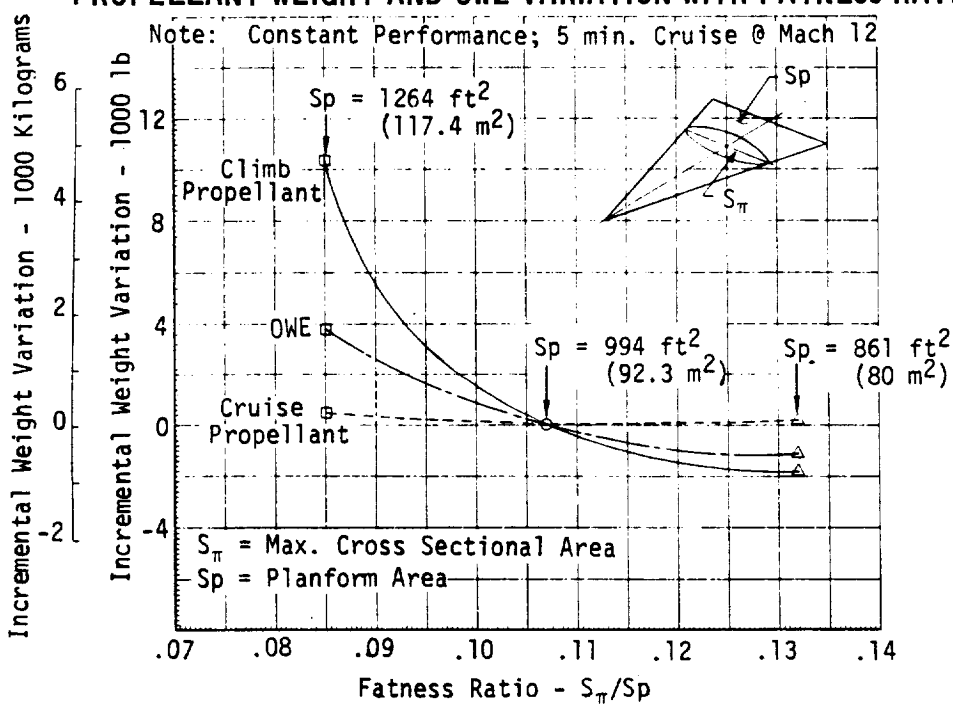
(U) A second interim study was performed to determine the best configuration shaping with regard to fatness ratio (Sπ/Sp), defined as the ratio of maximum vehicle cross section area to planform area. Three vehicle designs with varying Sπ/Sp were investigated. The chordwise location of Sπ was fixed at 75% of the wing root chord; wing sweep, and lower surface compression contours were held constant. Upper surface contours were shaped to obtain the variations in Sπ/Sp shown in Figure 2-5. On the basis of constant design mission performance, OWE, TOGW, and Sp are minimized at an Sπ/Sp of approximately 0.125. A comparison of propellant requirements by flight phase, in Figure 2-6, indicates that the acceleration climb is the dominant influence in determining the optimum Sπ/Sp. Thus, in the Phase III analyses, an Sπ/Sp of 0.125 is employed for the Mach 12 research aircraft concept. For an operational vehicle with extended cruise times and/or a lower cruise Mach number and altitude, lower values of Sπ/Sp may result in improved performance.

(U) FIGURE 2-5
MACH 12 ROCKET
FATNESS RATIO EFFECT ON OWE AND TOGW

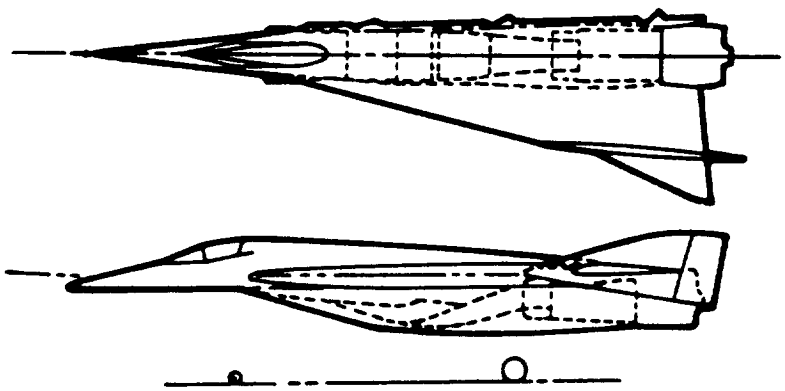


(U) Mach 6 Vehicle - Three inlet arrangements were investigated for use on the Mach 6 configuration. These designs illustrated in Figure 2-7 included two lower fuselage mounted inlet installations, one with vertical ramps and one with horizontal ramps, and a shoulder mounted horizontal inlet ramp design which was finally selected. The lower fuselage horizontal inlet ramp arrangement was somewhat more attractive, on the basis of relative inlet weight and aircraft drag analyses, than either the shoulder mounted inlet or the vertical ramp installation. However, both lower fuselage inlets suffered from the standpoint of possible foreign object damage during takeoff and ground operation. Furthermore, such an arrangement leaves very little lower surface area free for use in providing additional research configuration options. Therefore, the shoulder mounted design appeared most attractive and was selected for the Phase III refinement analysis.

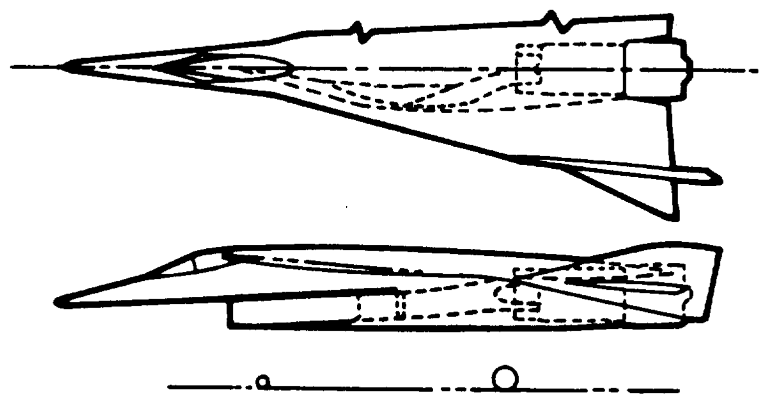
(U) FIGURE 2-6
MACH 12 ROCKET
PROPELLANT WEIGHT AND OWE VARIATION WITH FATNESS RATIO



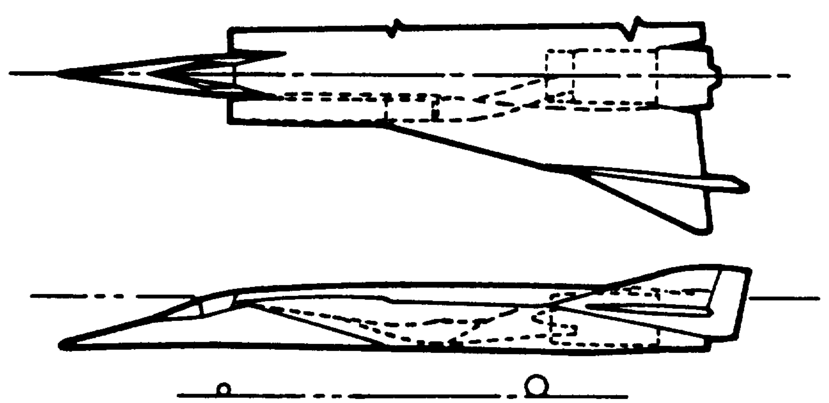
(U) FIGURE 2-7
INLET DUCT VARIATIONS



Lower Surface Horizontal Duct



Lower Surface Back to Back Bifurcated Duct



Horizontal Bifurcated Shoulder Inlet Duct

2.5 CONFIGURATION OPTIONS

(U) In order to expand the research potential of the two basic vehicle concepts selected, several modifications are considered for incorporation as configuration options. The research options investigated are noted below. In some cases, a number of alternate approaches were examined. For such cases, the vehicle design was developed for each approach and a selection of the most attractive approach was made prior to proceeding with the refinement analysis. Each option examined is judged as being a feasible representative approach to provide the capability to perform the desired research.

Mach 12 Vehicle

- Rocket powered horizontal takeoff (HTO)
- Rocket powered vertical takeoff (VTO)
- Scramjet engine installation (SJ)
- Convertible scramjet engine installation (CSJ)
- Ramjet engine installation (RJ)
- Subsonic turbojet low speed research (TJ)
- Thermal protection system variations (TPS)
- Armament installations (ARM)
- Staging operations (STG)
- Alternate rocket engine installation (J2S)

Mach 6 Vehicle

- Convertible scramjet engine installation (CSJ)
- Thermal protection system variations (TPS)
- Armament installations (ARM)
- Advanced turboramjet installation (JZ6)
- Ramjet engine installation (RJ)

3. STUDY METHODOLOGY

(U) The methods, techniques, and rationale employed to design, size, perform, weigh, and cost the two basic vehicles and the selected options are discussed in the following sections.

3.1 DESIGN METHODOLOGY

(U) The Phase III design methodology is predicated on further refining two basic flight vehicle concepts, each having inherent research applicability. These concepts were evolved during Phase I and Phase II design studies. The goal in Phase III was to further expand the research potential of these concepts by examining design methods to include provisions in the basic vehicle to accommodate various research options. The options considered were in the form of relatively simple and complete research packages that could be incorporated on the basic vehicles. The impact this approach has on basic vehicle design was examined from the standpoint of added structural requirements, increased weight, performance capabilities, research potential, and cost.

(U) The basic Mach 12 design is based on the all body, air launched, manned, rocket powered vehicle concept developed during Phase II of the study. The all body configuration is as described in Phase II insofar as the aircraft nose, forebody shape, ramp angle, and aft fuselage underbody shape are concerned. However, the fuselage upper shear line has been modified as a result of the fatness ratio study discussed in Section 2.4 resulting in a minor cross sectional shape difference. A fatness ratio of .125 appears "near optimum" for the research aircraft design mission selected and is employed in the design of the basic Mach 12 aircraft. The propulsion system incorporates "off-the-shelf" RL10-A-3-9 rocket engines. Five engines are employed and all are utilized during the acceleration and climb to test altitude and Mach number. At this point, four engines are shut down and the center one is throttled back to approximately 30% of full power for cruise flight.

(U) The basic Mach 6 vehicle evolved from the wing body, horizontal takeoff, manned, turboramjet powered, concept of Phase I. The vehicle is designed around the near-term compound turboramjet (STRJ11A-27) engine. This engine employs the existing JP fueled J58 turbojet engine as a core with a wraparound ramjet modification fueled by liquid hydrogen. The inlet arrangement, fuel tank integration, and inlet variable geometry have been refined to provide the vehicle with five minutes of cruise time at Mach 6.

(U) To expand the research capability of each vehicle, techniques of providing simple modification kits for various research options were developed and investigated for structural impact and test capability. For example, to install a convertible scramjet on the basic Mach 12 vehicle will require providing hard points in the basic structure which can be strengthened locally to carry and redistribute the resultant highly concentrated loads. Such hard points and structural beef up are required to accept the engine package as well as the engine actuators, ramp actuators and thrust and drag linkages. The design objective was to develop a structural configuration in the basic vehicle which would allow incorporation of the various research packages with only minor aircraft modifications required. Commonality of structural beef up and location of hard points was a fundamental goal. This was accomplished for the scramjet and convertible scramjet installations.

Furthermore, some of the same hard points will be usable in providing a horizontal takeoff (HTO) and vertical takeoff (VTO) modification for the air launched Mach 12 vehicle. Electrical, cryogenic, and hydraulic provisions can be provided in a similar manner for each modification option by designing a disconnect panel and thus provide easy access to the required power and fluids near the modified area.

3.2 AERODYNAMIC METHODOLOGY

(U) The methods employed to determine Phase III vehicle aerodynamic characteristics are presented in the following sections. The Phase II methods of predicting lift and drag are again employed to generate the values used in performing the Phase III vehicles. However, two computer programs not previously utilized in the study, are employed to substantiate the lift and drag values used in the performance analyses. These programs are the Gentry Arbitrary Body Program and the Harris Wave Drag Program, References (2) and (3). The Gentry program is also used in determining the supersonic and hypersonic static stability and control characteristics of the basic Phase III vehicles. These, in turn are employed in dynamic stability analyses to determine the unaugmented handling qualities of the bare airframe.

3.2.1 (U) LIFT AND DRAG ESTIMATION - The basic methods utilized in determining the lift and drag characteristics of the Phase III vehicles for performance analyses are the same as those employed in Phase II. These procedures are described in detail in Volume III of this report. Essentially, they consist of a component buildup of lift and drag contributions as determined from design charts for various configuration elements. In order to verify these Phase II methods, two additional aerodynamic computer programs are employed.

(U) The Gentry Arbitrary Body Computer Program, Reference (2), is utilized to substantiate the basic methods of predicting lift curves, induced drag, and skin friction drag at supersonic speeds. The Gentry Program computes surface pressures and skin friction forces. It then integrates these forces to obtain aerodynamic coefficients and stability derivatives for the various vehicle components, as well as the complete configuration. There are a wide variety of pressure calculation methods provided with the program. The theories and assumptions given in Figures 3-1 and 3-2 have been selected for use in the Gentry analysis of the basic vehicles. Orthomat plotter drawings employed to check the input geometry of each vehicle are shown in Figures 3-3 and 3-4.

(U) The Harris Wave Drag Computer Program, Reference (3), is utilized to verify the basic method of estimating zero-lift wave drag between Mach numbers of 1 and 3. This program is based on the supersonic area rule where the computed drag is the integrated average of the drag on equivalent bodies of revolution. As with the Gentry program, a complete description of the vehicle geometry is input to the Harris program for the drag analysis. However, the vehicle geometry input can be modified to include the effect of flow separation. This is accomplished by terminating the input body geometry at the aft body station where it is estimated that separation will occur.

(U) FIGURE 3-1
MACH 6 TURBORAMJET
GENTRY PRESSURE THEORIES
Impact/Shadow

S_{Ref} - 1103 ft^2 ($102.47\ m^2$) Span - 37.2 ft (11.34 m) l_{ref} - 913.0 in, (Body Length) (23.19 m)

Theory Employed in Analysis

Component/Mach No.	2.0	4.0	6.0
Fuselage	5/3	5/3	5/3
Boundary Layer Diverter, $(q/q_\infty) = .4$	4/3	4/3	4/3
Wing	13/3	13/3	13/3
Vertical Tail	4/3	4/3	4/3
Horizontal Tail	4/3	4/3	4/3

Theory Identification Code

Impact Areas

4 - Tangent Wedge Empirical
5 - Tangent Cone Empirical
13 - Delta-Wing Empirical

Shadow Areas

3 - Prandtl-Meyer Expansion From Free-stream

Skin Friction

Laminar - Reference Temperature Method
Turbulent - Spalding - Chi Method

(U) FIGURE 3–2
MACH 12 ROCKET
GENTRY PRESSURE THEORIES
Impact/Shadow

S_{ref} = 880 ft^2, Span = 31.7 ft, l_{ref} = 951 in (Body Length)
(81.75 m^2) (9.66 m) (24.15 m)

Theory Employed in Analysis

Component/Mach No.	2.0	4.0	6.0	9.0	12.0
Fuselage (Fwd., Mid. and Aft.)	5/3	5/3	14/3	14/3	14/3
Vertical Tail	4/3	4/3	4/3	4/3	4/3
Horizontal Tail	4/3	4/3	4/3	4/3	4/3

Theory Identification Code

Impact Areas

4 - Tangent Wedge Empirical
5 - Tangent Cone Empirical
14 - Dahlem - Buck Empirical

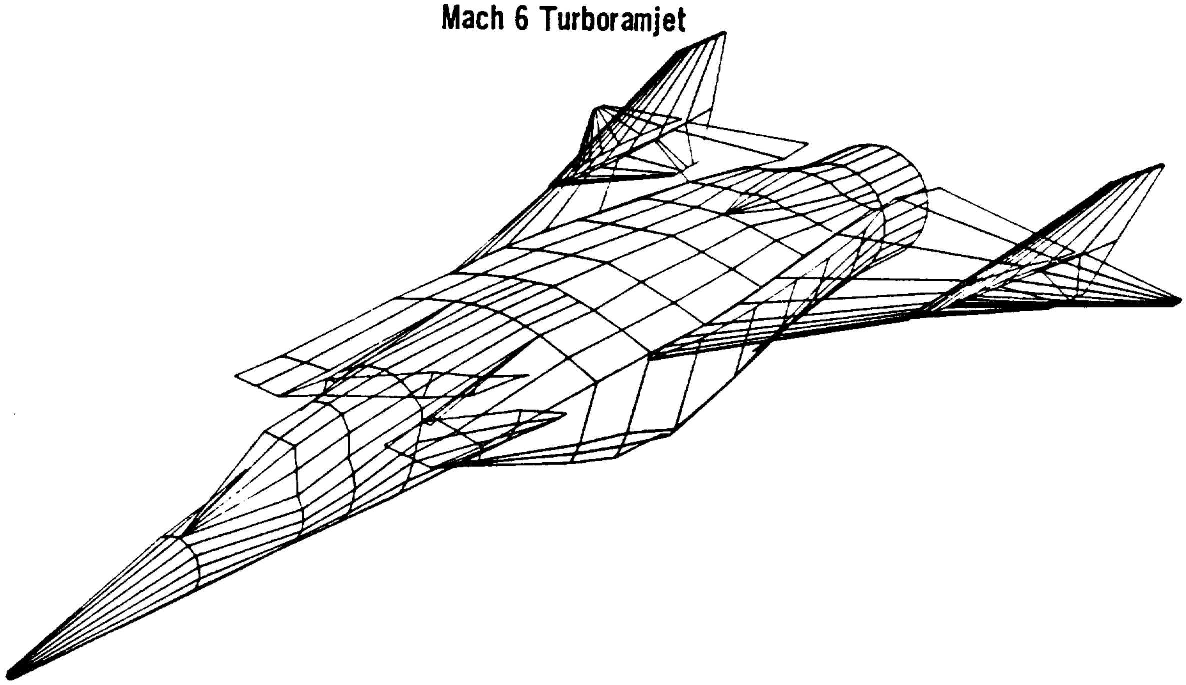
Shadow Areas

3 - Prandtl - Meyer Expansion
From Free-stream

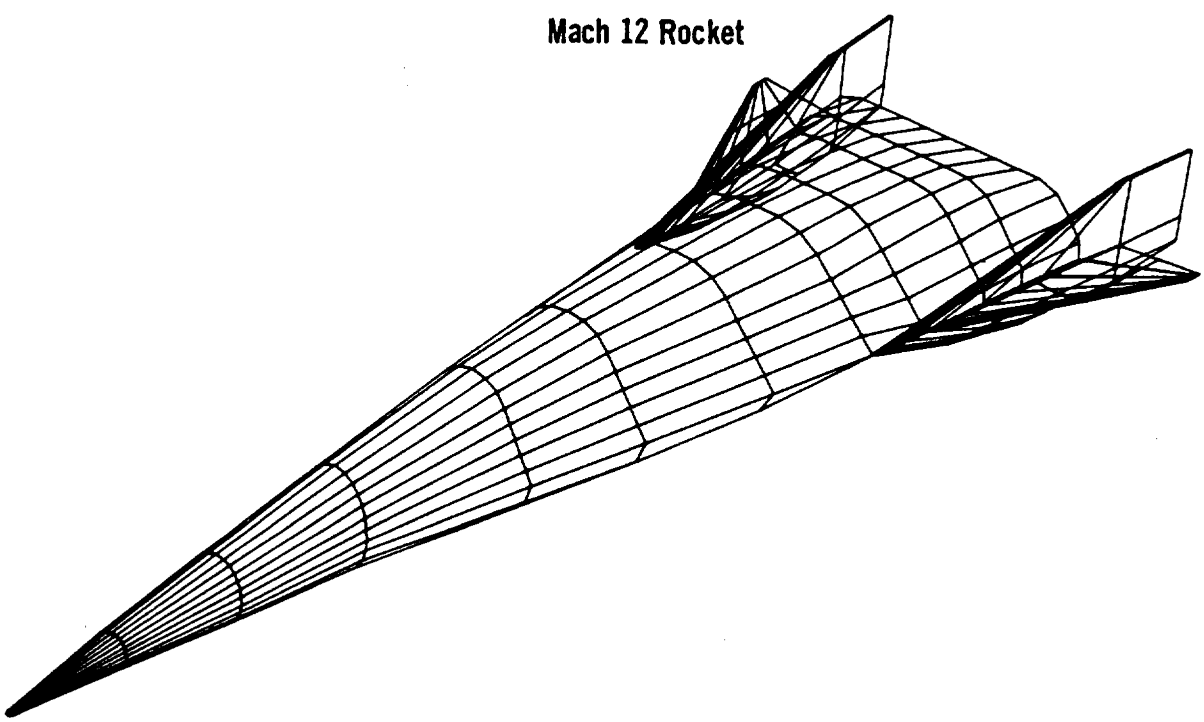
Skin Friction

Laminar - Reference Temperature Method
Turbulent - Spalding - Chi Method

(U) FIGURE 3-3
ORTHOMAT PLOT OF GENTRY INPUT GEOMETRY
Mach 6 Turboramjet



(U) FIGURE 3-4
ORTHOMAT PLOT OF GENTRY INPUT GEOMETRY
Mach 12 Rocket



3.2.2 (U) STABILITY AND CONTROL ANALYSIS - The Gentry Arbitrary Body Computer Program is employed to determine the longitudinal and lateral-directional static stability and control characteristics for the basic vehicles of Phase III at supersonic speeds. The effects of horizontal stabilator deflection are estimated using this program and these data are employed to determine the trim requirements during the climb and glide phases of flight. The Gentry program is also used to determine the effect on stability of deflecting the rudders outboard symmetrically for use as speed brakes. Subsonic stability and control characteristics are generated using component build-up techniques and design charts, e.g., Reference (4).

3.2.3 (U) HANDLING QUALITIES ANALYSIS - The trim stability and control derivatives obtained as output from the Gentry program, are utilized with the dynamic stability computer programs of References (5) and (6) to obtain the vehicle unaugmented longitudinal and lateral-directional handling qualities. These programs calculate the coefficients of the three-degree-of-freedom, small perturbation, longitudinal and lateral-directional equations of motion. These coefficients are then used to determine the coefficients of the characteristic equation and the numerators of the airplane transfer functions for the given control input. The characteristic equation and the transfer function numerators are factored and the factors are used to compute several of the more pertinent flying qualities parameters. These parameters are compared with the current military flying quality specifications, Reference (7).

3.3 PROPULSION METHODOLOGY

(U) The Phase III propulsion efforts required selecting and evaluating propulsion systems, and determining the installed engine performance for the

- Basic Mach 6 vehicle
- Basic Mach 12 vehicle
- Propulsion research options, both vehicles

Phase II results provided the general guidelines employed in selecting Phase III propulsion systems, and these were as follows:

- Use of near term technology engines where available
- Identification of appropriate systems from engine company studies where systems were not currently available (SJ and CSJ)
- Use of high performance engine air inlets consistent with available installation arrangements
- Selection of engine exhaust nozzles to match the engine operating characteristics.

Specific methods and ground rules employed to select and evaluate the Phase III propulsion systems are summarized in Figure 3-5. Where applicable, Phase II methods were utilized to determine the installed engine performance. These methods were supplemented when necessary by appropriate techniques, as noted in Figure 3-5.

(U) FIGURE 3-5
SUMMARY OF PROPULSION TECHNICAL APPROACH

Propulsion System	Component	Selection Criteria	Performance Methods
Rocket	Engine	Total program cost per Phase II results	Engine specification
	Nozzle	Preclude nozzle flow separation, using Phase II	
Subsonic TJ	Inlet	Typical current design	Inlet recovery from test data, as in Phase II; thrust & I_{sp} from engine specification, corrected for installation
	Engine	Availability, thrust	
	Nozzle	To match engine operating characteristics	
Turboramjet	Inlet	To match available installation	Inlet recovery from test data, as in Phases I, II; thrust & I_{sp} from engine specification, corrected for installation
	Engine	Total program cost, thrust	
	Nozzle	To match engine operating characteristics	
Scramjet		Retained Phase II design	Inlet efficiencies from inviscid, perfect gas analysis; thrust & I_{sp} from MCAIR cycle analysis; drag of vehicle forebody assessed as vehicle drag (all as in Phase II).
CSJ		Engine company study	
SJ, CSJ	Inlet	Propulsion system efficiency, integration	
	Nozzle	Large expansion area, integration (as in Phase II)	

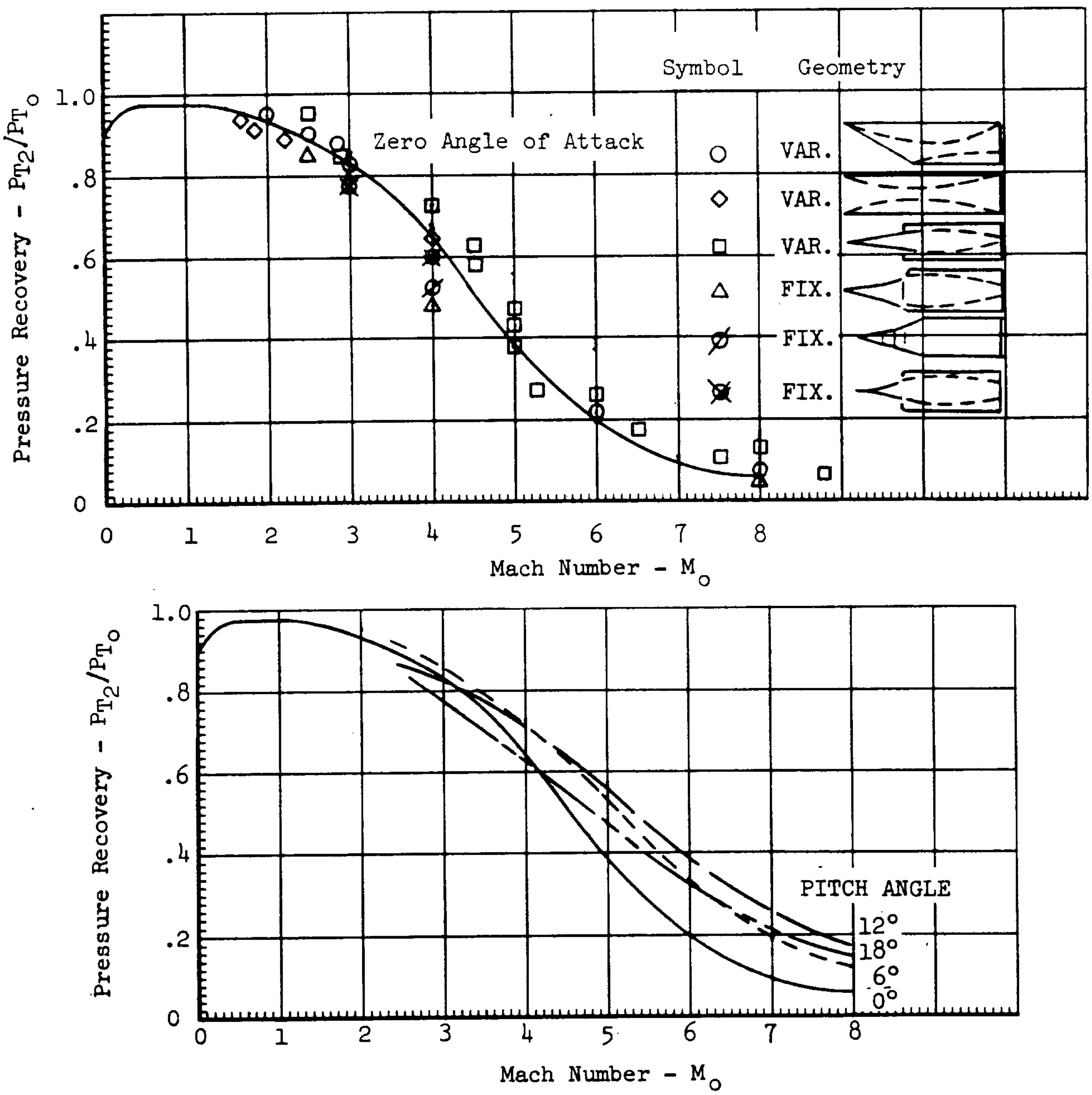
(U) Some refinements in vehicle performance were made by assessing the amounts of inlet bleed and bypass drags more rigorously than in Phase II, using the method of Reference (8). Likewise, mean values of inlet pressure recovery, including angle of attack effects, were determined based on a compilation of test data. The pressure recovery used for the turbojet and turboramjet flight profiles is shown on Figure 3-6.

3.4 PERFORMANCE METHODOLOGY

(U) The basic Mach 6 and Mach 12 vehicles are sized to meet the design missions described in Section 2.3 and thereby provide five minutes of cruise at their respective design Mach numbers. The performance capability of these vehicles is then determined for various modifications when they are added to provide selected research configuration options. The methods and techniques employed are presented in the following sections.

3.4.1 (U) BASIC VEHICLES - The approach employed to size the two basic vehicles consists essentially of matching vehicle fuel volume available with the fuel volume required to meet the design missions described in Section 2.3. The techniques used to accomplish this are the same as those employed during Phase I and Phase II, and discussed in Volume II with one exception. In Phase III, the Mach 12 rocket vehicle acceleration -climb trajectory was computed using the Method of Steepest Descent Trajectory Optimization Program. Prior to this, a closed form trajectory solution, described in Volume II was employed.

(U) FIGURE 3-6
TJ AND TRJ INLET PRESSURE RECOVERY



3.4.2 (U) <u>CONFIGURATION OPTIONS</u> - To provide configuration options applicable to various research needs, modifications to the basic vehicles were investigated. These included various launch modes, propulsion systems, thermal protection systems, armament systems and staging systems (two stage recoverable booster). The effect these various modifications have on the performance of the basic vehicles is determined in terms of test Mach number and test time. Where the research option modifications dictate a change in mission profile trajectory studies were performed to determine appropriate profiles.

3.5 WEIGHT METHODOLOGY

(U) Weight estimation methods are unchanged from those previously reported. However, refined scramjet and convertible scramjet weight estimation methods were used in Phase III and are presented in Section 3.5.1. Structural weight increments for accommodation of the modifications were calculated using the basic estimating equations with allowance for the local structural differences associated with each modification. Weight changes to the various systems, such as electrical, hydraulic, and armament, etc. were not based on a particular set of estimation equations because each modification was unique and limited in its scope. Therefore, each system was analyzed in detail according to the appropriate design parameters. All weight increments for each modification are discussed completely in Sections 4.10 (MACH 12 RESEARCH VEHICLE), and 5.10 (MACH 6 RESEARCH VEHICLE).

3.5.1 (U) SCRAMJET AND CONVERTIBLE WEIGHT ESTIMATION METHODS - Weight estimation equations for the scramjet and convertible scramjet were derived from previous studies conducted in support of the "Hypersonic Scramjet Vehicle Study", Reference (9). In addition to the basic scramjet module weight, two more weight equations were developed to account for the retractable forward ramp structure and its actuation system which is unique to this study. A complete description of the scramjet installation and operation is contained in Section 4.2.

(U) For weight estimation purposes the scramjet structure was divided into the internal compression duct, combustor section and expansion nozzle. Each of these sections have distinct pressure and temperature profiles which directly influence the weight as shown in Figure 3-7.

(U) For a given scramjet size the structural weight is primarily a function of pressure since the structure is regeneratively cooled to 2000°F (1366°K). However, the regenerative cooling panels and plumbing weight are affected by the temperature variation between the sections.

(U) Convertible scramjet weight was estimated similarly, but weight was added for the thrust vector control surfaces and a slightly longer (approximately 15 inches (38.1 cm)) combustor section. Weight increments for the scramjet module extension and retraction system as well as the structural modification to the research vehicle are estimated from the equations previously presented in Volume II.

(U) Scramjet and convertible scramjet weights are derived from the following equations:

γ SJ/CSJ $\quad = \sum_{i=1}^{8} \gamma_i$

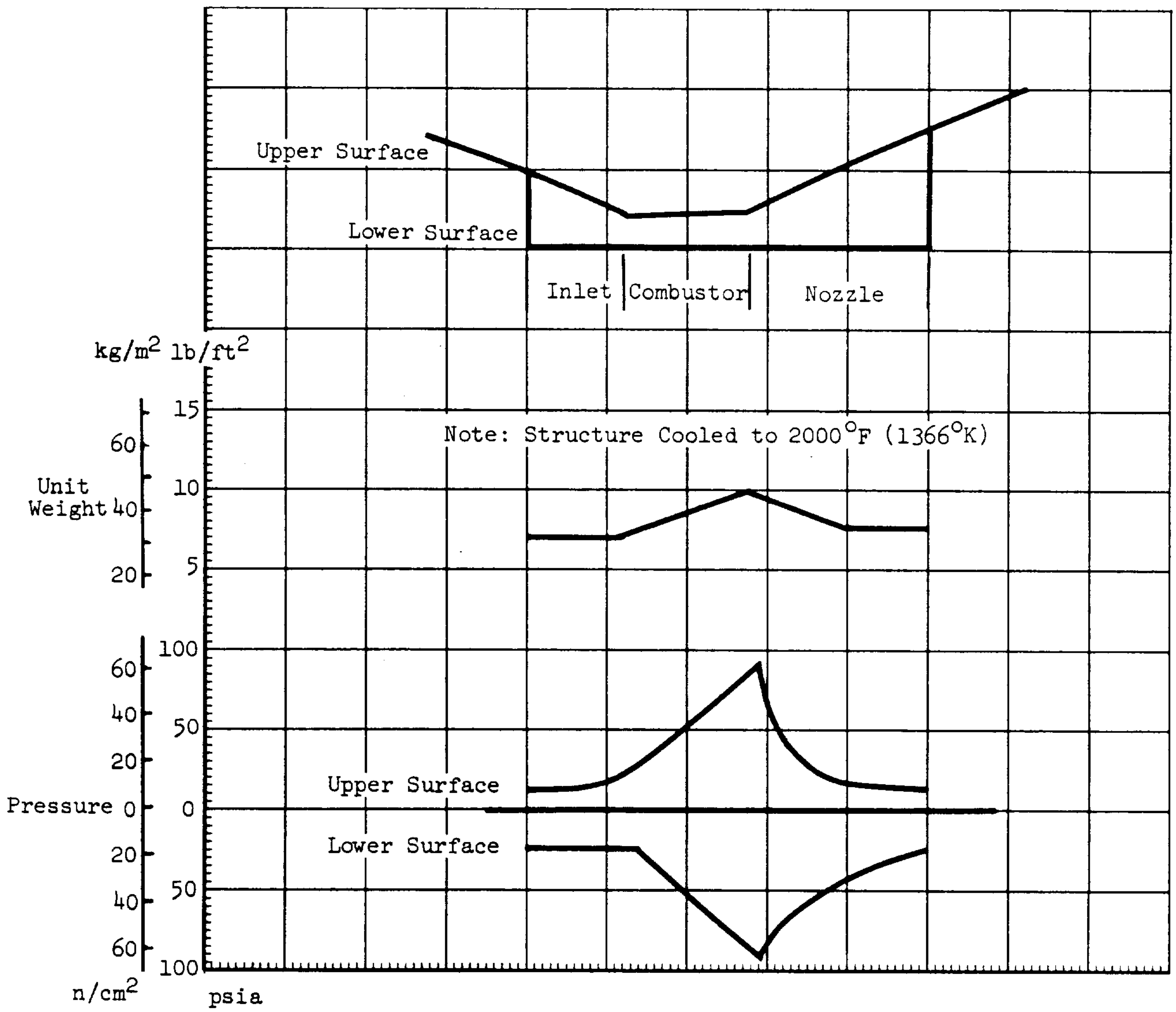
γ_1 Structure $\quad = \gamma_{IC} + \gamma_C \; \gamma_{EN}$

γ_{IC} (Internal Compression Duct) $\quad = K\,(S_{IC})\,(P)^n\,(N)$

γ_C (Combustor) $\quad = K\,(S_C)\,(P)^n\,(N)$

γ_{EN} (Expansion Nozzle) $\quad = K\,(S_{EN})\,(P)^n\,(N)$

(U) FIGURE 3-7
TYPICAL SCRAMJET WEIGHT DISTRIBUTION



Y_2 = Thrust Vector Control Surfaces = $K (S_D) (N)$

Y_3 = Lip and Trailing Edge = $K (L_L + L_{TE})$

Y_4 = Fuel Injectors = $30 (N)$

Y_5 = Fuel System Cooling Mechanism = $K (S_T) (N)$

Y_6 = Actuated External Ramps = $K (S_R) (P)^n (N) (K_T)$

Y_7 = Ramp Actuation = $K (Y_6)$

Y_8 = Ramp Regenerative Cooling = $K (S_R)$

where:

K = Constant, Correlation (these values differ for each equation)

K_T = Material Constant for Temperature and Failure Mode Distribution

L_L = Length Lip

L_{TE} = Length Trailing Edge

N = Number of Modules

n = Pressure Exponent

P = Pressure, Maximum (Ultimate)

S_C = Surface Area, Combustor Section per Module

S_D = Surface Area, Thrust Vector Control Surfaces

S_{IC} = Surface Area, Internal Compression Section per Module

S_{EN} = Surface Area, Expansion Nozzle Section per Module

S_R = Surface Area, External Ramp, Total

S_T = Total Surface Area per Module = $S_{IC} + S_C + S_{EN}$

Y = Estimated Weight

3.6 COST METHODOLOGY

(U) The methodology employed in the development of the total system costs for both the Mach 6 and Mach 12 aircraft is essentially the same as that employed in Phase II with the exception of the development of the flight vehicle maintenance/repair costs which have been modified.

(U) In addition to developing the costs for the basic aircraft, the incremental costs of the various research options were also derived using the same costing methods. To provide visibility into the cost effects of each of the research options, two cost increments were determined. These were: (1) the cost of the basic vehicle resulting from incorporating the structural provisions to accept the research package in the initial basic vehicle design and (2) the cost of developing and incorporating the research package as a field modification into the basic vehicle. Furthermore, the cost estimating parameters for the scramjet and convertible scramjet engine have been significantly modified.

(U) These costing refinements are presented in the following sections along with a recap of the ground rules and assumptions used in the cost analysis.

3.6.1 (U) GROUND RULES AND ASSUMPTIONS - The following ground rules and assumptions define the framework within which the cost is determined. The basis for their selection is two fold, namely, (1) to effect economy and (2) to establish a basis for deriving the acquisition and operating costs. The majority of the ground rules and assumptions appear reasonable and feasible within the scope of past flight research programs.

(U) Ground Rules - Ground rules are those criteria specified in the work statement and implied in the cost philosophy which were selected to develop the flight research vehicle program costs. The ground rules used in Phase III are essentially the same as those employed in Phase II and are as follows:

- Minimum cost-to-fly program (experimental shop approach similar to ASSET program)
- Soft tooling
- Static and fatigue testing limited to element tests rather than full scale models
- Limited reliability program
- "Zero Defects" program not employed
- Maximum use of existing equipment
- Maximum use of existing facilities
- Three flight research test vehicles in the program (similar to the X-15)
- A separate flight hardware airframe is provided for structural testing

- o Seven spare engine ship sets are allocated for configurations requiring only one engine for both acceleration and cruise (for a total of 10 engines)
- o Five spare engine ship sets are allocated for configurations requiring multiple acceleration and cruise engines (for a total of 40 engines)
- o Five year operational test program
- o The flight research program consists of 200 flights (similar to X-15 program)
- o Limited pre-delivery flight test program
- o All the research packages are installed at Edwards AFB
- o All costs are in 1970 dollars and include allowance for prime contractor earnings of 10 percent

(U) Assumptions - Assumptions are defined as those criteria selected to aid in the development of the flight research vehicle program costs. The following assumptions were used in the development of the cost elements:

- o The C-5A aircraft will be the carrier aircraft for the air launched configurations
- o Air launched configurations will be transported externally (pod) by the C-5A carrier aircraft from the contractor's facility to Edwards AFB.
- o HTO launched configurations will be ferried in the turbojet mode from the contractor's facility to Edwards AFB.
- o Air launched configurations will be transported by the C-5A carrier aircraft from the recovery sites to the launch site (Edwards AFB).
- o HTO launched configurations will be launched from Edwards or Holloman AFB.
- o All air launched configurations will land at Edwards.
- o Two C-5A carrier aircraft will be assigned for transporting and launching the air launched configurations.
- o Maintenance/Repair of the flight research vehicles will be performed at Edwards AFB.
- o Air launched configurations will be launched from Cecil NAS (Naval Air Station) and Eglin AFB for high Mach number tests and from Dyess, Holloman and Perrin AFB for low Mach number tests.
- o All vehicles will use their total propellant including reserves, during each mission.

3.6.2 (U) MAINTENANCE/REPAIR COSTS - A significant change was made in Phase III in the derivation of the flight vehicle maintenance/repair cost, cost element III-3 of the operational cost category. In the previous two phases, maintenance/repair costs were computed as percentages of the flight research vehicle's investment cost. These percentages varied with Mach number and ranged from 1.5 to 2.75% of the flight research vehicle's investment cost per flight. However, in Phase III, a more detailed approach was used to derive maintenance/repair costs. A refined approach was used in Phase III in which the maintenance/repair costs were developed for each of the four basic systems: namely, (1) airframe, (2) propulsion, (3) miscellaneous subsystems, and (4) avionics. Costs were developed to reflect both material and labor costs and are summarized in Figure 3-8.

(U) Maintenance and repair costs were significantly reduced in Phase III due to the following criteria:

(1) No modifications will be performed on the basic vehicle other than those planned for

(2) Well managed program.

(U) The basis for the methods were obtained from two major sources:

- MCAIR engineering estimates of spare requirements for the airframe components, miscellaneous subsystems, and avionics shown in Figures 3-8 and 3-9.

- X-15 data presented in Figures 3-10 and 3-11 obtained from References (10) and (11).

(U) Mach 6 & 12 Vehicles - The sequence of steps employed in the development of the maintenance/repair costs for the Mach 6 & 12 vehicles are presented in Figures 3-12 and 3-13 and are discussed below.

(U) Step 1: Airframe Material - Airframe material costs for items 2 and 3 shown in Figure 3-8 for the Mach 12 vehicle were derived on the basis of the number of spares required, as obtained from Figure 3-9 and the cost per square unit shown in Figure 3-3. For example, the number of sets of columbium alloy shingles, item 2A(a) in Figure 3-8, was based on the number of flights one set of shingles would be expected to survive before replacement, which is 34 (shown in last column in Figure 3-9). Since each vehicle is scheduled to fly 67 missions, one set of columbium alloy shingles is required per vehicle. The maintenance/repair material cost shown in Figure 3-8 was developed from the following equation:

$$.20 \times \Sigma \text{ Airframe Investment Cost}$$

(U) The Mach 6 vehicle does not have any shingles or leading edge surfaces that have to be replaced prior to 67 flights, which is the number of flights allocated per vehicle.

(U) FIGURE 3-8
MAINTENANCE/REPAIR COST SUMMARY 200 TOTAL FLIGHTS -3 VEHICLES

MAJOR SUBSYSTEMS	CONFIGURATIONS C233-1	CONFIGURATIONS C210-1	BASIS
	(THOUSANDS OF DOLLARS)		
I COMPONENT AND MATERIAL COST			
A AIRFRAME MATERIAL			
(1) Maintenance & Repair	15,936	25,778	
(2) Thermal Protection			
(a) Shingles			
Columbium Alloy	908	-	3 Sets Required for the C233-1 Configuration $500/lb x 605 lb x 3 Sets = $907,500
Tantalum Alloy	432	-	9 Sets Required for the C233-1 Configuration $600/lb x 80 lb x 9 Sets = $432,000
Subtotal	1,340		
(3) Control Surfaces			
(a) Columbium Alloy	414	-	3 Sets Required for the C233-1 Configuration $500/lb x 276 lb x 3 Sets = $414,000
(4) Leading Edges	711	-	3 Sets Required for the C233-1 Configuration 3 Sets x $500/lb 474 lb = $711,000
(5) Expendables			
(a) Landing Skids	1,300	-	197 Units Required for the C233-1 Configuration 197 Units x $75/lb x 88 lb = $1,300,200
Total Airframe Material	19,701	25,778	
B COMPONENT AND MATERIAL COST			
(1) Engine Spare Components	8,375	9,104	$41,880/Flight for C233-1 Configuration $45,520/Flight for C210-1 Configuration
(2) Miscellaneous Subsystems			C233-1 — C210-1
(a) Fuel System	700	903	2 Required, Unit Price - $350,000 $451,560
(b) APU	3,600	4,644	9 Required, Unit Price - 400,000 516,000
(c) Engine Controls	57	74	1 Set Required, Set Price - 57,000 74,000
(d) Instruments	975	1,257	3 Sets Required, Set Price - 325,000 419,000
(e) Hydraulics	450	580	3 Sets Required, Set Price - 150,000 193,300
(f) Electrical	150	194	3 Sets Required, Set Price - 50,000 64,670
(g) ECS	300	387	3 Sets Required, Set Price - 100,000 129,000
Total	6,232	8,039	(Quantities Obtained from Internal Estimates)
(3) Avionics	2,892	2,892	2 Sets Required, Set Price = $1,446,000
	1,807	1,807	5 Sets x 25% x $1,446,000/Set = $1,807,000
Total	4,699	4,699	
(4) Payload	750	645	3 Sets x $500,000/Set x 50% = $750,000 (C233) 3 Sets x $430,000/Set x 50% = $245,000 (C210)
Grand Total Matl.Cost	39,757(1)	48,265(2)	
II LABOR COST			
A AIRFRAME	9,855	9,450	
B PROPULSION	2,025	2,700	
C MISC. SUBSYSTEMS	675	405	
D AVIONICS	945	945	
Grand Total Labor	13,500(3)	13,500(4)	
GRAND TOTAL LABOR & MATERIAL COST	53,257	61,765	
REFURBISHMENT COST PER FLIGHT	266	309	
FLIGHT VEHICLES INVESTMENT COST	31,738	48,258	
% OF FLIGHT VEHICLE INVESTMENT COST PER FLIGHT	0.84%	0.64%	

(1) 75% of Total Maintenance/Repair Cost
(2) 78% of Total Maintenance/Repair Cost
(3) 25% of Total Maintenance/Repair Cost
(4) 22% of Total Maintenance/Repair Cost

(U) FIGURE 3-9
HYFAC AIRFRAME MAINTENANCE/REPAIR REQUIREMENTS

STRUCTURAL ELEMENT	CONSTRUCTION	RELATED EXPERIENCE LAB COUPONS & PANELS	HARDWARE EXPERIENCE	PROGRAM COSTING BASIS NO. OF FLIGHTS
SHINGLES	.010 TYPE GAGES			
TITANIUM ALLOY	HONEYCOMB-WELD	AFFDL-TR-68-130	F-4, GEMINI	67
	SINGLE-FACED CORRUGATED SANDWICH			
RENE '41	GAS TUNGSTEN ARC WELD	-	GEMINI, ENGINE COMPONENTS	67
T.D. NiCr	RESISTANCE WELD	30 CYCLES @ 2200°F (MDAC) AFFDL-TR-68-292	FIN AND RUDDER PROGRAM (MCAIR)	67
COLUMBIUM ALLOY	ELECTRON BEAM WELD	AFFDL-TR-68-210	ASSET, BGRV	34
		MAC-RPT G143 (100 CYCLES 2400°F)	NONE	
TANTALUM ALLOY	ELECTRON BEAM WELD	AF 33 (615)-3935, BPSN 6(6313608-62405364)		17
CONTROL SURFACE	.030 TYPE GAGE			
T.D. NiCr	SKIN STRINGER (MECHANICAL)	SAME		67
COLUMBIUM ALLOY	SKIN STRINGER (WELDED)	SAME		34
NOSE				
RENE '41	COOLED TO 1500°F	SAME		67
LEADING EDGE	1" D.A. - .020 GAGE			
T.D. NiCr	SKIN STRINGER	SAME		67
COLUMBIUM ALLOY	SKIN STRINGER	SAME		34
LANDING SKIDS				
ABLATIVE COAT	COATED STEEL		GEMINI	1

(U) FIGURE 3-10
X-15 REFURBISHMENT COST
(Millions of Dollars)

	1964	1965	Total	Avg.
Airframe				
Airframe Spares	1.334	0.870	2.204	1.102
Contractor Airframe Parts Repair	1.287	1.030	2.317	1.159
Airframe Engr. Support, Contractor	2.482	2.929	5.411	2.706
Shop Fabrication, Contractor	0.011	0.041	0.052	0.026
Shop Support, 3 Men, NASA	0.030	0.038	0.068	0.034
Engineering Support, NASA	0.026	0.040	0.066	0.033
Miscellaneous	0.060	0.060	0.120	0.060
Total	5.230	5.008	10.238	5.120
Propulsion System				
YLR99 Engine Maintenance, USAF	0.200	0.200	0.400	0.200
Engine Component Spares	0.400	0.921	1.321	0.661
Engine Engr. Support, Contractor	1.000	0.092	1.092	0.546
Shop Support, NASA	.030	0.025	0.055	0.027
Engineering Support, NASA	.013	0.020	0.033	0.017
Miscellaneous	.030	0.030	0.060	0.030
Total	1.673	1.288	2.961	1.481
Avionics				
Instrumentation Maint., Contractor	0.040	0.026	0.066	0.033
Inertial Flight Data System Sup.	0.220	0.220	0.440	0.220
Shop Support, NASA	0.010	0.012	0.022	0.011
Engineering Support	0.013	0.020	0.033	0.016
Miscellaneous	0.010	0.010	0.020	0.010
Total	0.293	0.288	0.581	0.290
Miscellaneous Subsystems				
APU Unit Overhaul	0.092	0.133	0.225	0.1125
Materials	--	0.001	0.001	0.0005
Total	0.092	0.134	0.226	0.1130
Grand Total	7.288	6.718	14.006	7.004
Avg. Per Flight	.270	.210		

(U) FIGURE 3-11
X-15 REFURBISHMENT COST

	1964 Cost M$	1964 % of Total	1965 Cost M$	1965 % of Total
Airframe				
1) Labor	2.585	49.43	3.084	61.58
2) Materials	2.645	50.57	1.924	38.42
Total	5.230	100.00	5.008	100.00
Engine				
1) Labor	1.258	75.19	.355	27.56
2) Materials	.415	24.81	.933	72.44
Total	1.673	100.00	1.288	100.00
Avionics				
1) Labor	.293	100.00	.288	100.00
2) Materials	--	--	--	--
Total	.293	100.00	.288	100.00
Miscellaneous Subsystems				
1) Labor	.092	100.00	.133	99.25
2) Materials	--	--	.001	.75
Total	.092	100.00	.134	100.00
Total Labor	4.228	58.01	3.860	57.45
Total Material	3.060	41.99	2.858	42.55
Total	7.288	100.00	6.718	100.00

Total Successful Flights - 59 (1964 and 1965)

Average Labor Cost Per Flight - .137 $\overline{M}$ Dollars
Average Material Cost Per Flight - .100 $\overline{M}$ Dollars
Average Labor and Material Cost Per Flight - .237 $\overline{M}$ Dollars

(U) FIGURE 3-12
ESTIMATING TECHNIQUE EMPLOYED IN THE DERIVATION OF THE MACH 12 VEHICLE MAINTENANCE/REPAIR COST

ITEM NO.	ITEM	METHOD EMPLOYED	BASIS
1.	Materials Airframe a) Shingles, Control Surfaces, Nose Cone, Leading Edges and Landing Skids	Cost of Spares Required for These Components	MCAIR Engineering Estimates (Fig. 3-8 & 3-9)
	b) Maintenance/Replacement Material	CER 20% of Total airborne investment cost based on 200 flight program	MCAIR Engineering Estimate.
2.	Engine	CER .002276 x Total Eng. Inv. Cost	X-15 Data shown in Fig. 3-11.
3.	Miscellaneous Subsystems	Cost of Spares Required for These Components	MCAIR Engineering Estimates (Fig.3-8)
4.	Avionics (Includes Payload)	Cost of Spares Required for These Components	MCAIR Engineering Estimates (Fig.3-8)
5.	Total Material	Material Costs for Items 1 thru 4	
6.	Total Labor Cost	CER (150 men x $18,000/man-year x 5 years)	MCAIR Engineering Estimate and X-15 data.
7.	Labor Airframe	CER 73% of Total manpower allocated for maint./repair x $18,000 per man-year x 5 years	MCAIR Engineering Estimate and X-15 data.
8.	Engine	CER 15% of Total manpower allocated for maint./repair x $18,000 per man-year x 5 years	MCAIR Engineering Estimate and X-15 data.
9.	Miscellaneous Subsystem	CER 5% of Total manpower allocated for maint./repair x $18,000 per man-year x 5 years	MCAIR Engineering Estimate and X-15 data.
10.	Avionics and Payload	CER 7% of Total manpower allocated for maint./repair x $18,000 per man-year x 5 years	MCAIR Engineering Estimate and X-15 data.

(U) FIGURE 3-13
ESTIMATING TECHNIQUE EMPLOYED IN THE DERIVATION OF THE MACH 6 VEHICLE MAINTENANCE/REPAIR COST

ITEM NO.	ITEM	METHOD EMPLOYED	BASIS
1.	Materials Airframe a) Shingles, Control Surface, Nose Cone, Leading Edges and Landing Skids	No requirement for these components	
	b) Maintenance/Replacement Material	CER 20% of Total Airborne Investment Cost Based on 200 Flight Program	MCAIR Engineering Estimate
2.	Engine	CER .002276 x Total Eng. Inv. Cost	X-15 Data shown in Fig. 3-11
3.	Miscellaneous Subsystems	Cost of Spares required for these components	MCAIR Engineering Estimate (Fig. 3-8)
4.	Avionics (Includes Payload)	Cost of Spares required for these components	MCAIR Engineering Estimate (Fig. 3-8)
5.	Total Material	Material costs for Items 1 thru 4	
6.	Total Labor Cost	CER (150 men x $18,000 per man-year x 5 years)	MCAIR Engineering Estimate and X-15 data.
7.	Airframe	CER 70% of total manpower allocated for maint./repair x $18,000 per man-year x 5 years	MCAIR Engineering Estimate and X-15 data
8.	Engine	CER 20% of total manpower allocated for maint./repair x $18,000 per man-year x 5 years	MCAIR Engineering Estimate and X-15 data.
9.	Miscellaneous Subsystems	CER 3% of total manpower allocated for maint./repair x $18,000 per man-year x 5 years	MCAIR Engineering Estimate and X-15 data.
10.	Avionics and Payload	CER 7% of total manpower allocated for maint./repair x $18,000 per man-year x 5 years	MCAIR Engineering Estimate and X-15 data.

(U) The item designated "maintenance and repair material" in Figure 3-8 includes the material used for minor modifications, major and minor structural repairs and scheduled and unscheduled maintenance and constitutes the largest material dollar value of the total material cost.

(U) Step 2: Engine Material - Engine material component costs were derived from the following equation:

.002276 x Total Engine Investment Cost

where: .002276 is the ratio of the engine material cost per flight to the total engine investment cost based on the X-15 aircraft program.

The total engine investment cost of the Mach 12 vehicle is $18,400,000, while it is $20,000,000 for the Mach 6 vehicle.

(U) Engine component material costs were derived on a cost per flight basis using the X-15 background data presented in Figure 3-11. A total of 1.348 M dollars was spent on engine components for the X-15 program for the years 1964 and 1965. Ammortizing the total engine component cost of 1.348 M dollars over the number of successful flights (59), the cost per flight of $22,847 was obtained. The cost per flight was then divided by the investment cost of the X-15 engine ($10,040,000) to obtain the fraction of the engine investment cost spent for material components per flight which was .002276. This fraction was then multiplied by the engine investment costs of the Mach 6 and 12 vehicles ($20,000,000 and $18,400,000) to obtain the material component costs per flight of $45,520 and $41,880 shown in Figure 3-8.

(U) Step 3: Miscellaneous Subsystems Material - Miscellaneous subsystem component material costs were derived from the spares requirements and their associated costs presented in Figure 3-8.

(U) Step 4: Avionics and Payload Material - The material costs for the avionics and payload systems were derived from the spares requirements and their associated costs presented in Figure 3-8.

(U) The quantities of airframe, avionics and miscellaneous subsystems components required were obtained from internal and external sources based on F-4, ASSET, BGRV, Gemini, Mercury, and X-15 data.

(U) Step 5: Total Material - Sum of material costs for the four basic systems.

(U) Step 6: Labor Cost - The total labor cost for the Mach 6 and 12 vehicles was obtained by multiplying the number of NASA operations and maintenance personnel estimated to be required (150 men) by the average annual salary per man ($18,000) by 5 years. A value of 13.5 M dollars was obtained for the total labor cost.

(U) Step 7: Airframe Labor Cost - The labor costs for the Mach 6 and 12 airframes were obtained by multiplying the total labor cost by 70% and 73% respectively. A larger percentage labor factor was allocated to the Mach 12 vehicle than for the Mach 6 vehicle because of the greater amount of wear associated with the Mach 12 vehicle. These percentage values were based on engineering estimates using X-15 data shown in Figure 3-11 as a background source.

(U) Step 8: Engine Labor Cost - The labor costs apportioned to the Mach 6 and 12 engines were obtained by multiplying the total labor costs by 20% and 15% respectively. The percentages used were based on engineering estimates. Since the turbo-ramjet engine used in the Mach 6 vehicle is more complex than the rocket engines employed in the Mach 12 vehicle, a larger percentage labor factor was allocated for the Mach 6 vehicle than for the Mach 12 vehicle.

(U) Step 9: Miscellaneous Subsystem Labor - Labor costs attributed to the miscellaneous subsystems for the Mach 6 and 12 vehicles were obtained by multiplying the total labor cost by 3% and 5% respectively. The Mach 12 vehicle was allocated the larger percentage for the same reason as was given in Step 7. Both percentage labor factors were based on engineering estimates.

(U) Step 10: Avionics and Payload Labor Costs - A labor percentage factor of 7% was used for both the Mach 6 and 12 vehicles. This factor was based on engineering estimates and was multiplied by the total labor cost to obtain the labor cost associated with the miscellaneous subsystems.

3.6.3 (U) INSTALLATION PROVISIONS AND RESEARCH PACKAGES COSTS - The costs for installation provisions and research packages were derived separately. Installation provision costs were derived manually while the research package costs were derived by the same computer program used to derive the basic vehicle costs. The differences in Defense Contractors Progress Report (DCPR) weight involved in the basic airframe structure to provide the installation provisions were small; hence, it was found to be more expeditious to compute these costs manually.

(U) The DCPR weights for the installation provisions were derived from the respective weight statements. For a particular DCPR weight, a weighted material density factor, a weighted production complexity factor, and a tooling complexity factor were derived using the same parameters employed in the Phase II analysis. The installation provisions DCPR weight was added to the DCPR weight of the basic vehicle and the cost elements were derived in the following manner:

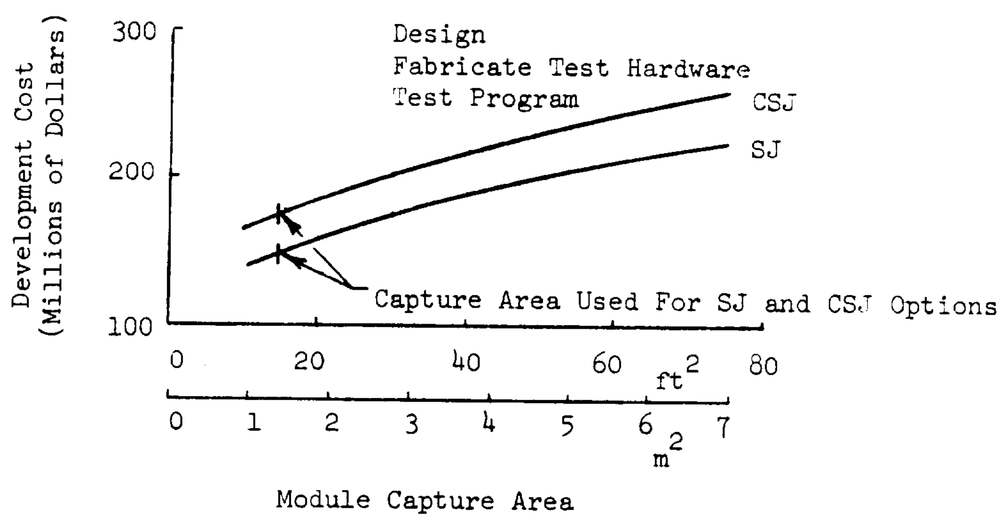
(U) Engineering - The engineering manhours for the new total DCPR weight were derived using the Phase II parameters. From these manhours values were subtracted the manhours for the basic vehicle, leaving the manhours for the added weight only. The manhours for Airframe Engineering were then adjusted by the weighted material density, for the added weight only, as was done in Phase II and the total manhours (Airframe and Subsystems) were converted to the price level using the Phase II parameters.

(U) Tooling - The Tooling manhours for the new total DCPR weight were derived using the Phase II parameters. From these manhour values were subtracted the manhours for the basic vehicle, leaving the manhours for the added weight only. These manhours were then adjusted for the Tooling complexity introduced by the Advanced Materials, which were derived for the added weight only, using the Phase II parameters. They were then converted to the price level using the Phase II parameters.

(U) Production - The Production manhours for the new total DCPR weight were derived using the Phase II parameters. From these manhours values were subtracted the manhours for the basic vehicle, leaving the manhours for the added weight only. The manhours for Airframe Production were then adjusted by the weighted production complexity.

3.6.4 (U) SCRAMJET AND CONVERTIBLE SCRAMJET ENGINE COSTS - The scramjet and convertible scramjet engine development costs were changed significantly in Phase III from those employed in Phases I and II. In Phase III, 150 $\overline{M}$ dollars was used for the scramjet development cost while 175 $\overline{M}$ dollars was employed for the convertible scramjet development cost. These costs are for an engine module capture area of 14.3 ft^2 (1.33 m^2) as shown in Figure 3-14. A breakdown of these development costs is shown in Figure 3-15 for the following elements: (1) engineering design, (2) fabrication of test hardware and tooling, and (3) test program. The development cost of the SJ engine was reduced 25 $\overline{M}$ dollars from that of the CSJ engine to account for the more limited Mach number operating range. These costs are derived from the AiResearch data presented in Section 4 of Volume V. The cost of ground test facilities for these two engines is also shown in Figure 3-15 (147 $\overline{M}$ dollars). This cost is based on the E9 facility described in Volume IV, Part 2.

(U) FIGURE 3-14
SJ AND CSJ DEVELOPMENT COSTS



(U) Investment costs for the SJ and CSJ engines are shown in Figure 3-16. The investment cost for the CSJ was obtained from the AiResearch data precented in Section 4 of Volume V. The SJ investment costs are from the Marquardt hardpoint data presented in Section 4 of Volume V.

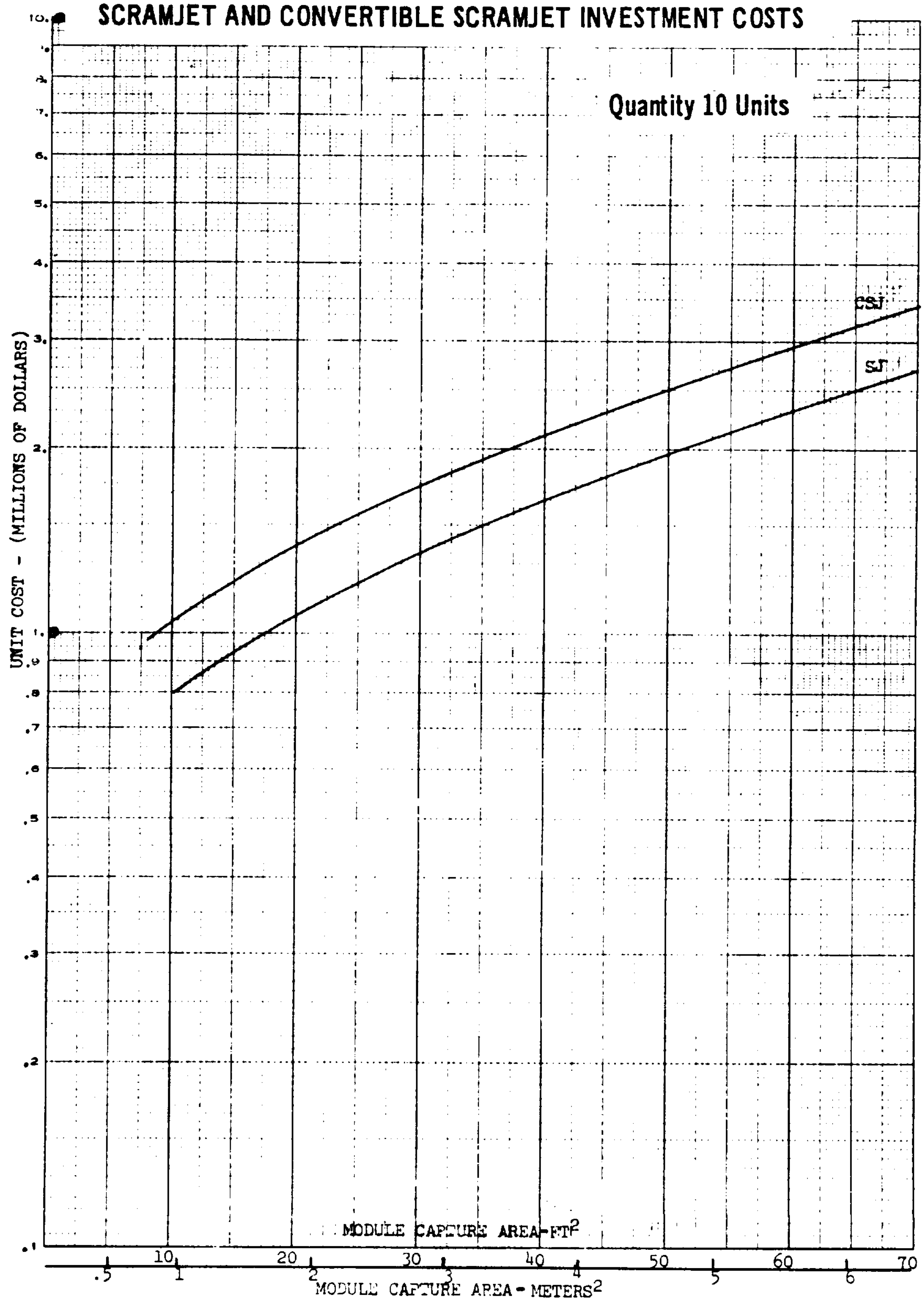
(U) FIGURE 3-15
SCRAMJET AND CONVERTIBLE SCRAMJET DEVELOPMENT COSTS
(Millions of 1970 Dollars)(1)

	SJ	CSJ
Design	50	60
Test Hardware	55	65
Test Support	45	50
TOTAL	150 (2)	175 (2)
Facilities	147	147

(1) Through PFRT

(2) A_c = 14.3 ft^2 (1.33 m^2)

(U) FIGURE 3-16
SCRAMJET AND CONVERTIBLE SCRAMJET INVESTMENT COSTS



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4. MACH 12 VEHICLE SYNTHESIS

(U) The basic Mach 12 air launched rocket vehicle is an all body shape and is designed to provide a five minute steady state cruise capability at Mach 12. This vehicle is capable of conducting a broad scope of research. To expand the research capability of the vehicle, the adaptability of various additional research options by modification of the basic vehicle have been examined. The feasibility of initially designing the vehicle to contain structural provisions and thus accept such research options has been confirmed. A vehicle initially designed in this manner will provide the capability to obtain valuable test data in many interesting areas of research.

(U) The vehicle configuration and general arrangement is illustrated in Figure 4-1. A summary of the vehicle capability and costs is given in Figure 4-2. The basis for these data is developed in the following sections.

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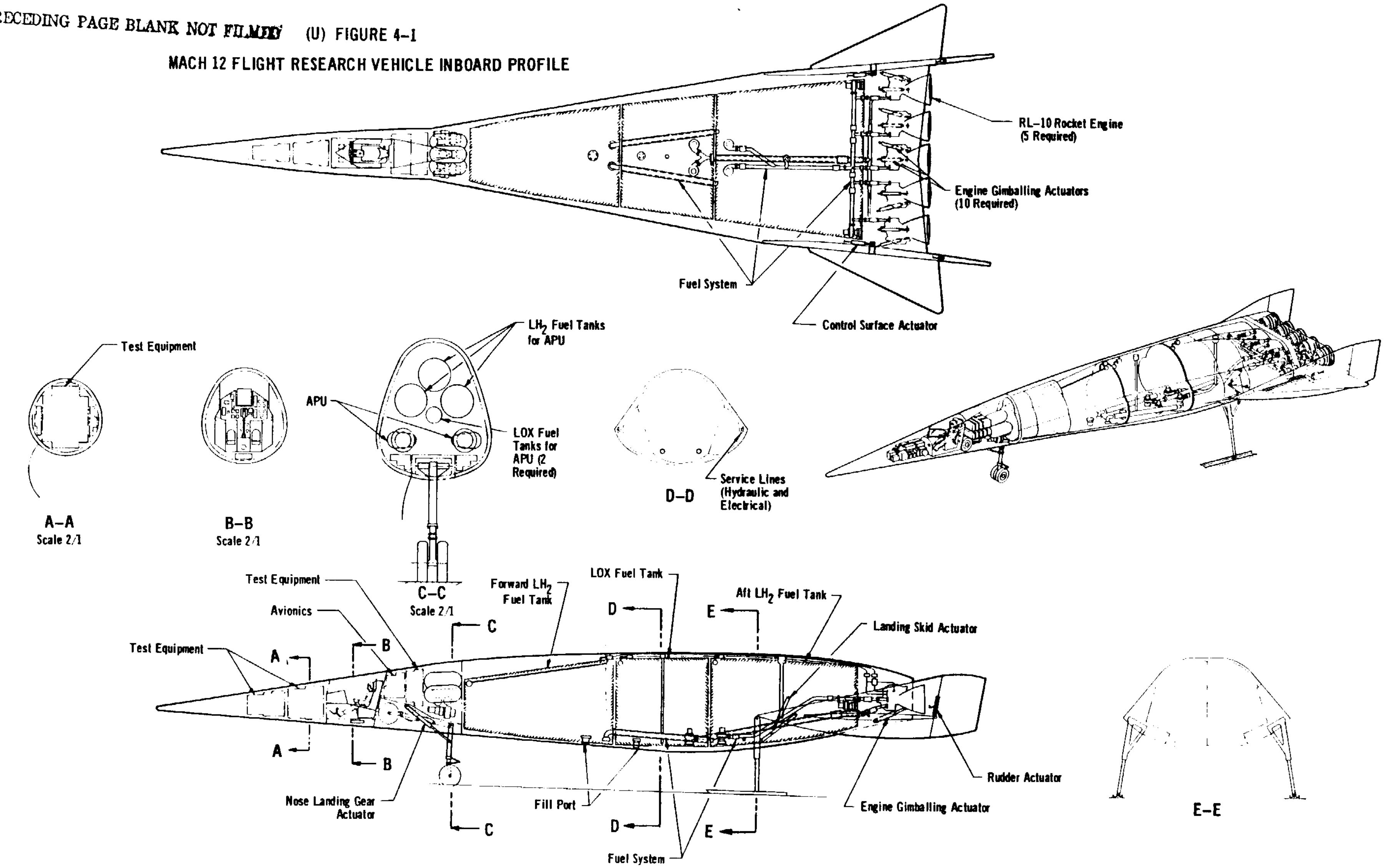
(U) FIGURE 4-1

MACH 12 FLIGHT RESEARCH VEHICLE INBOARD PROFILE



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(U) FIGURE 4-2
CHARACTERISTICS SUMMARY – MACH 12 MANNED, AIRLAUNCHED, ROCKET VEHICLE
Fixed Size

Configuration Description	Performance		Weight – Lb (Kg)		Acquisition Cost (Million Dollars)
	Mach	Time (Minutes)	OWE	TOGW	
Basic Vehicle	12	5	23,340 (10,584)	79,650 (36,129)	262
Scramjet Option	11.5 10.7	0 > 5	29,309 (13,294)	69,200 (31,389)	313[5]
Convertible Scramjet Option	11.8 11.1	0 > 5	30,798 (13,970)	52,300 (23,703)	320[6]
TPS Option	12	5	23,613[1] (10,710)	79,923[1] (36,252)	268
Armament Option	10.3	5	28,038[2] (12,710)	84,348[2] (38,300)	272[7]
Staging Option	Varies with Weight of Staged Vehicle	-	24,538[3] (11,130)	80,848[3] (36,672)	276
HTO Option	9.4 7.6	0 > 5	23,423 (10,624)	79,733 (36,166)	273
VTO Option	8.7 6.9	0 > 5	23,682 (10,742)	79,992 (36,284)	277
Turbojet Option	0.8	Extended	42,305 (19,189)	52,305[4] (23,725)	294

1 Includes 200 Lb (91 Kg) Allowance for TPS Research Package
2 Includes 3300 Lb (1497 Kg) Allowance for Missiles
3 Excludes Weight of Staged Vehicle
4 Includes 10,000 Lb (4536 Kg) Allowance for JP Fuel
5 Excludes $150 MIL Engine and $147 MIL Ground Facility
6 Excludes $175 MIL Engine and $147 MIL Ground Facility
7 Excludes Missile Cost
Note: Ramjet Option – Design Concept Only, Was Developed

4.1 DESIGN REQUIREMENTS

(U) As a result of the Phase II trade studies, some revisions were made in the design criteria and ground rules as noted in Figure 4-3. During the Phase II trade studies, the baseline Mach 12 aircraft structure was protected from the thermal environment by a passive insulation system. However, the results of the thermal protection system (TPS) trade study showed a significant gain in aircraft performance using a water wick cooling system in the design. Therefore, for the Phase III refinements, the basic Mach 12 aircraft incorporated the water wick TPS. Other additions to complete the Phase III ground rules are described in the following paragraphs.

(U) Propulsion System - One result of the Phase II trade study was that using off-the-shelf engines permits a significant reduction in program costs. This result was employed in Phase III, in that near-term engines were used for both the Mach 6 and Mach 12 aircraft.

(U) Fatness Ratio - The study of vehicle "fatness", described in Section 2.4 showed that using a fatness ratio of 0.125 resulted in good structural efficiency and volumetric utilization as well as the best vehicle performance. The Mach 12 aircraft incorporated a 0.125 fatness ratio for all the refinement studies.

(U) Oxidizer/Fuel Ratio - Selection of the O/F ratio of 6:1 used in the Phase III vehicle refinements was described in Section 2.4 and was used for the Mach 12 rocket aircraft.

(U) FIGURE 4-3
PHASE III CRITERIA AND GROUND RULES FOR MACH 12 AIRCRAFT

Payload Weight Payload Density	1500 lb (.68 kg) (1) 20 lb/ft^3 (320 kg/m3) (1)
Design Limit Load Factor (n_z) Taxi Structure (Maneuvering) Thermal Protection Dynamic Pressure	 2.0 g 5.0 g 3.5 g 2500 psf (11.9 N/cm^2)
Convertible Scramjet Inlet Pressure	100 psi (68.9 N/cm^2)
Landing Speed Landing Sink Speed	200 kts (103 m/sec) 20 fps (6.08 m/sec)
LH_2 Tank Pressure LO_2 Tank Pressure	10 psig (6.9 N/cm^2) 25 psig (17.2 N/cm^2)
LH_2 Temperature LO_2 Temperature Ullage, All Body Configuration	30°R (16.65°K Subcooled 163°R (90°K) Normal Boiling Point 2.5%
Tankage Concept	Integral
Thermal Protection External Surfaces Allowable Heat into Fuel	 Active Insulation (Water Wick) (1) 100 Btu/ft^2-hr (31.5 watt/m^2)
Fuel Reserves, I_{sp} Reduction Rocket	 1%
Flight Research Program Time Number of Research Aircraft Test Time Operational Life	5 yrs. (157.8 Msec) 3 5 Minutes (300 seconds) 200 Flights
Crew Compliment Max Angle of Attack, Approach Speed Max Angle of Attack, Flight Speed	1 15° (.262 rad) 20° (.349 rad)

(1) Changes from Phase II ground rules.

4.2 CONFIGURATIONS - BASIC AND VARIATIONS

(U) The fundamental objective of the design studies was to refine the basic vehicle design and to examine the feasibility of building into the basic vehicle structural provisions which would allow a simple adaption of various optional research capabilities. The specific modifications presented are not implied as being the only approach to meet the stated objective, but rather, as being representative of how a modification kit concept could be applied. Descriptions of the various configurations studied and the structural concepts employed, including structural arrangement and materials, are presented in the following paragraphs.

4.2.1 (U) VEHICLE DESIGN - Configuration B233 has been carried over from Phase II of this study as the basic Mach 12 class vehicle and is designated C233 for Phase III studies and refinements.

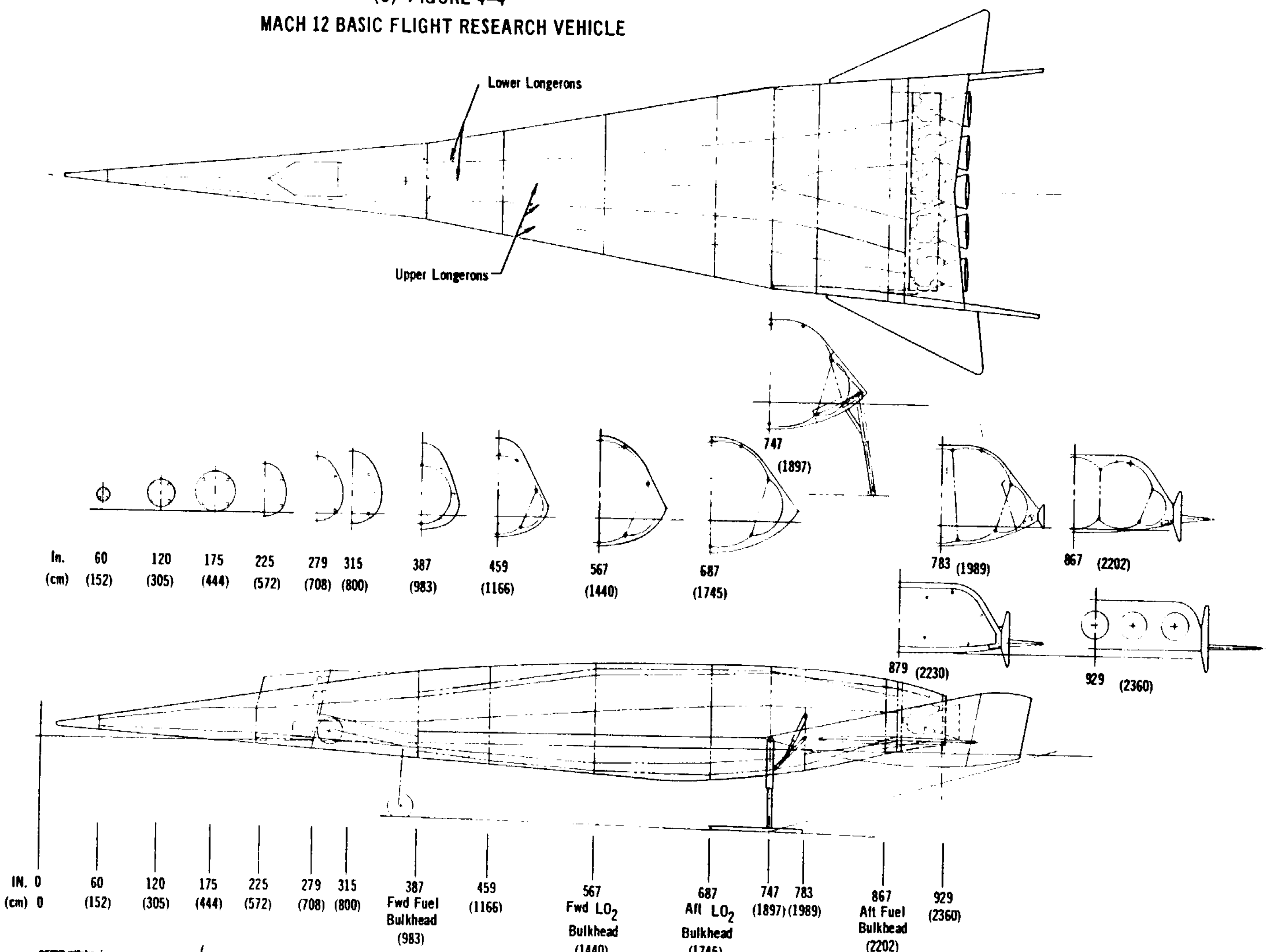
4.2.1.1 (U) Basic Vehicle Design - The basic vehicle is a manned, air launched, all body shaped vehicle powered by five near-term RL10-A-3-9 rockets.

(U) At the beginning of Phase III, parametric studies were continued in order to improve the performance of the basic vehicle. The results of these studies, previously discussed in Section 2.4, showed that improved vehicle performance was obtained by using a fatness ratio of 0.125 and a rocket oxidizer to fuel weight mixture ratio of 6. These features were incorporated in the aircraft and the vehicle was performed and sized to provide a five minute steady state cruise capability at Mach 12. A drawing of the resulting aircraft is shown in Figure 4-4. Volume and wetted area curves for the vehicle are given in Figure 4-5.

(U) The vehicle is air launched from the right wing of a specially equipped C-5A launch aircraft. A basic wing pylon, Figure 4- 6, which picks up hard points on the C-5A wing spars, also picks up three hard points on the upper surface of the vehicle. The alternate pylon shown indicates the changes required to incorporate in the basic pylon design an interchangeable capability for use with the staging configuration option. Both designs maintain the same research aircraft ground clearance when mounted on the C-5A wing.

(U) The basic pylon and basic Mach 12 vehicle are installed on the C-5A wing between the fuselage and the inboard engine nacelle as shown on Figure 4- 7. The close proximity of the research vehicle to the C-5A engine indicates a C-5A modification will be required to carry a vehicle of this size. Two possible modifications are an engine relocation or redesign of the landing gear and fuselage housing. The engine relocation could be accomplished either by moving the engine pylon outboard or by relocating the engine pylon combination to the upper surface of the wing at the original wing station. Figure 4-7 shows the relationship of the vehicle and relocated engines with an unchanged landing gear pod. Either engine relocation approach appears feasible and will provide adequate clearance. The C-5A jet engine wake impingement on the test aircraft is shown for C-5A idle power and takeoff power and was taken from the C-5A Operational Planning Document, MER 400A, published by the Lockheed-Georgia Company, Marietta, Georgia. The C-5A cruise power jet engine wake data at test vehicle launch conditions were not available. However, it is judged that sea level takeoff power jet engine wake is the most severe condition that is encountered during the test vehicle carriage and launching operation.

(U) FIGURE 4-4
MACH 12 BASIC FLIGHT RESEARCH VEHICLE



Physical Characteristics

Geometry:	Wing	Tip	Vertical Tail
Sweep: LE	80^0	65^0	77.8^0 Upr, 76.5^0 Lwr
TE	6.25^0	6.25^0	11^0
C_R in.(M)	761 (19.3)	157 (4.02)	267 (6.78)
C_T in. (M)	200 (5.08)	0 –	56.70 (1.44)
b/2 in. (M)	110 (2.78)	63 (1.58)	73.99 (1.88)
t/c		3%	3%
Overall Length ft (M)	83.3	(26.60)	
Total Span ft (M)	30.3	(9.30)	
Height ft (M)	13.8	(5.03)	
Areas: ft^2 (M^2)			
Planform	813.0	(75.5)	
Wing	684.82	(63.62)	
Tips (Total)	74.0	(6.86)	
Vertical Tail (total)	175.40	(16.33)	
Wetted (Total)	2399.00	(222.87)	
Volumes: Ft^3 (M^3)			
Total	4073.	(115.0)	
Propellant Tank	2575.	(72.9)	
Fuel	1725.3	(48.9)	
Oxidizer	677.3	(19.2)	
Weights: lb(kg)			
TOGWE	79,650	(36,129)	
Propellant			
Fuel	8,040	(3,648)	
Oxidizer	48,270	(21,896)	
OWE	23,340	(10,585)	
Propulsion:			
	Accel.	Cruise	
Engine: Rkt	RL10-A-3-9	RL10-A-3-9	
Thrust (lb)	22,750 (ea)		
(N)	(101,192)		
Weight (lb)	321 each		
(kg)	(145.6)		

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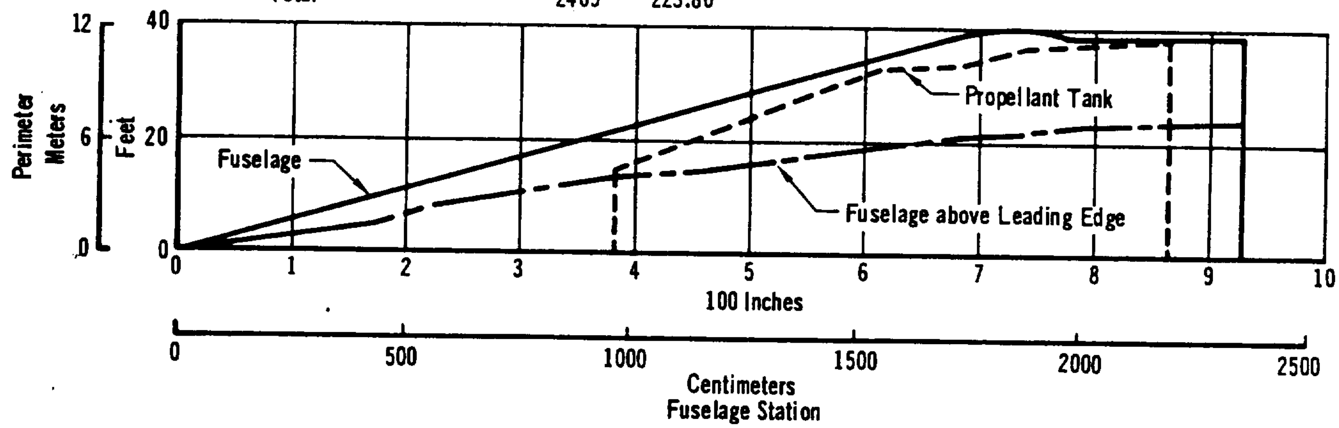
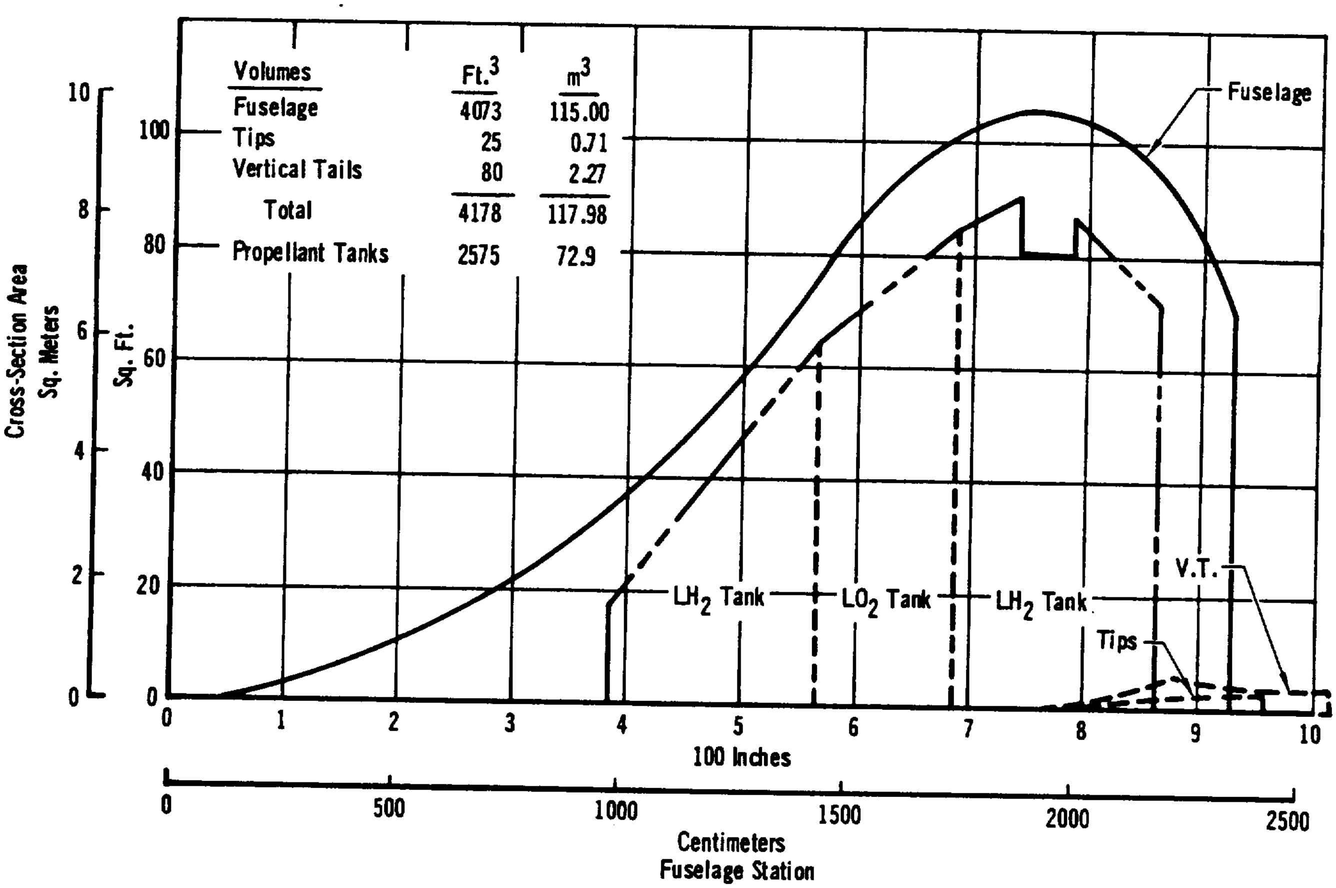
(U) FIGURE 4-5
MACH 12 VEHICLE
Wetted Area and Volume Plot
S_P = 813 Ft^2 (75.5 m^2)

Wetted Areas	Ft^2	m^2
Fuselage	1910	177.40
Tips	148	13.75
Vertical Tails	351	32.65
Total	2409	223.80

	Ft^2	m^2
Fuselage above Leading Edge	1106	102.60
Propellant Tank	1200	111.46





Volumes	Ft.^{3}	m^3
Fuselage	4073	115.00
Tips	25	0.71
Vertical Tails	80	2.27
Total	4178	117.98
Propellant Tanks	2575	72.9

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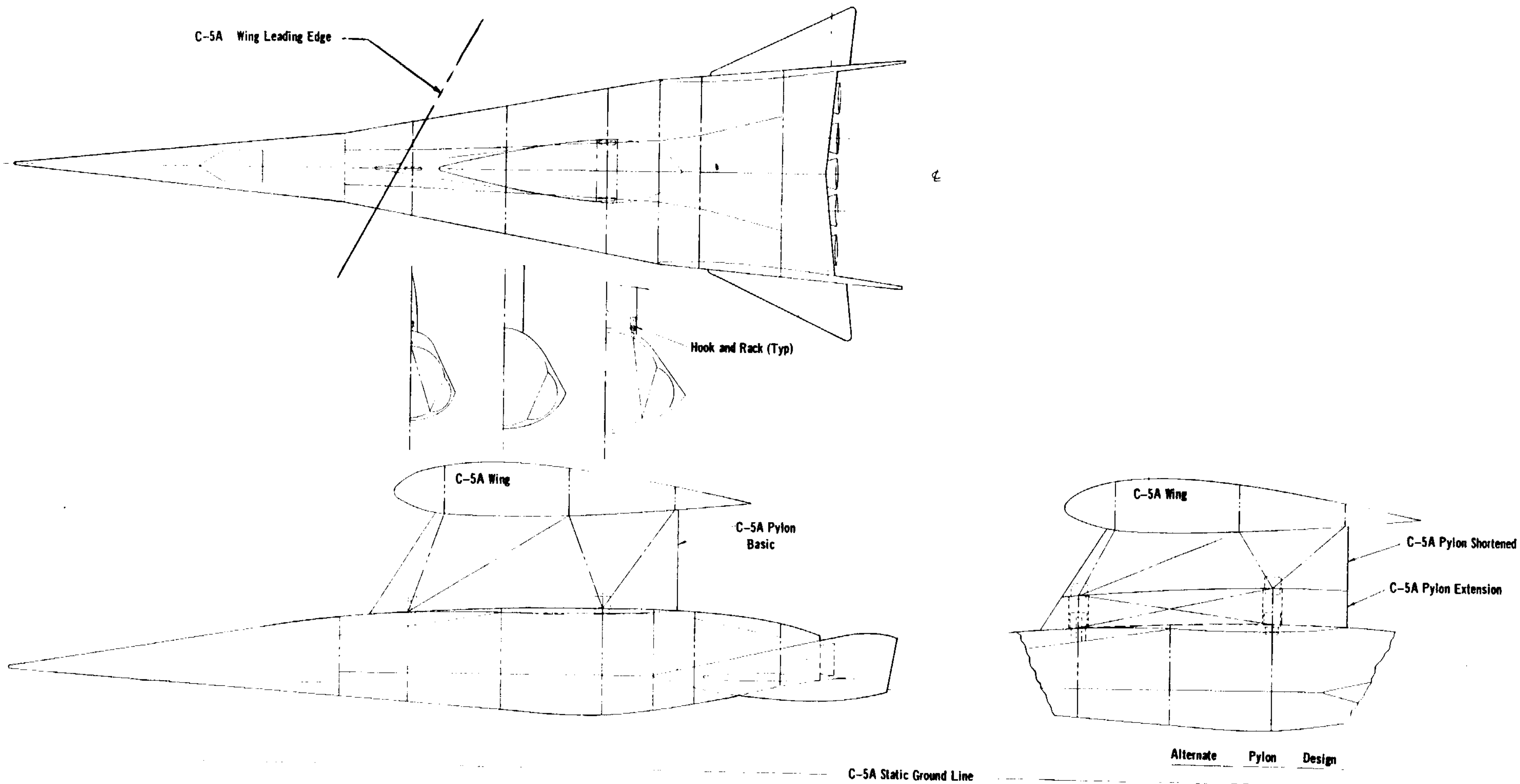
(U) FIGURE 4-6
MACH 12 MOUNTING PYLON FOR C-5A



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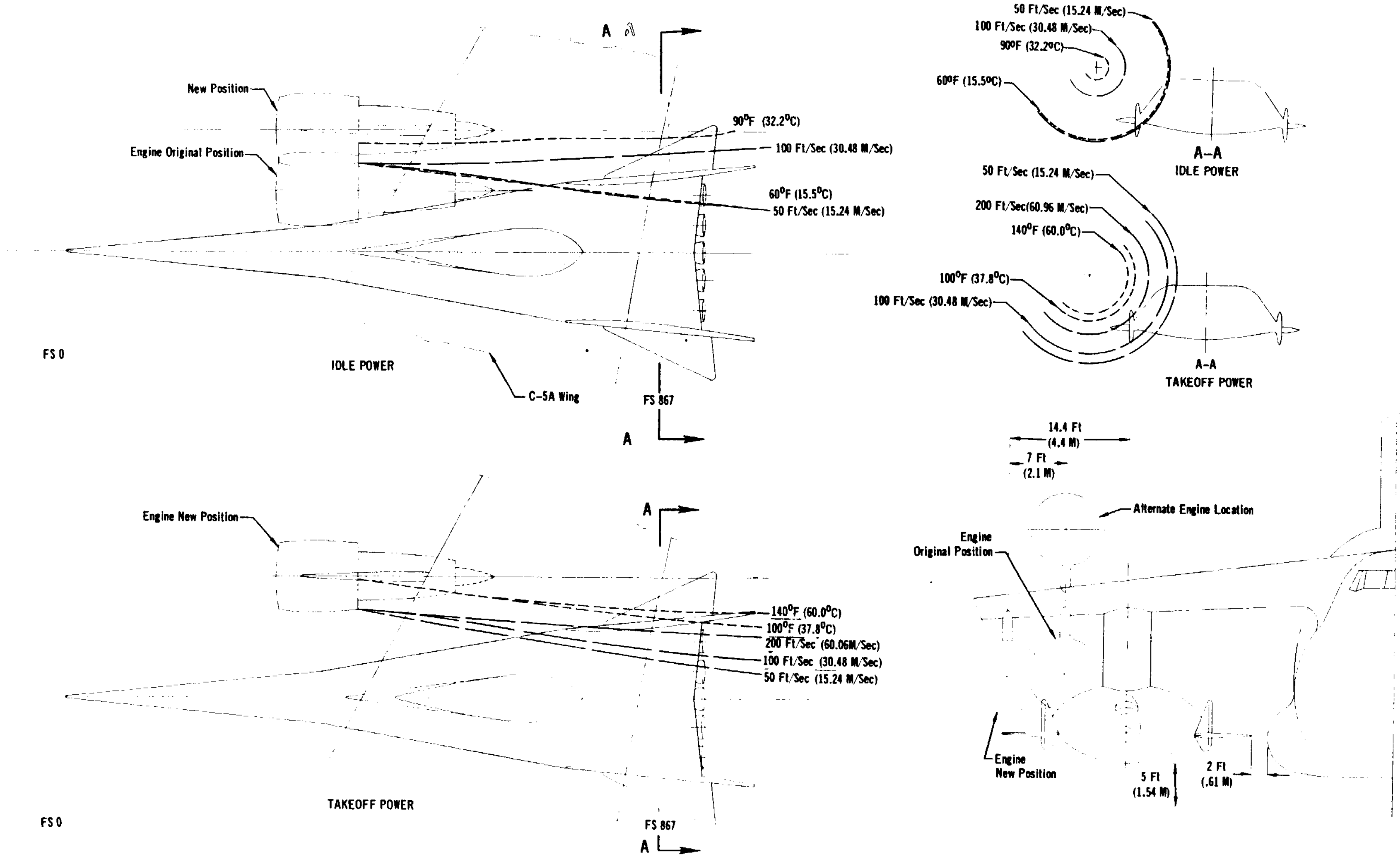
(U) FIGURE 4-7
C-5A ENGINE RELOCATION FOR MACH 12 AIRCRAFT INSTALLATION



(U) Figure 4-8 illustrates the change required in the landing gear housing pod. The clearances are adequate but not as great as with the engine relocation. The C-5 landing gear would require rearrangement and possibly reduction in quantity of wheels. However, this appears feasible, since the landing gear redesign would be made for the specific mission of launching the research vehicle and would be operated from hard surfaced airfields, thus resulting in an appreciable reduction in the load-carrying requirement for the gear. Again, the idle power and takeoff power C-5A engine wake velocities and temperatures are shown impinging on the test vehicle. The impingement is more pronounced, but not considered damaging under these power settings. The takeoff power setting of the C-5A is judged the most severe conditions during carriage and launch operations.

(U) Each redesign will require a structural analysis trade study of the C-5A aircraft to determine the advantages and the most economical solution. It is recommended that the engine relocation be investigated as the primary alteration as it is expected to require the least modification to major structure.

(U) Fuel tank pressurization, crew life support systems requirements, and general electrical check-out and power requirements will be furnished by the C-5A aircraft to the research vehicle prior to launch through the wing pylon.

(U) At Mach 0.8 and at 35,000 ft (1068 m) altitude, with the C-5A aircraft heading toward the landing base, the research vehicle is released. After release, the five rocket engines are ignited and the vehicle climbs to cruise altitude and Mach number, where four of the rockets are shut down and the centerline engine is throttled back for cruise flight. During the cruise period all flight data are recorded on board and telemetered to ground stations along the ground track. After five minutes of cruise flight, the propellants are depleted and an unpowered glide and descent is made. Energy management to achieve a high key position over the landing site is accomplished by modulating vehicle L/D. A 360 degree overhead approach to a landing is made.

(U) The aircraft windshield and canopy are integral with the cockpit which raises and lowers as required during the flight. To withstand the high thermal environment and aerodynamic forces during the acceleration, cruise and descent portion of the flight, the cockpit is retracted. The cockpit is raised during the final portion of the unpowered glide to permit the pilot to make the approach and landing visually.

(U) Along with a normal complement of avionic equipment, the vehicle will have a 1500 pound (680.4 kg) instrumentation payload which can be utilized in a manner best suited for the particular research. Instrumentation electronics may be partially replaced by addition of other equipment deemed necessary for the flight. Each research option modification kit will probably require some alteration to the payload to best utilize the test capability of the aircraft.

(U) The exhaust plumes from the outboard engines of the five RL10-A-3-9 rocket propulsion system expand so as to impinge on the vertical tail surfaces during climb/acceleration. During cruise the outboard engines are shut down and the single center engine plume does not impinge on any vehicle surface. When the plume meets the

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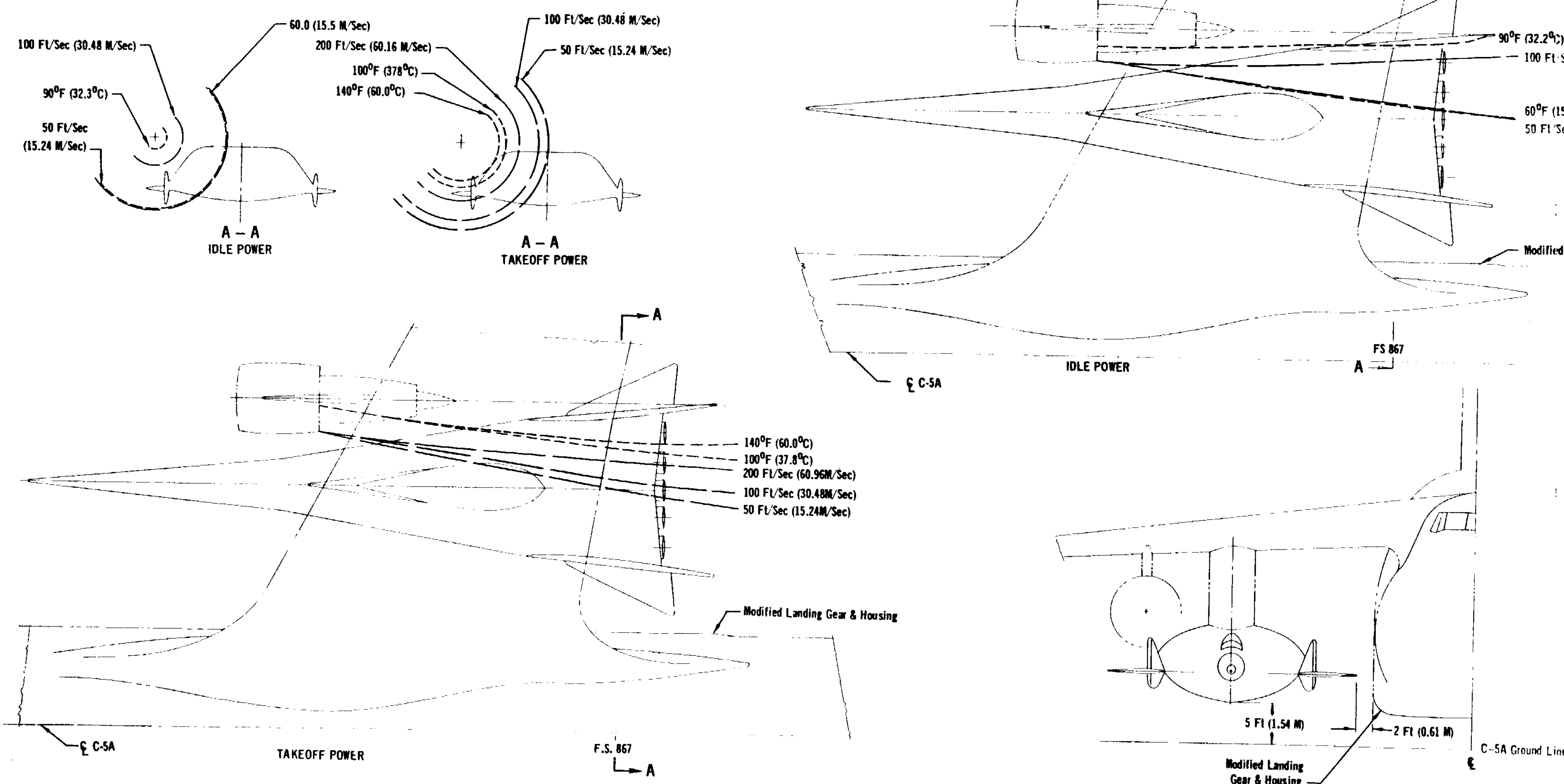
(U) FIGURE 4-8
C-5A LANDING GEAR MODIFICATION FOR MACH 12 AIRCRAFT INSTALLATION



vertical tail surfaces, it is along the rudder/speed brake panels, which have columbium shingle structure. Total temperature of the plume gas is about 2000°R (1111°K) less than the Mach 12 cruise air temperature for which this surface was designed. Thus the plume impingement does not constitute the critical temperature load on this component. In the event that the plume flow gasdynamics (shocks, boundary layer separation, etc.) create local heat transfer problems, additional thermal protection such as ablative coatings or a higher temperature capability material such as tantalum may be required.

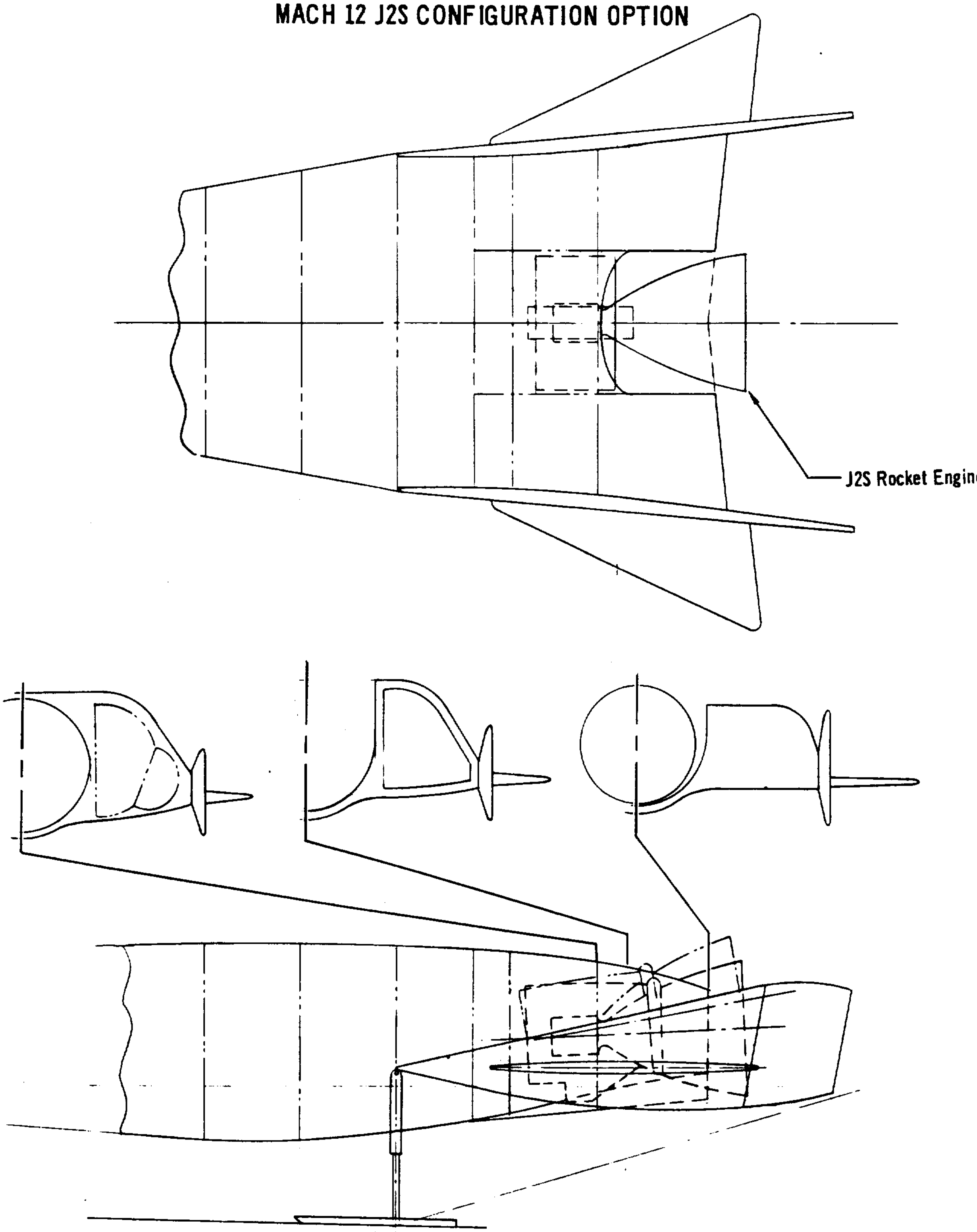
(U) While the engine concept employed as the basis for the study refinement uses five RL10-A-3-9 rockets, another attractive option is also available using a single J2S off-the-shelf rocket. This design alternative, illustrated in Figure 4-9, would require different structural load paths to redistribute the basic thrust loads because of its large physical size and thrust. Either deep throttling or engine pulsing would be required to achieve cruise flight. Roll, pitch and yaw control are available through gimballing of the RL10-A-3-9 rockets; however, an additional roll control system would be required for the J2S concept. Such control requirements are only necessary for the vertical takeoff demonstration and possibly during maximum altitude pull up and recovery demonstrations. Performance synthesis for this alternate engine concept was not determined, but it is judged that the weight and cost differences would not be significant.

4.2.1.2 (U) <u>Horizontal Takeoff (HTO) Option</u> - The ability to takeoff and land horizontally from conventional air fields will provide additional valuable research on the subsonic flight handling and landing characteristics of hypersonic vehicle configurations.

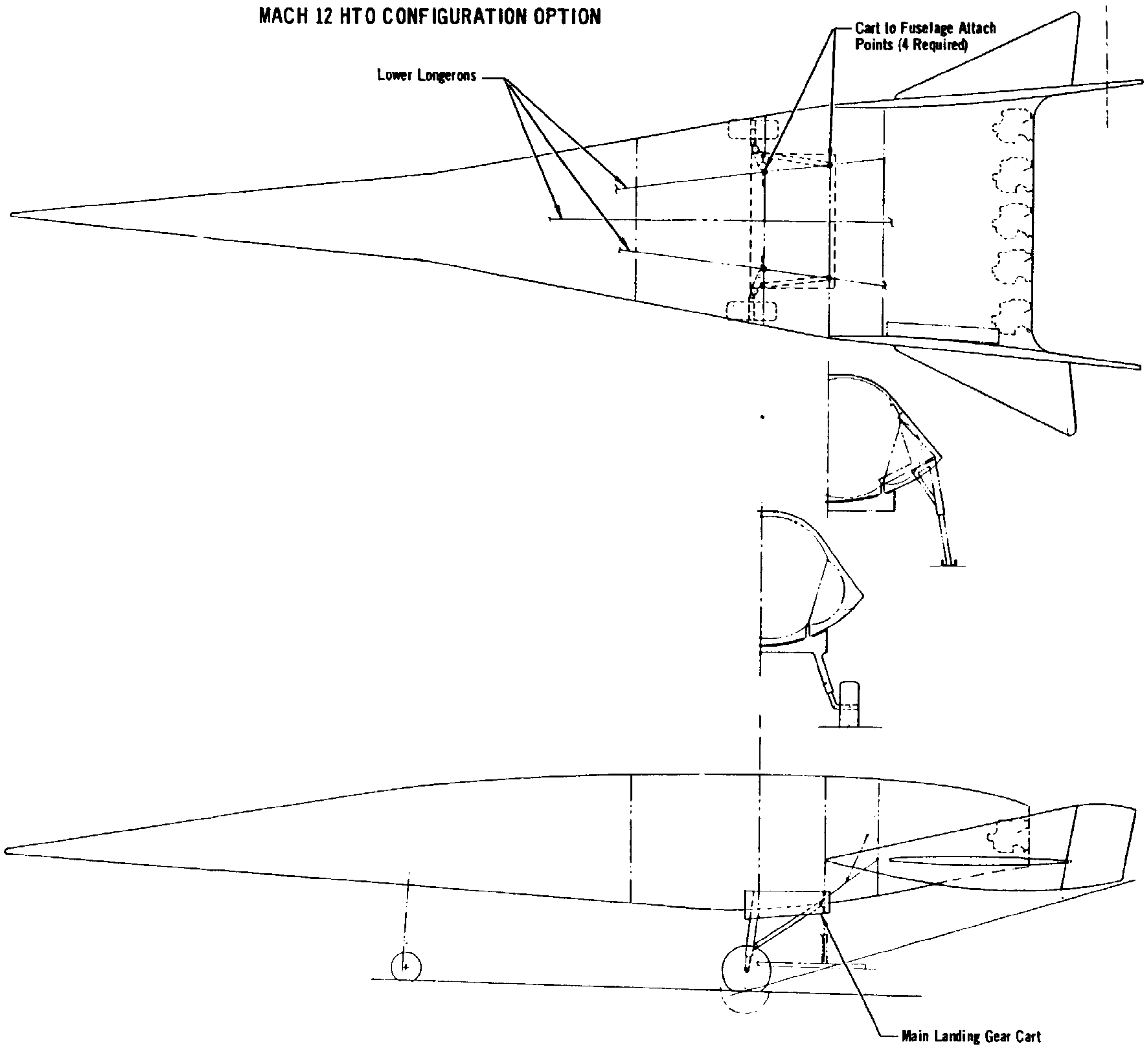
(U) The basic test vehicle is designed for air launch and therefore uses main landing gear skids positioned well aft on the aircraft to minimize strut length and to provide the best weight distribution on the landing gears and is not influenced by the normal aircraft takeoff rotational requirements. To convert the skids directly to a wheeled main gear would require that the main landing gear be moved forward and lengthened. When retracted this would cause a reduction in fuel tankage volume. The forward positioning and lengthening of the main gear would be necessary to permit aircraft rotation and aft fuselage ground clearance during ground takeoff. The fuel volume would be reduced because the longer wheeled gear occupies more usable volume than the shorter skid gear. Thus an approach was developed that would not result in a penalty to the basic aircraft.

(U) The approach adopted, as illustrated in Figure 4-10, is to use a dolly type gear attached to the under side of the fuselage at predetermined hardpoints utilizing remotely controlled disconnect pins. The wheeled dolly is rolled under the jacked up aircraft and the gear cradle is attached to the lower fuselage. The aircraft is lowered onto the fixed gear and the skid main gear is retracted. The nose gear strut is extended to compensate for the added main gear dolly height. The main gear must be designed to permit landing of the aircraft on the dolly after dumping excess fuel if the remotely controlled fasteners fail and the gear dolly cannot be separated.

(U) FIGURE 4–9
MACH 12 J2S CONFIGURATION OPTION



(U) FIGURE 4-10
MACH 12 HTO CONFIGURATION OPTION



Basic Modification for HTO

1. Add capability to extend nose gear 21 inches (53.3 cm) at takeoff.
2. Structural beef-up of bulkheads. Also, add structural fittings to bulkheads for main landing gear cart attachment and release mechanism.
3. Beef up fuel tank shear webs and lower longerons for main landing gear drag loads.
4. Remove (5) RL-10 rockets with ϵ = 32 and add (5) RL-10 rockets with ϵ = 7.4.
5. Modify fuselage fairing for shorter engines.

Main Landing Gear Cart

1. Cart is released at pilot's option.
2. Cart is retrieved by parachute recovery system.

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(U) Normal HTO flight sequence includes take-off, separation of the gear dolly, completion of the required cruise mission, and landing on the vehicle's normal skid landing gear. The separation and recovery of the main gear dolly can be accomplished in many different ways. The one illustrated is based on a pilot operated system over a designated ground area. A parachute recovery system on the gear dolly is initiated on positive separation of the dolly and aircraft. A ground recovery team locates and returns the dolly for inspection, refurbishment and reuse. The extended nose landing gear is held in the extended position until the gear dolly is separated from the aircraft. At this time the strut is returned to the normal extended position and then retracted. Normal nose gear extension is used for landing with the skid main gear.

(U) The other basic change required for HTO capability is replacing the basic aircraft RL10-A-3-9 rockets with RL10-A-3-9A rocket engines with a lower expansion ratio of 7.4 required for ground level operation. A thrust reduction is accepted and results in a shorter cruise time. The fairing around the engines is replaced with a different fairing to accommodate the short engine nozzles.

4.2.1.3 (U) Vertical Takeoff (VTO) Option - The vertical takeoff option offers research capability for possible operational systems. The ground level operation requires the low expansion ratio engines used on the HTO modification. The engine fairings will be the same for both modifications.

(U) To accomplish VTO, ground handling facilities are required to position the aircraft and restrain it in the vertical position until takeoff. One method of fulfilling these requirements is illustrated in Figure 4-11. An actuated tilt launch beam hinged on the edge of a rocket exhaust cooling/deflector system is placed in the horizontal position. The launch cart, which is a multiroller saddle for the lower surface of the aircraft, is fastened to the aircraft by six remotely controlled attachment pins. A set of ground handling wheels fitted to the main gear skids allows the aircraft to be rolled into position and aligned with the tilt beam rail. The aircraft positioning cables are connected to the cart and the nose gear strut is pumped up to level the aircraft with the tilt beam rail. The aircraft positioning winch pulls the launch cart back along the tilt beam rail. When the launch cart is fully engaged with the tilt beam rail, the skid gear ground handling wheels are removed and the skid gear and nose landing gear are retracted. The aircraft positioning winch now moves the vehicle along the beam until the aft end engages the hold back pins on the vertical portion of the horizontal tilt beam. The aircraft positioning cables are disconnected from the launch cart and stowed.

(U) The tilt beam actuator erects the tilt beam assembly with the aircraft and locks it in the vertical position. The aircraft is held vertically by the tilt beam assembly at the hold back pins and horizontally by the launch cart rollers in the tilt beam rails. Vehicle fueling and pilot ingress would be accomplished with the vehicle in the vertical position. Both of these operations would be accomplished by a gantry or elevator type ground equipment mechanism.

(U) The launch procedure is ignition of the rockets and at a predetermined thrust level, nozzle chamber pressure or other desirable parameter, release of the hold back pins. The rocket thrust moves the vehicle and launch cart up the tilt beam and off to a vertical takeoff. The control system is required to gimbal the outboard rockets for roll pitch,and yaw control during the climb out.

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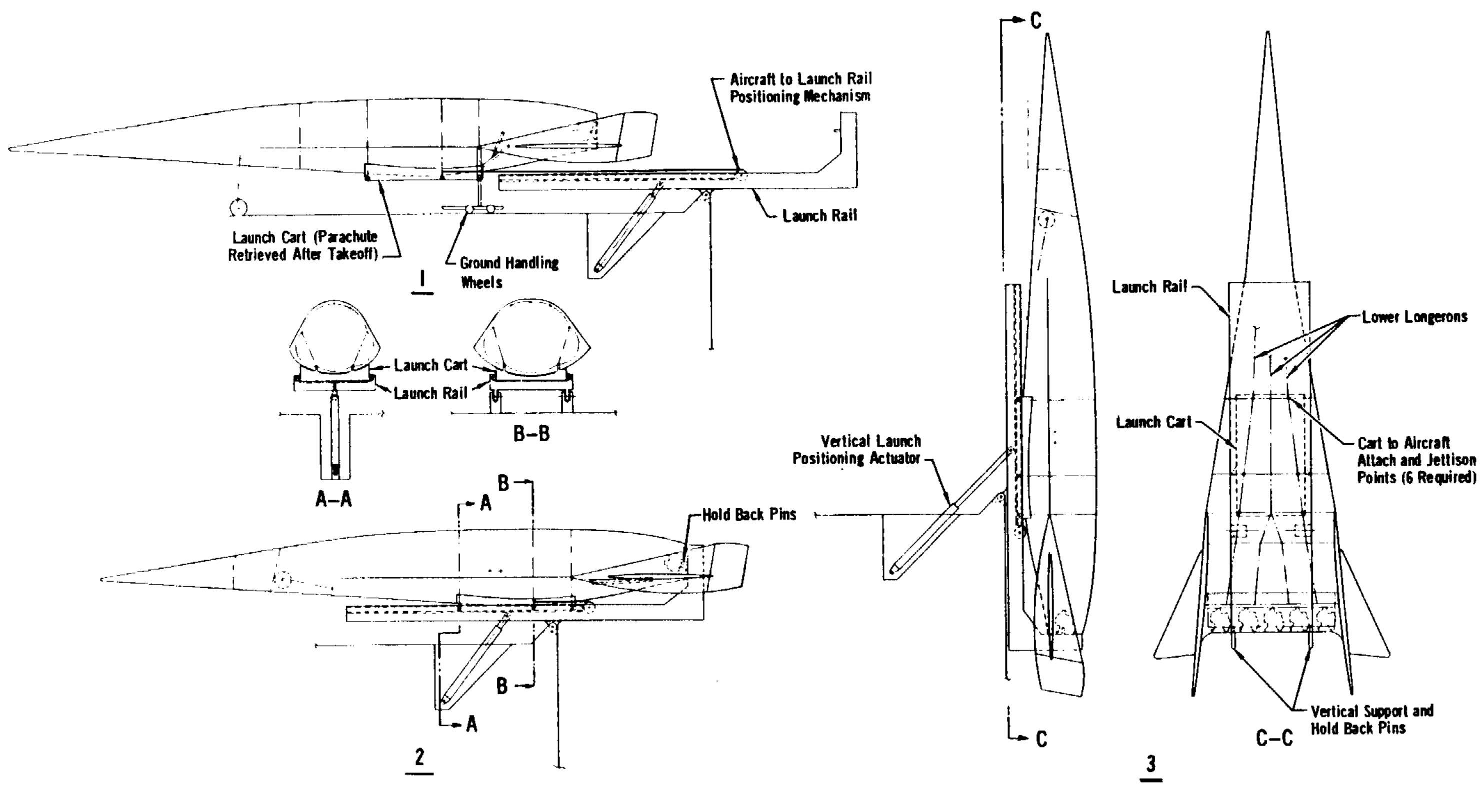
(U) FIGURE 4-11
MACH 12 VTO CONFIGURATION OPTION



Basic Modification for VTO

1. Add capability to extend nose gear 15 inches (3.8.1 cm).
2. Structural beef-up of bulkheads and add structural fittings to bulkheads for launch cart attachment and release mechasism.
3. Beef-up engine mount structure in the area of the vertical support points.
4. Remove (5) RL10–A–3–9A rocket with $\epsilon = 32$ and add (5) RL10–A–3–9A rockets with $\epsilon = 7.4$.
5. Modify fuselage fairing for shorter engines.

Launch Preparation

1. Extend nose gear, attach the launch cart to aircraft and ground handling wheels to skid gear. Align aircraft with launch rail and attach positioning mechanism.
2. Move aircraft on to launch rail, retract landing gear skids and attach hold back pins. Disconnect positioning mechanism.
3. Rotate aircraft into launch position and add fuel and crew.
4. Launch Sequence:
   - A. Start rocket engines.
   - B. When the required thrust or rocket chamber pressure is obtained, the hold back pins are released.
   - C. The aircraft moves up the launch rail until the launch cart is disengaged from the launch rail.
   - D. The launch cart is jettisoned on pilot command and is recovered by parachute recovery system.
   - E. Landing accomplished by using normal skid main gear and normal nose gear in conventional manner.

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(U) After the launch cart clears the launch tilt beam rail, the cart is jettisoned manually at the pilot's option or automatically at some predetermined height. When cart and aircraft separate, a parachute recovery system is deployed from the launch cart. After recovery by a ground crew the cart is inspected and refurbished for reuse.

(U) Cruise flight, approach and landing are the same as for the basic vehicle.

4.2.1.4 (U) Convertible Scramjet (CSJ) Option - The research value of air breathing hypersonic propulsion systems development makes this option, illustrated in Figures 4-12 and 4-13, one of the more important variations achievable with a Mach 12 vehicle. The convertible scramjet is designed to operate efficiently between Mach 3.5 and Mach 12. Subsonic burning in the scramjet is utilized between Mach 3.5 and Mach 6 with a switch-over to supersonic burning from Mach 6 to Mach 12.

(U) The important parameters required for this modification are the proper inlet ramp angles, correct exit nozzle contour and proper positioning of the engine modules with respect to Mach number. The variations of these parameters will constitute a major segment of the research program.

(U) The forebody fuselage angle has been established as described in Phase II. This conic lower surface acts as the first ramp of the CSJ inlet. The second ramp is fixed and established by angles from the CSJ inlet lip. The third and final inlet ramp is variable and as the CSJ modules are raised or lowered,the ramp, hinged on the forward end, follows the upper edge of the CSJ inlet cowl. The CSJ module consists of a short section of fixed inlet, the combustion section and a fixed portion of the expansion nozzle. The entire module is regeneratively cooled and moves as a unit. The exit nozzle is a defined contoured shape on the lower surface of the vehicle aft body. The first portion of the nozzle is hinged at the upper edge of the CSJ module exit nozzle and telescopes into the fixed portion of the exit nozzle when the engine module is actuated. The aft portion of this nozzle is fixed and created by re-shingling the fuselage with differently shaped shingles and standoff supports.

(U) Figure 4-14 shows the CSJ module in its various operational positions and Figure 4-13 shows the support and actuation method. The basic CSJ module is positioned in the retracted position at launch. The rocket engines boost the vehicle through the transonic drag rise at Mach 1.0 and on up to Mach 3.5. At any time after the transonic drag rise up to Mach 3.5 the CSJ is lowered into the initial low speed extended operating position. The engine module forward actuators extend to rotate the engine into a horizontal position at which time the aft actuators extend at the same rate as the forward actuators and vertically translate the engine module to its lowest position. The third inlet ramp follows the upper cowl lip by separately programmed ramp actuators. The movable portion of the exit nozzle follows the engine exit nozzle by mechanical linkage. This engine module lower position will permit operation of the CSJ engine with subsonic burning up to Mach 6.0. Although the shock wave is not on the inlet cowl lip, the inlet efficiency is acceptable during this subsonic combustion mode of operation. At Mach 8.0, the bow shock wave falls on the engine cowl inlet lip and the four module actuators and the two third inlet ramp actuators are programmed with reference to Mach number to maintain the "shock

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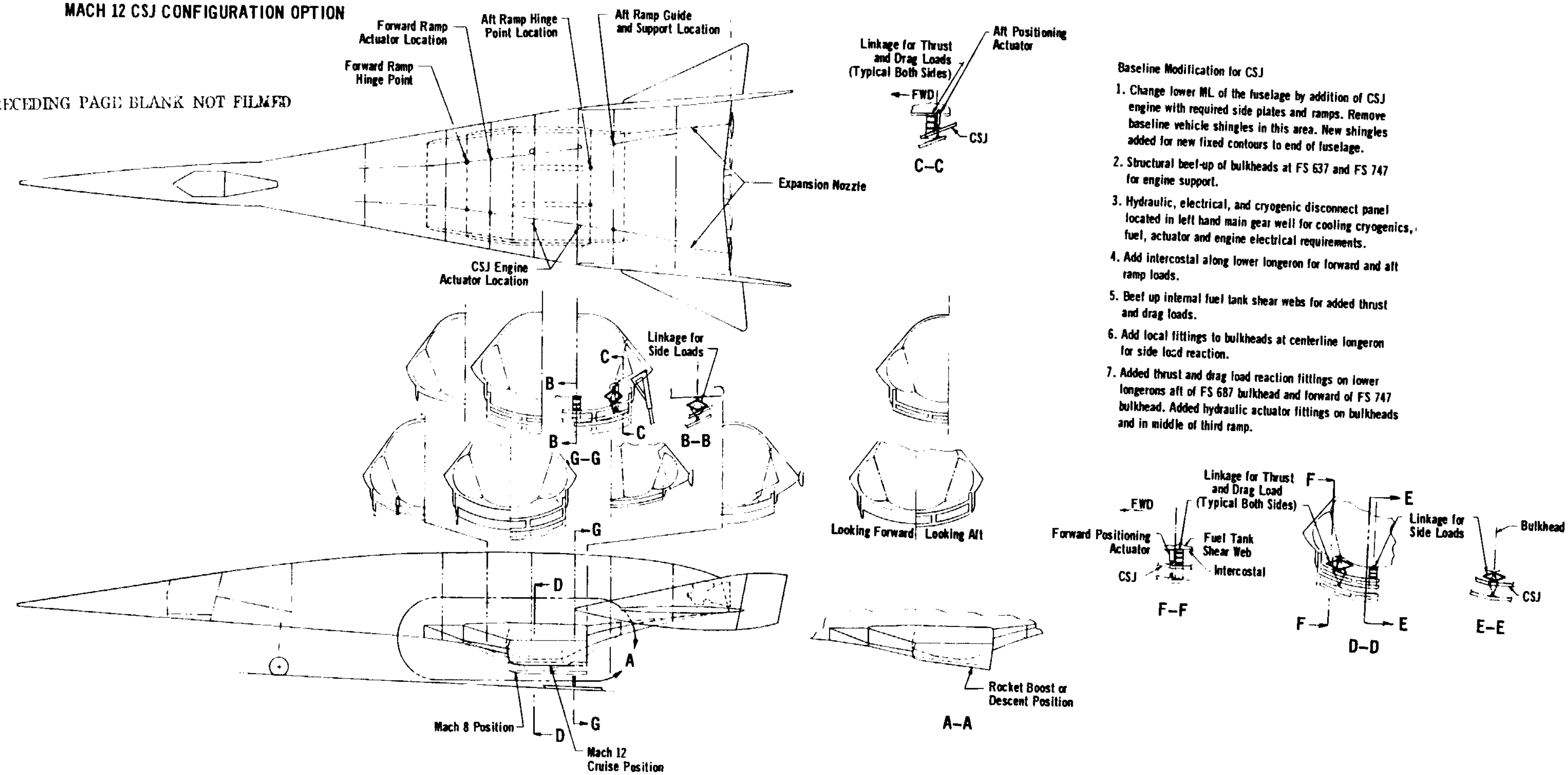
(U) FIGURE 4-12
MACH 12 CSJ CONFIGURATION OPTION



Baseline Modification for CSJ

1. Change lower ML of the fuselage by addition of CSJ engine with required side plates and ramps. Remove baseline vehicle shingles in this area. New shingles added for new fixed contours to end of fuselage.
2. Structural beef-up of bulkheads at FS 637 and FS 747 for engine support.
3. Hydraulic, electrical, and cryogenic disconnect panel located in left hand main gear well for cooling cryogenics, fuel, actuator and engine electrical requirements.
4. Add intercostal along lower longeron for forward and aft ramp loads.
5. Beef up internal fuel tank shear webs for added thrust and drag loads.
6. Add local fittings to bulkheads at centerline longeron for side load reaction.
7. Added thrust and drag load reaction fittings on lower longerons aft of FS 687 bulkhead and forward of FS 747 bulkhead. Added hydraulic actuator fittings on bulkheads and in middle of third ramp.

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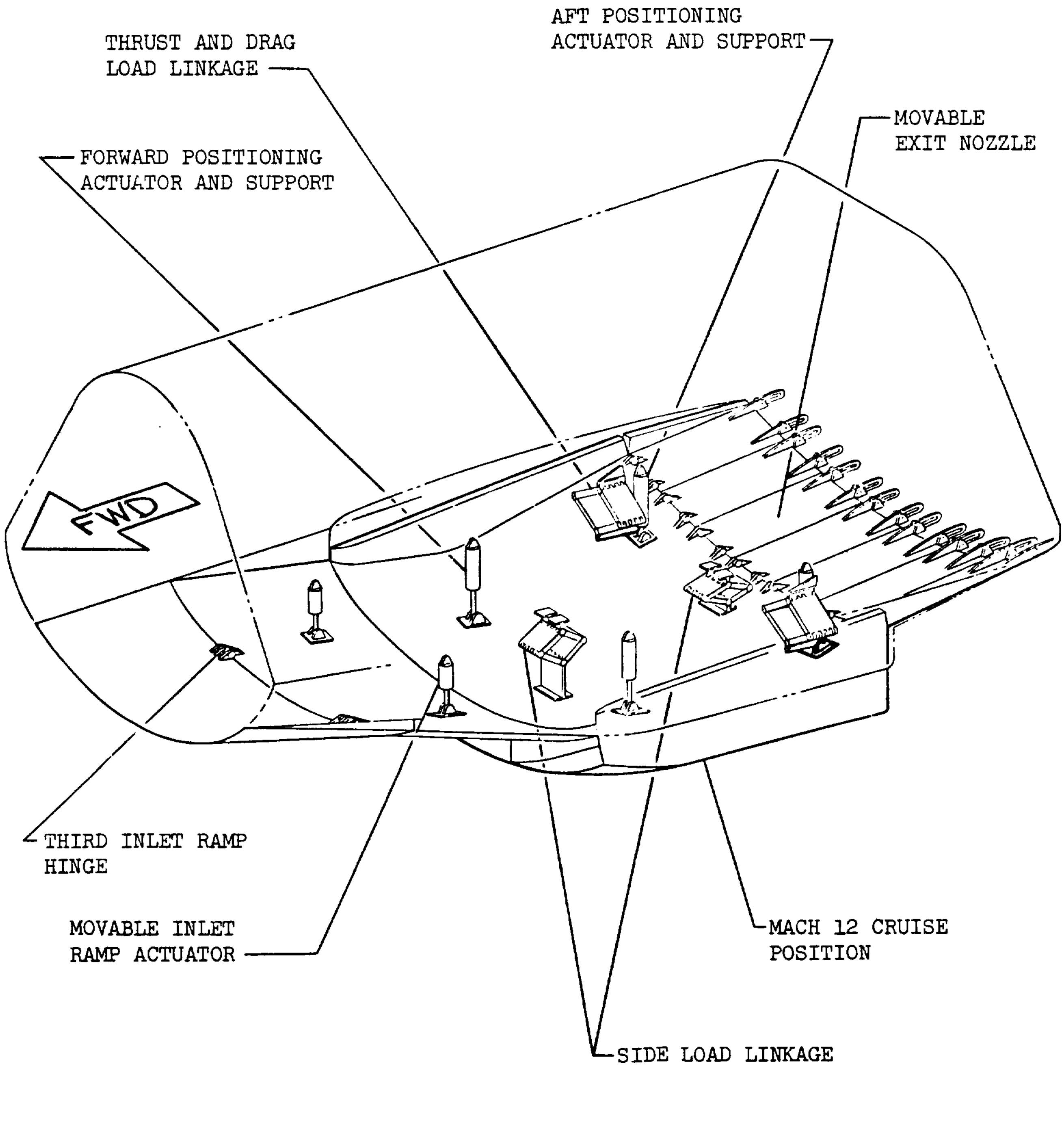
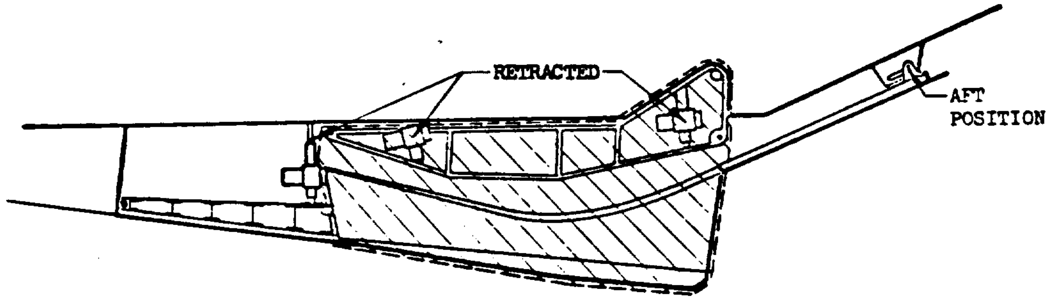
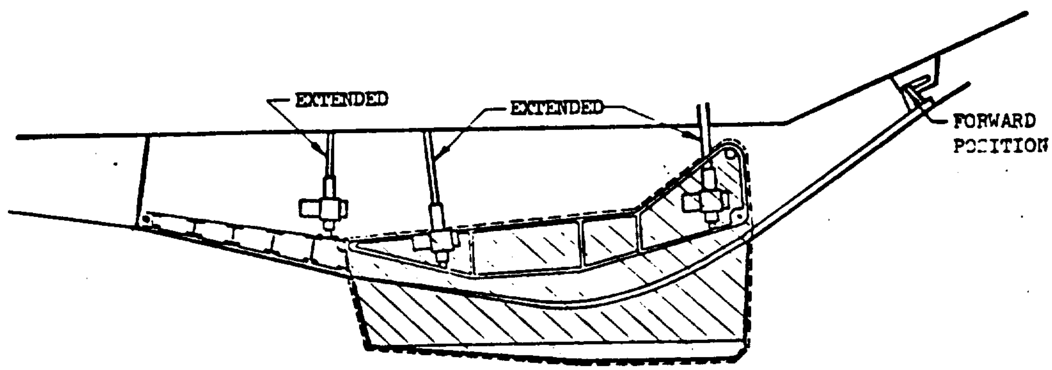
(U) FIGURE 4-13
CSJ/SJ ACTUATION AND SUPPORT MECHANISM



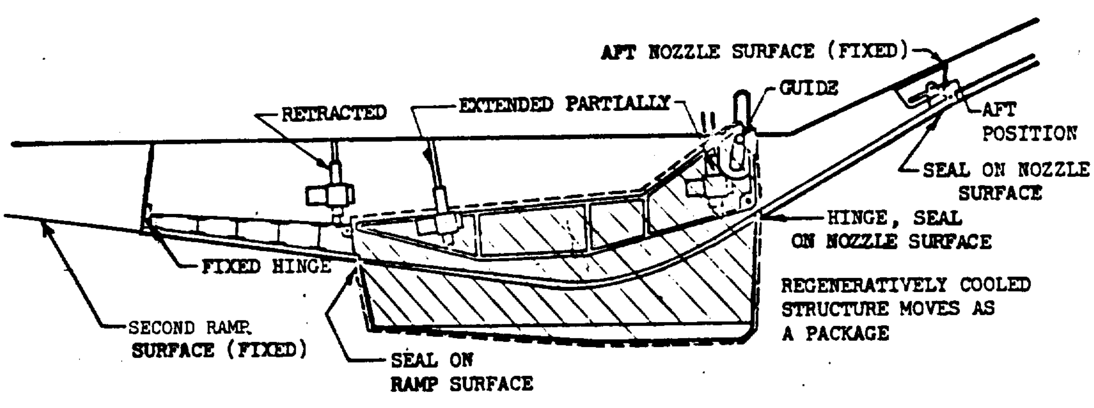
(U) FIGURE 4-14
CSJ/SJ OPERATING SEQUENCE



Rocket Boost or Descent Position



Mach 8 Position



Mach 12 Cruise Position

on lip" condition from Mach 8.0 to Mach 12.0. The four engine module actuators will retract the module to achieve the required inlet conditions up to Mach 12.0. The Mach 12 condition is the uppermost module position and is maintained for the cruise portion of the flight to fuel depletion. At this time the third inlet ramp is in a fully retracted position and the engine module forward actuators rotate the engine cowl lower lip up flush with the third inlet ramp and effectively close the inlet for the glide and descent portion of the flight. This is the initial retracted position of the CSJ module.

(U) Figure 4-13 illustrates the method by which the thrust, drag, vertical and side loads are transmitted from the CSJ engine module to the basic aircraft structure. Four scissors type linkages are used to transmit the fore and aft loads and the side loads to the aircraft structure. Vertical loads are reacted by the four actuators. The engine module rotation to the closed position is accomplished by aligning the aft lower actuator attachment points and a secondary hinge on the thrust load linkages to form a hinge line for module rotation. The secondary hinge point is the load pickup point and these linkages are not attached to the CSJ except through this hinge point. When the module is rotated sufficient clearance is maintained between the linkage and CSJ module to permit rotation of the module to the closed position. The forward side load linkage is compressed when the forward actuators retract and rotate the module to close the inlet.

4.2.1.5 (U) Scramjet (SJ) Option - This option, Figure 4-15A, is very similar to the convertible scramjet option described previously. The primary difference between the two systems is that the scramjet engine has no subsonic burning mode and therefore is used only between Mach 8 and Mach 12. The internal modules are different in that the combustion chamber is shorter and the exit nozzle contour is optimized for the Mach 8 to 12 operating range. The number and location of fuel spray nozzles are also different.

(U) Figure 4-15B shows the exit nozzle contour change and the shortened module length. The module and ramp actuators,scissors linkage load transfer system and module pivoting for retraction are the same as for the CSJ. The vehicle must be accelerated by rocket to Mach 8 before the SJ module is lowered for operation. The engine will then accelerate to and cruise at speeds approaching Mach 12 on the SJ alone.

4.2.1.6 (U) Thermal Protection System (TPS) Option - The basic approach for thermal protection system testing is to provide a complete peripheral test bay in the area of the forward fuel bulkhead as illustrated in Figure 4-16. The test bay extends 3 ft (0.9m)aft and 2 ft(0.6m) forward of the forward fuel bulkhead. Use of this bay provides simultaneous cryogenic and non-cryogenic testing of various types of TPS. The bay is created by recessing the basic vehicle structure from 2.5 inches (6.35 cm) to 6.0 inches (15.25 cm) inside the vehicle exterior mold line. Continuous load paths are maintained by tapering the structural shell over a 5 foot (1.53 m) section forward and aft of the TPS test bay.

(U) The TPS test area has the basic vehicle thermal protection system as the primary design. This TPS includes radiation shingles, passive insulation, an air gap and then a blanket water wick, which all are external to the basic structure.

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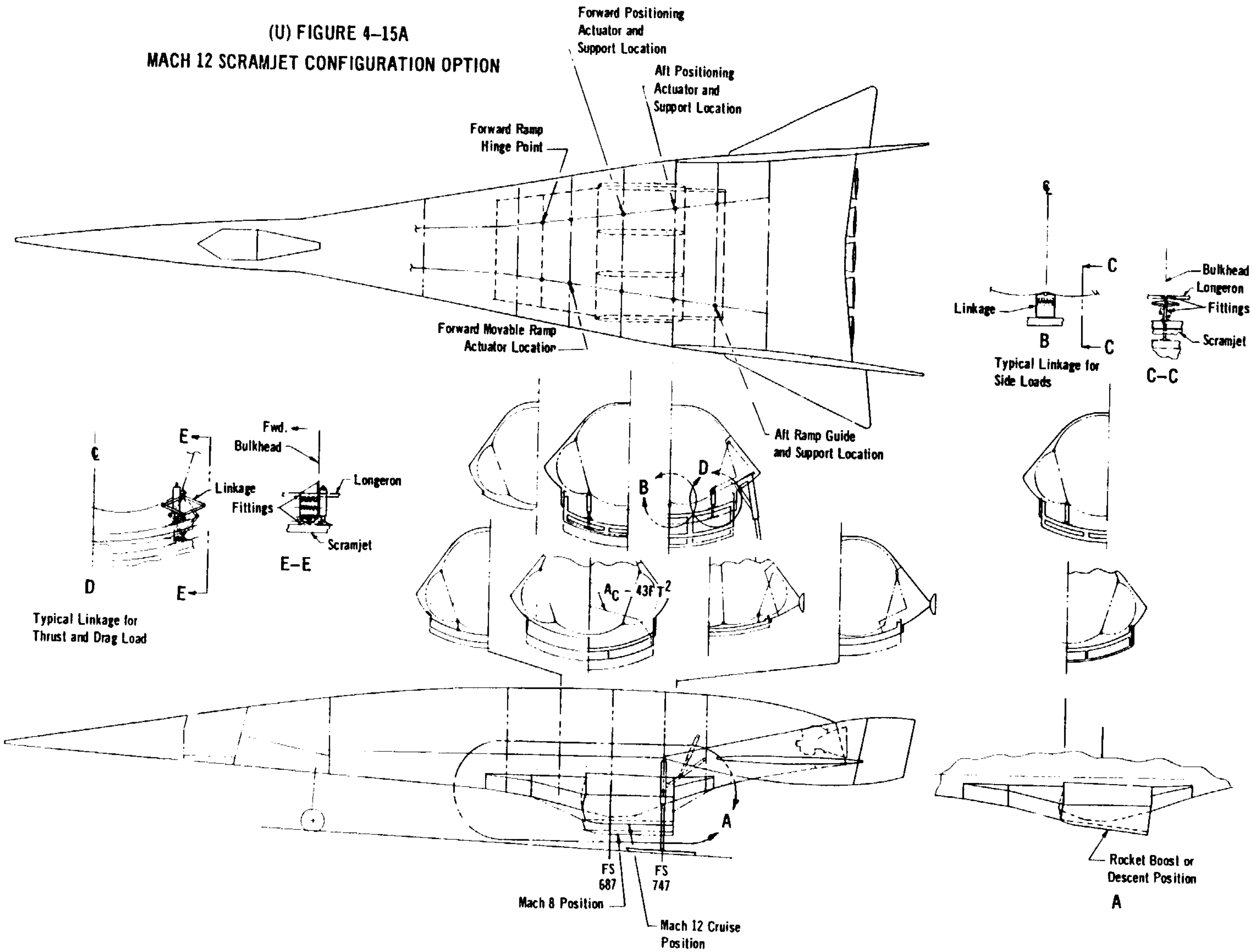
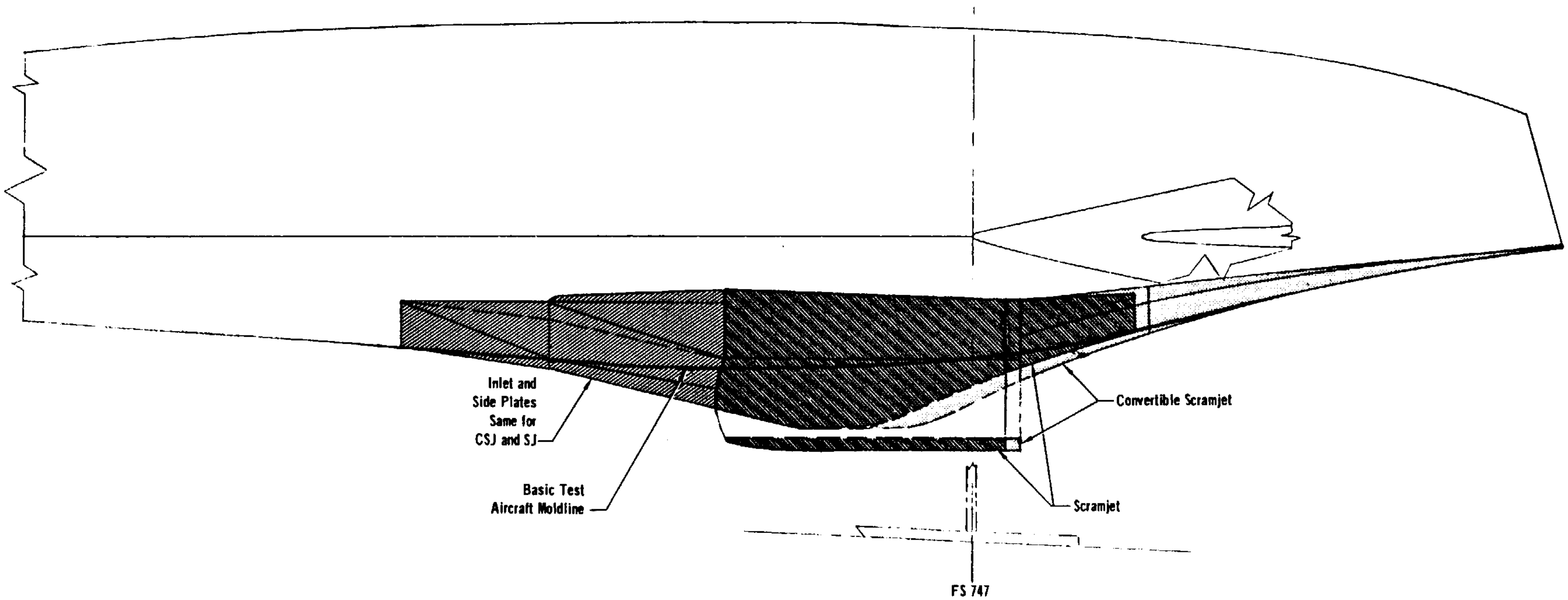
(U) FIGURE 4-15A
MACH 12 SCRAMJET CONFIGURATION OPTION



Baseline Modification for Scramjet

1. ML change by addition of scramjet engine with required side plates and ramps. Remove baseline vehicle shingles in this area.
2. Structural beef-up of bulkheads for engine support at FS 687 and FS 747.
3. Hydraulic, electrical, and cryogenic disconnect panel located in LH main gear well for cooling cryogenics, fuel, actuator hydraulics and engine electrical requirements.
4. Add intercostal along lower longeron for forward and aft ramp loads.
5. Beef-up internal fuel tank shear webs for thrust and drag load reactions.
6. Added local fittings to bulkheads at centerline longeron for side load reaction
7. Added thrust and drag loads reaction fittings on lower longeron aft of FS 687 bulkhead and forward of FS 747 bulkhead. Added hydraulic actuator fittings on bulkheads and middle of third ramp.

(U) FIGURE 4-15B
CSJ/SJ COMPARISON



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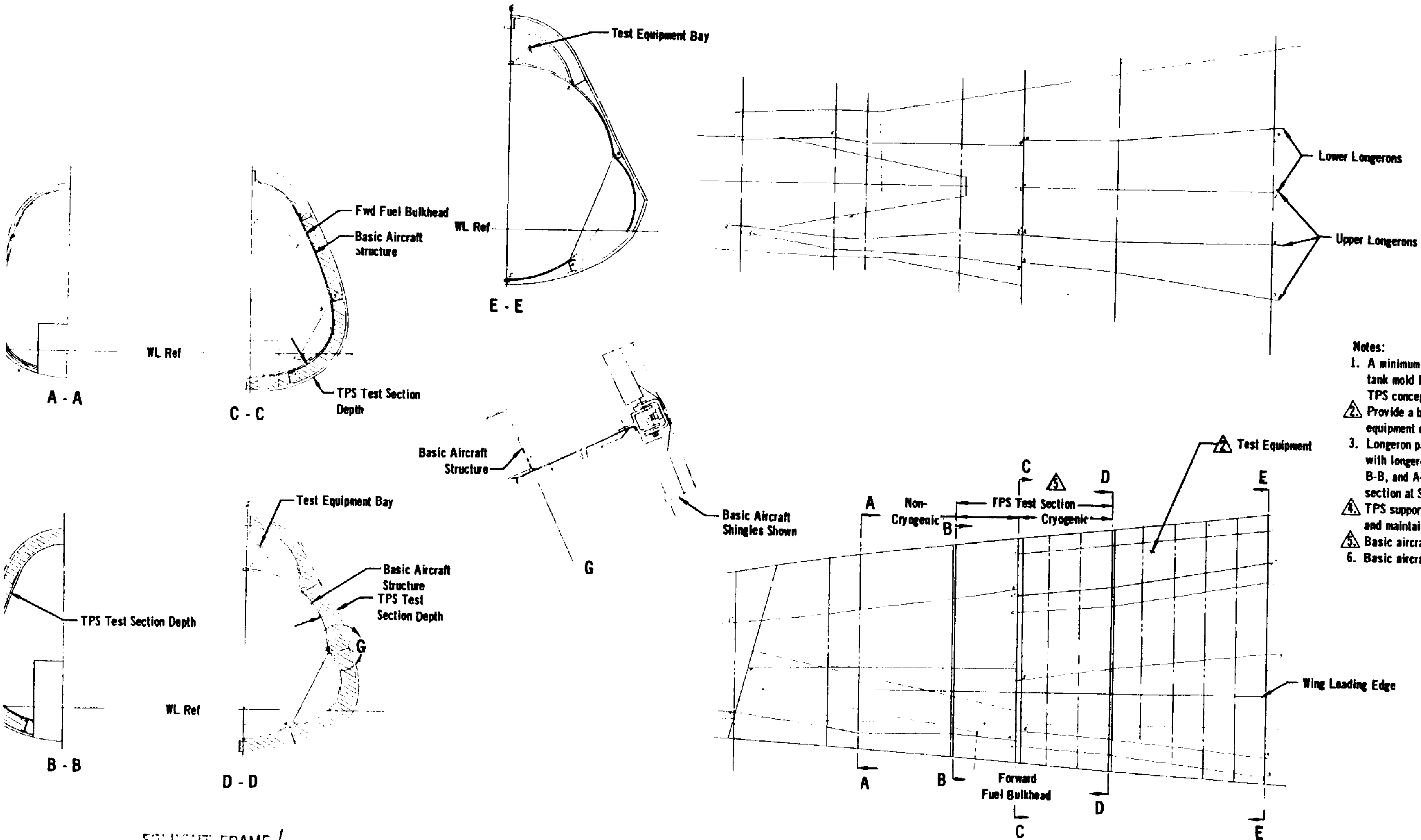
(U) FIGURE 4-16
MACH 12 TPS TEST BAY CONFIGURATION



Notes:

1. A minimum distance of 6 inches (15.24 cm) is maintained be tank mold line and the fuselage mold line for installation of TPS concepts.
2. Provide a bubble above fuel tank between section C-C and equipment compartment (i.e. pumps, pickups, coolant tanks,
3. Longeron paths slope inward from Section E-E and D-D and with longeron kick loads taken in structural rings at Section B-B, and A-A. Structure is tapered gradually to full depth of section at Sections D-D and B-B.
4. TPS support intercostals can be varied in depth to accomoda and maintain outside moldline.
5. Basic aircraft TPS removable for test TPS installation in thi
6. Basic aircraft active TPS is shown.

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Different insulations, shingles and/or purging systems can be installed in the test bay. An equipment bay is provided above the forward fuel tank for special equipment, various cryogenic reservoirs, instrumentation, etc.

(U) This peripheral test bay provides good flexibility in testing systems for long periods of time. Actual environmental conditions are encountered and monitored throughout the flight. All tests will be full scale actual hardware specimens subjected to the temperature, pressure and air flow environment in cryogenic and non-cryogenic structural applications.

(U) An alternate TPS system investigated is illustrated in Figure 4-17. It consists of a detachable pod on the aft end of the fuselage. The pod has a flat surface test table, as a continuation of the 4° lower shear forebody line, in the high temperature, high compression area of the pressure field. The test table is four feet (1.22 m) wide and eight feet (2.44 m) long and is actuated by four screw jacks to position the table and specimen to the proper depth for a flush moldline. The aft fairing of the modification is designed for minimum base drag at low speed and can serve as a storage area for cryogenic reservoirs, pumps, instrumentation, etc. The test table would require flat specimens of the TPS to be tested and positioned on the table which may or may not be cryogenically cooled.

(U) Failure of any TPS in this alternate arrangement does not endanger the vehicle, however, other tests, such as aerodynamic, lower surface thermodynamics, convertible scramjet or scramjet options could not be performed simultaneously with this TPS testing arrangement. The reduced testing flexibility of the pod system appears less desirable than the peripheral test bay system.

(U) The research required to develop a reusable, easily refurbished TPS is best accomplished by the peripheral test bay. Additional research into hot structure, passively insulated structure, protective coatings, etc, can be made on the vertical fins, movable wing tips or a section of the nose cone.

4.2.1.7 (U) Armament Option - Applicable Mach 12 operational systems include tactical military aircraft with armament dispensing capability. The approach taken is illustrated in Figure 4-18 and is based on the premise that the primary armament to be tested is the guided missile. No detailed consideration has been given to other types of weapons.

(U) The basic vehicle modification consists of the addition of two 20-inch (51 cm) diameter, aft opening, missile launch tubes. Current missiles probably could not survive launching into a Mach 12 environment and a completely new generation of missiles will have to be developed for operational use. Aft firing launch tubes appear to be the most feasible firing system under these conditions.

(U) The launch tubes are mounted in a thermally protected structure attached to the upper surface of the vehicle basic structure. The basic water wick system is maintained between the launch tubes and the vehicle structure although the vehicle shingles are replaced with new contoured shingles to blend the launch tube contour into the fuselage shape. The buried launch tube nose reduces vehicle wave drag and provides a missile trajectory well above the engine wake and vertical fin structure.

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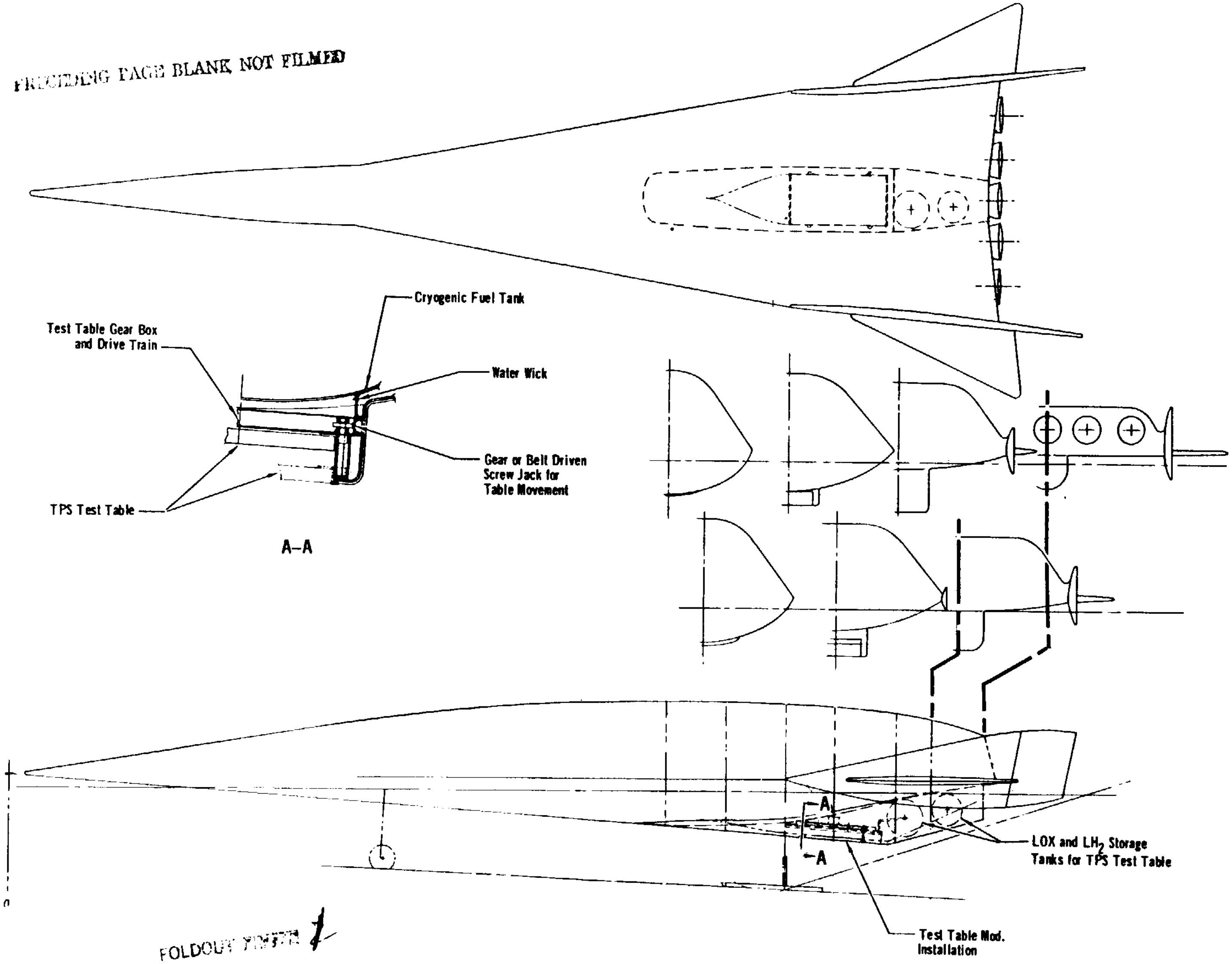
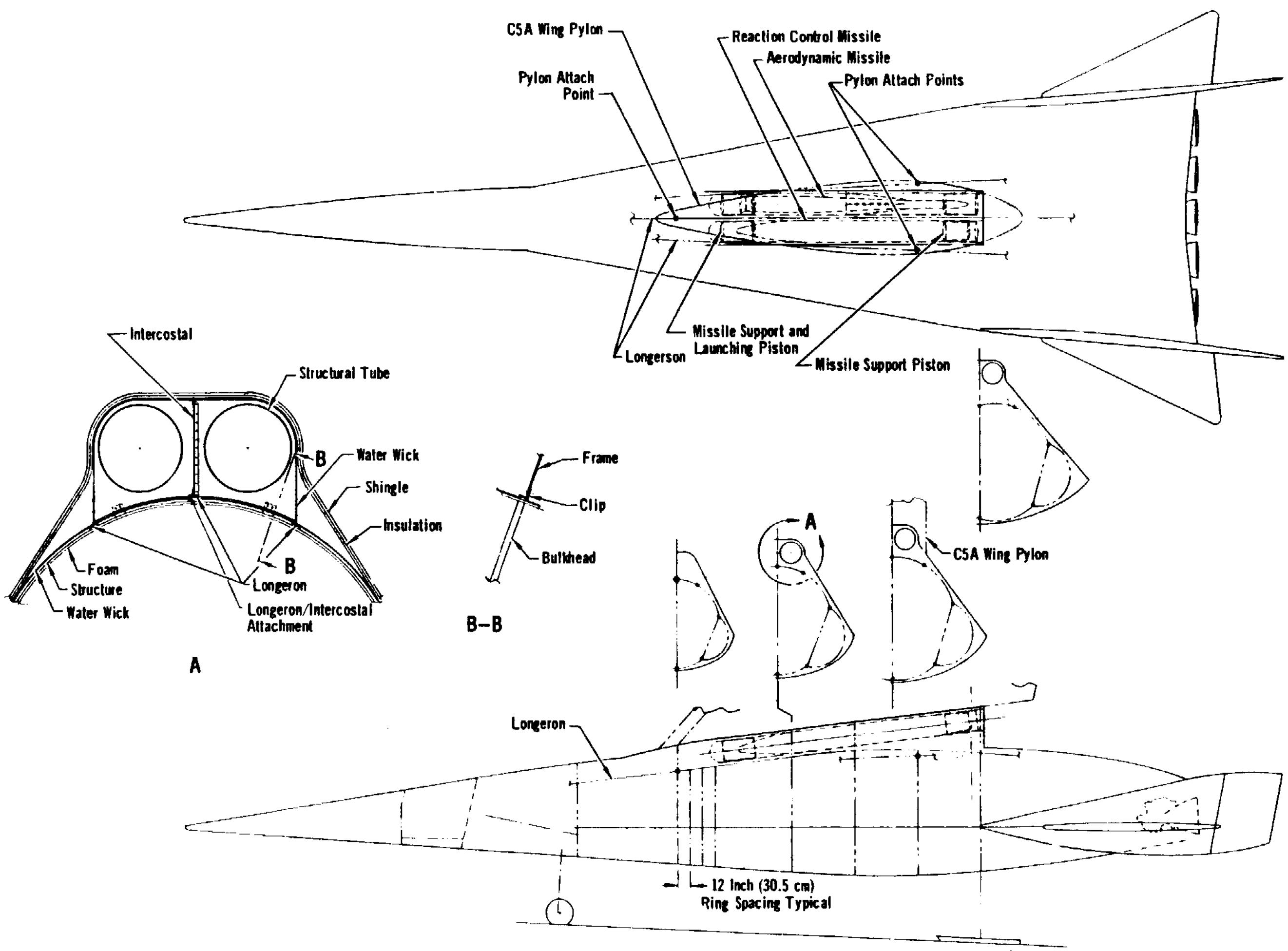
(U) FIGURE 4-17
MACH 12 ALTERNATE TPS CONFIGURATION OPTION



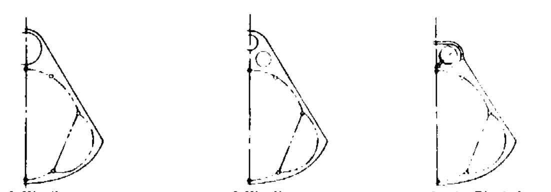
(U) FIGURE 4-18
MACH 12 ARMAMENT CONFIGURATION OPTION



Baseline Modification for Armament

1. Change upper ML by addition of new shingles.
2. Add structural clips to bulkheads for armament pod side and vertical loads.
3. Add structural member for longeron/intercostal tie for missile ejection thrust.
4. Use standard C-5A wing launch pylon with attach points. Modify pylon fairing at aft end.

Note: Alternate launch tube arrangements and launch systems.



1 Missile	3 Missiles	4 Bombs Ejected
30 In. Dia. – 220 In. Long	15 In. Dia. – 220 In. Long	16 In. Dia. – 100 In. Long
(76.2 cm Dia. – 558.8 cm Long)	(38.1 cm Dia. – 558.8 cm Long)	(40.6 cm Dia. – 254 cm Long)

(U) Test missiles will vary in size. The launch tube as designed can accept missiles 20 feet (6.10 m) long and 20 inches (51 cm) in diameter. The same physical envelope will accommodate several variations of the basic modification, one 30 inch (76 cm) diameter tube, three 15 inch (38 cm) diameter tubes or a four bomb upward ejecting bomb bay.

(U) The basic dimensions of 20 inch (51 cm) diameter and 20 feet (6.10 m) long will permit most missiles to be tested. Missiles would preferably be reaction controlled for this type of mission and altitude,but could be a finned type if the fins do not exceed or can be folded within the 20 inch (51 cm) diameter. The initial testing will be ballasted, motorless shapes for separation data only. A second testing stage could include motor firings as retro-firing missiles for low altitude targets. The final testing would require a target acquisition system and lock-on capability for a fully operational guided,powered missile.

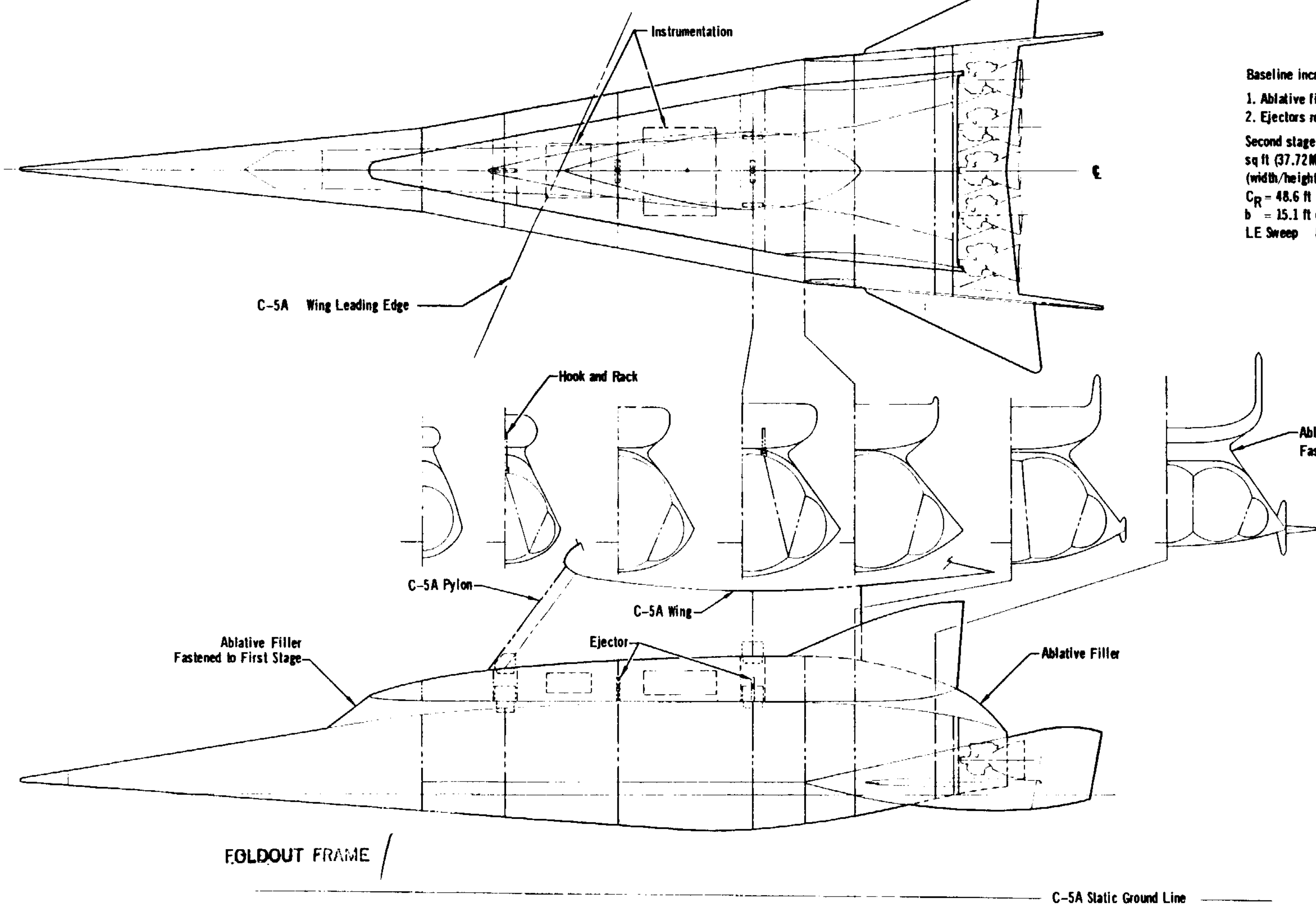
(U) All missiles would be designed to fit the 20 inch (51 cm) tube diameter with sabots to center and support the missile in the launch tube. The inside diameter or shape of the sabot will depend on the individual missile to be supported. The forward sabot includes a gas generating cartridge and the aft sabot incorporates shear pins. Each sabot is designed in multiple segments for separation after missile launch. The launching is accomplished by firing the gas generating cartridge in the forward sabot. The gas pressure builds up in the forward chamber and exerts a force on the aft sabot shear pins. When the pins are sheared, the missile moves aft and out of the launch tube. When the sabots clear the launch tube, they separate and fall away. The missile motor can be started in one of many ways electrical contact after shearing the aft sabot shear pins, radio signal, umbilical cord or firing the gas generating cartridge can initiate a time delay fuse. The type of guidance systems testing could begin with inertial systems and evolve into the more complex laser, radar or heat guidance systems.

4.2.1.8 (U) Staging (STG) Option - The staging option illustrated in Figure 4-19 can provide data on vehicle separations similar to reusable booster operations by properly scaling weight, inertia, air load and velocity parameters. Aerodynamic, thermodynamic and dynamic information of this nature is of high research interest as indicated by the research requirements analysis.

(U) The staging modification mounts an expendable second stage on the upper surface of the basic vehicle. The second stage picks up the C-5A pylon attach fittings on the upper surface of the basic vehicle and acts as a load carry through to the C-5A pylon attachment fittings. The second stage will be structurally capable of transmitting loads from the basic aircraft to the C-5A pylon.

(U) The second stage, being expendable, is made of aluminum with heavy conventional stringer, skin and frame construction. The vehicle has instrumentation and telemetering equipment aboard to record and transmit the required data. The basic vehicle trajectory does not generate a thermal build up problem but sealing between the vehicles is required. An ablative material is bonded to the basic vehicle and seals the gap between the vehicles but is not bonded to the second stage vehicle. Positive separation of the vehicles is achieved by using gas generating cartridges and kicker pistons. This system is similar to shuttle operational concepts currently envisioned. Continuing design studies may result in a different

(U) FIGURE 4-19
MACH 12 STAGING CONFIGURATION OPTION



Baseline increments required for air launched staging

1. Ablative filler requires shingle modification.
2. Ejectors require beef-up at associated bulkhead.

Second stage vehicle, S_p = 50% of basic vehicle = 406 sq ft (37.72M^2), cross section at 75% of C_R is basic 4/1 (width/height) ellipse.

C_R = 48.6 ft (14.8M)

b = 15.1 ft (4.6M)

LE Sweep 80°

concept being selected for the final design. Simulation of other staging systems concepts could be equally well accommodated in the research vehicle. The physical shape of the second stage vehicle, Figure 4-20, is of a representative all body shape of approximately 50% of the basic vehicle planform and appears consistent with present space transportation system study concepts.

(U) The proposed alternate C-5A wing pylon, Figure 4-19 will have a removable section which will permit installation of a typical two stage vehicle on the shortened pylon with adequate ground clearance.

4.2.1.9 (U) Subsonic Turbojet (TJ) Option - The purpose of the option, Figure 4-21, is to examine the subsonic realm of operations with an all body shape. The initial thoughts were to utilize existing turbojets for landing and horizontal takeoff studies. The alternate configuration shown in Figure 4-21 was feasible but the mounting pylons are so large that the aerodynamics during landing and takeoff will be altered sufficiently to make it unattractive. The primary arrangement with two turbojets in a single nacelle on a short pylon with subsonic pitot type inlets is more attractive. The location of the fuel for conventional JP burning engines is in the LO_2 tank on the vehicle center of gravity. This tank is designed for a dense fluid, and only minor fuel system changes are required such as fuel lines and pumps. A complete purge and insulation removal is required in the LO_2 tank before the bladder hangers or lacing fittings are installed in the cavity. The JP bladder type fuel cell complete with pumps, shut off system and lines will be installed. The aft location of the turbojet engines shifts the c.g. and ballast will replace payload. To simulate the flight characteristics of the basic vehicle, idle power will be used for landing and the HTO landing gear dolly will be employed as a fixed landing gear.

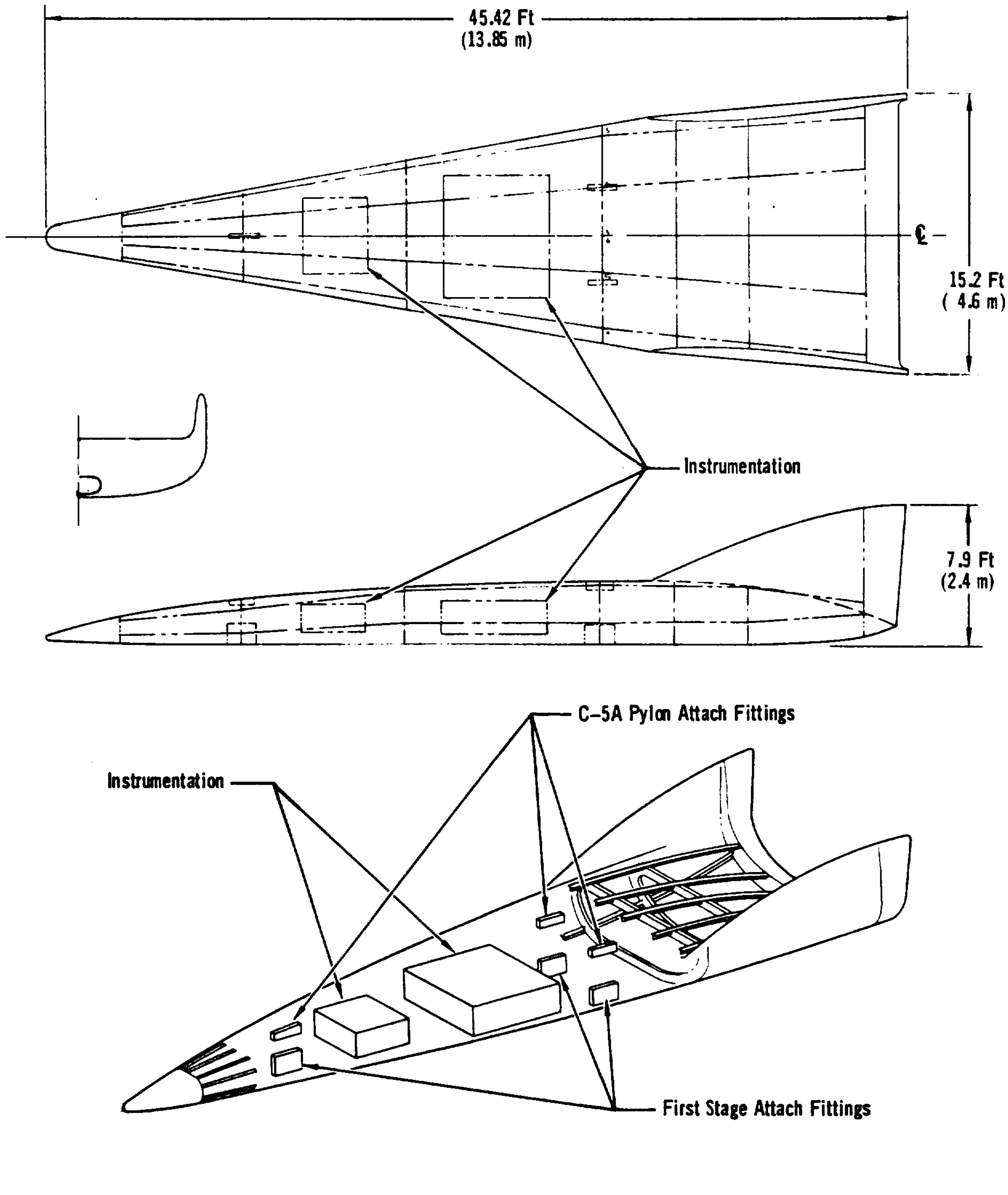
4.2.1.10 (U) Ramjet (RJ) Option - Conventional ramjet engines did not integrate well with the all body shape. Various configurations were studied in Phase I. None of the potential operational systems studied utilizes a pure ramjet for cruise power and the most predominate use of this propulsion is for missile systems using small engines.

(U) This modification, Figure 4- 22, provides a flying test bed for various size missile powering ramjets up to 20 inches (51 cm) in diameter with fixed or variable inlets. Fuel for the ramjets is carried in the mock missile body or is taken from the vehicles fuel supply. Instrumentation would permit gathering a full range of data on the ramjet operation up to flight speeds in excess of Mach 6.

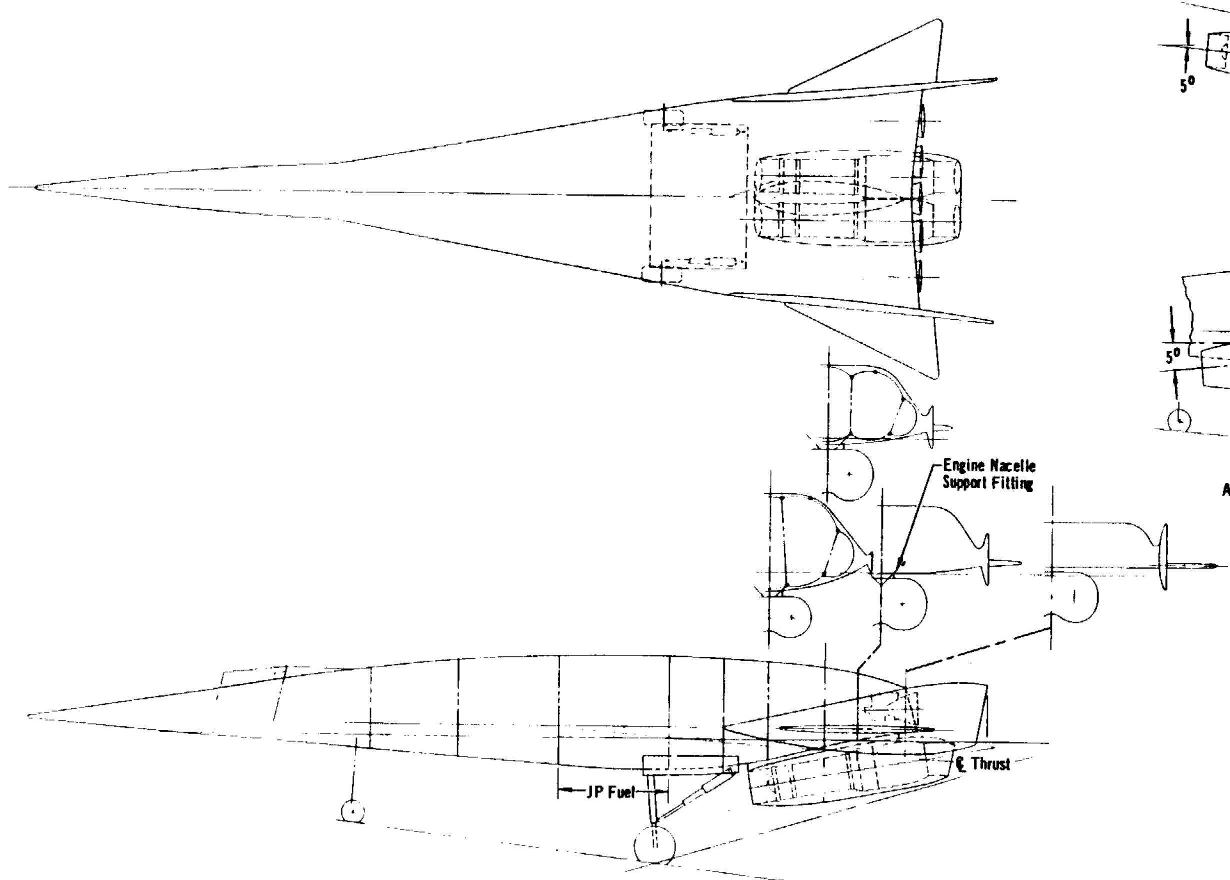
(U) The thrust line is well below the center of gravity of the aircraft but the relative thrust from these engines is small and the effect is easily trimmed out.

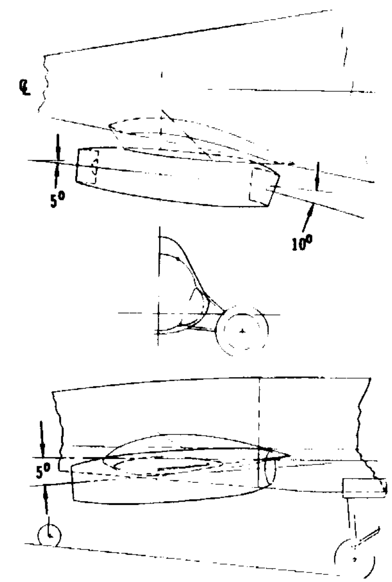
(U) The lower surface flow field for the basic vehicle was estimated to be sufficiently uniform that special diverters were not needed. It was recognized that shock waves from the ramjet would impinge on the basic vehicle, resulting in local heating rates that are higher than design values. At the intended maximum test conditions for ramjet operation (Mach 6 - 6.5), total air temperature and hence the maximum surface temperature possible is consistent with the use of columbium shingles. Since the aft lower surface on the basic vehicle does not require the use of columbium shingles (see Figure 4-76), some re-shingling of the aircraft in this area may be required.

(U) FIGURE 4-20
REPRESENTATIVE SECOND STAGE VEHICLE



(U) FIGURE 4-21
MACH 12 SUBSONIC TURBOJET CONFIGURATION OPTION





Alternate Engine Location

Alternate Engine Location Limitations

1. Exhaust Impingement on Control Surfaces.
2. Aerodynamic characteristics distorted by large pylons.
3. Poor stability and control with one engine out.

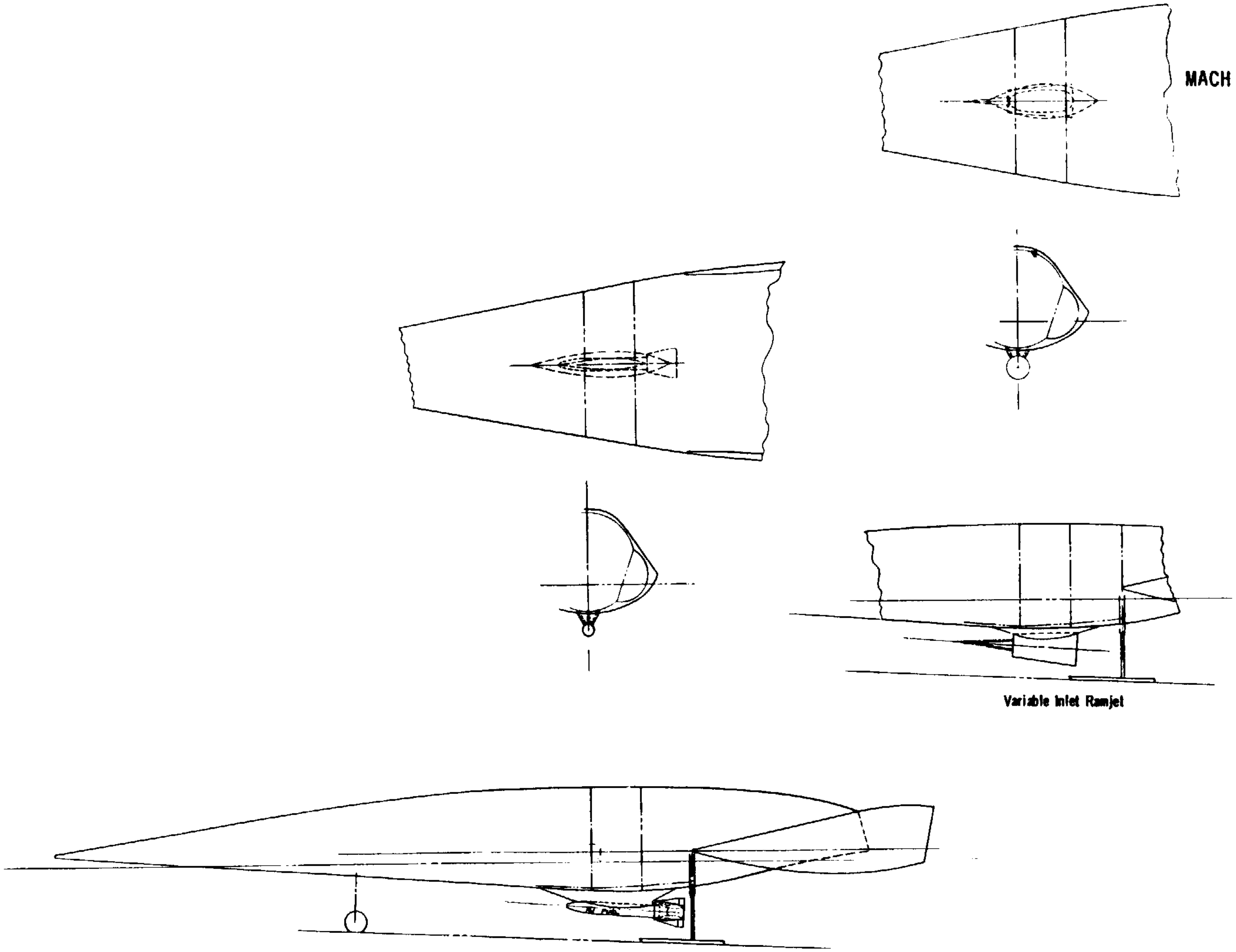
Baseline Modifications

1. Subsonic turbojet installation
   a. Structural beef-up of bulkheads
   b. Support fittings for engine nacelle
   c. Bladder type JP fuel tank inside LOX tank
   d. Fuel line to engines. Line is external or directly under shingles.
   e. Engine controls to cockpit
   f. Beef-up longeron at thrust mount
2. Landing gear
   a. Structural beef-up of bulkheads
   b. Add fixed main landing gear (skids retracted)
   c. Replace nose landing gear with extendible nose landing gear
3. Engine considerations
   a. Required uninstalled thrust per engine approximately 15,000 pounds (33,000 kilograms). Minimum T/W = 0.60 (installed thrust).
   b. May or may not have afterburners to meet thrust requirements.
   c. Subsonic operation only.
   d. Fixed nozzle diverts exhaust as much as 15° for CG thrust alignment.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

(U) FIGURE 4-22
MACH 12 RAMJET TESTBED CONFIGURATION OPTION



Variable Inlet Ramjet

Fixed Inlet Ramjet
(Missile Application)

Baseline Modification for RJ

1. Structural beef-up of bulkheads for ramjet engine supports.
2. Add BL intercostals for engine support fittings and fore and aft loads.
3. Add local fittings to bulkheads for side load reactions from sway braces.
4. Modify shingles to accommodate sway braces, engine support structure, and fairing.
5. Beef-up shingles on aft lower fuselage for increased sonic fatigue strength.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

4.2.2 (U) STRUCTURAL ARRANGEMENT - The structural arrangement employed for the basic Mach 12 aircraft and the changes required to provide for the selected research options are discussed in the following paragraphs.

4.2.2.1 (U) Basic Vehicle - An efficient, reliable, structure has been provided for the basic research aircraft by use of an external thermal protection system (TPS), cryogenic fuel tankage which is integral with the primary structure, direct load paths, and efficient materials. Wherever practical, conventional materials and state-of-the-art construction methods have been used to minimize cost and development risks. The basic aircraft structure arrangement is illustrated in Figure 4-23. Important thermal/structural details are described below.

(U) Since the Mach 12 research aircraft is an all-body configuration, the fuselage is the major structural component. All major systems, except control surfaces, are located in the fuselage and all loads applied to the aircraft have a direct effect on the fuselage design. The fuselage is of stiffened skin construction with integral multi-bubble fuel tanks. Major bulkheads which separate the fuel tanks and the cockpit enclosure are primary load redistribution members. Fuselage rings are located at 12 inch (30.5 cm) spacing and major longerons are located at tank bubble intersections. The rings and longerons are located on the external surface of the fuselage skin. This provides a smooth internal tank surface for application of the cryogenic insulation (discussed below). The longerons and rings support the external TPS and have internal, interconnected passages for supplying the water which is essential to the water-wick TPS operation. Longitudinal internal shear webs connecting tank bubble intersections redistribute tank fuel loads, carry primary fuselage shear, stabilize longerons, and serve as fuel baffles. Extensive use is made of commercially available aluminum alloys in the fuselage primary structure.

(U) Internal insulation and vapor barriers are used in all cryogenic tanks to minimize heat input to the propellants and prevent cryo-depositing of air on the external surface. A cross-linked Polyvinylchloride foam insulation is used in the liquid oxygen tank and a Polyurethane foam insulation is used in the liquid hydrogen tank. The insulation, in the form of tiles, is bonded to the tank walls with polyimide adhesives. A multi-layer FEP Teflon coated Kapton-H liner is used in all cryogenic tanks to minimize leakage of propellants into the cryogenic insulation. The use of a multi-layer system improves the reliability. Impregnation of the cryogenic insulation with propellants reduces the effectiveness of the insulation and can lead to an increase in boil-off and cryopumping on the tank external surface.

(U) The fuselage primary structure is protected from the thermal environment by a water wick TPS. This system is comprised of an external surface shingle used to radiate heat energy back to the atmosphere, a layer of Dynaflex insulation wrapped in a reflective foil, and an encapsulated water saturated silica blanket attached to the primary structure. The water wick TPS details are shown in Figure 4-24 and its operation is discussed in Section 4.6. The shingle is a single faced beaded sandwich with continuous supports at each longitudinal edge and along its longitudinal centerline as illustrated in Figure 4-24. The shingles are designed to carry only local airloads. Expanding joints are provided at the shingle edges to permit free thermal expansion and reduce thermal stresses.

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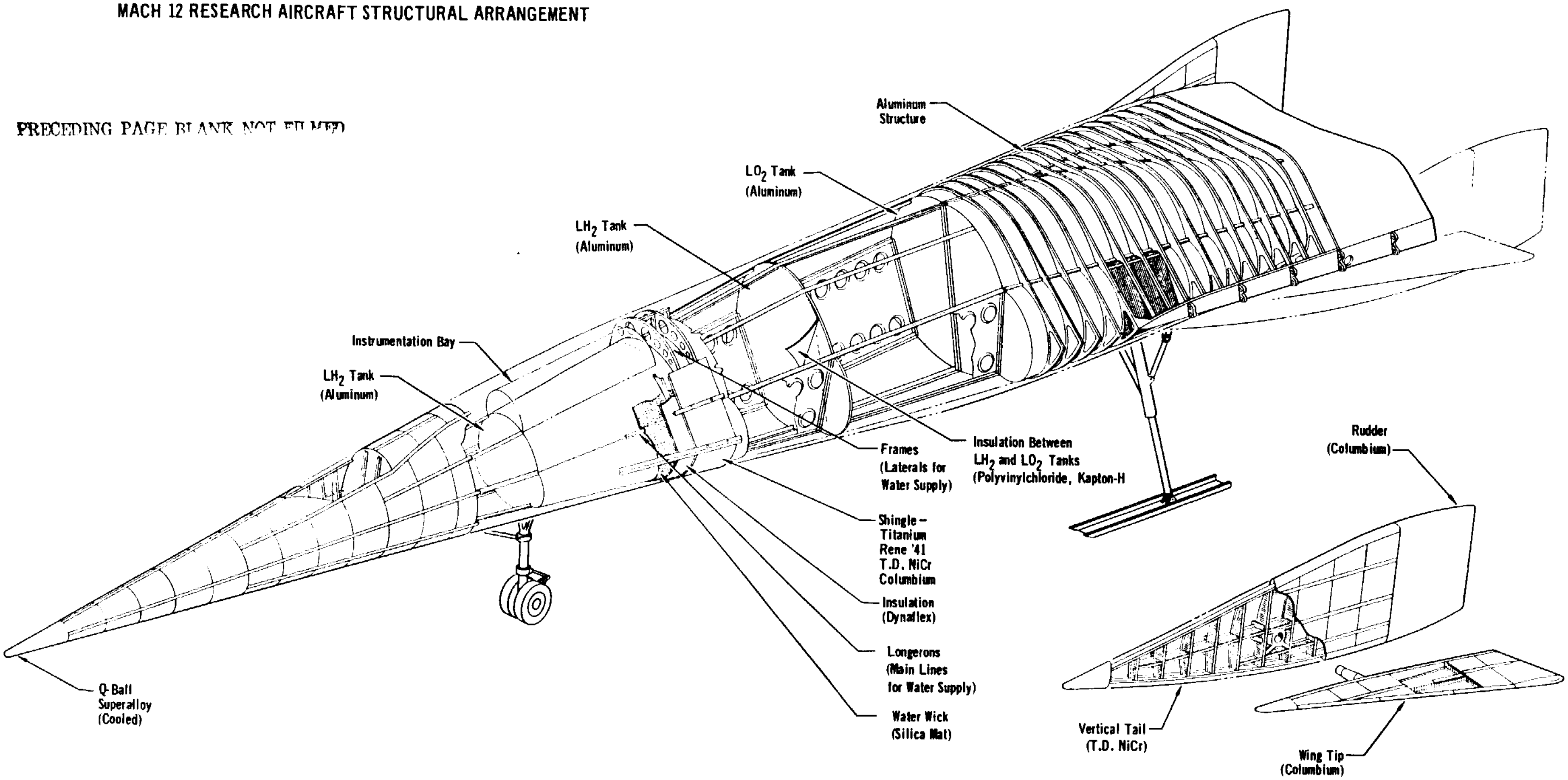
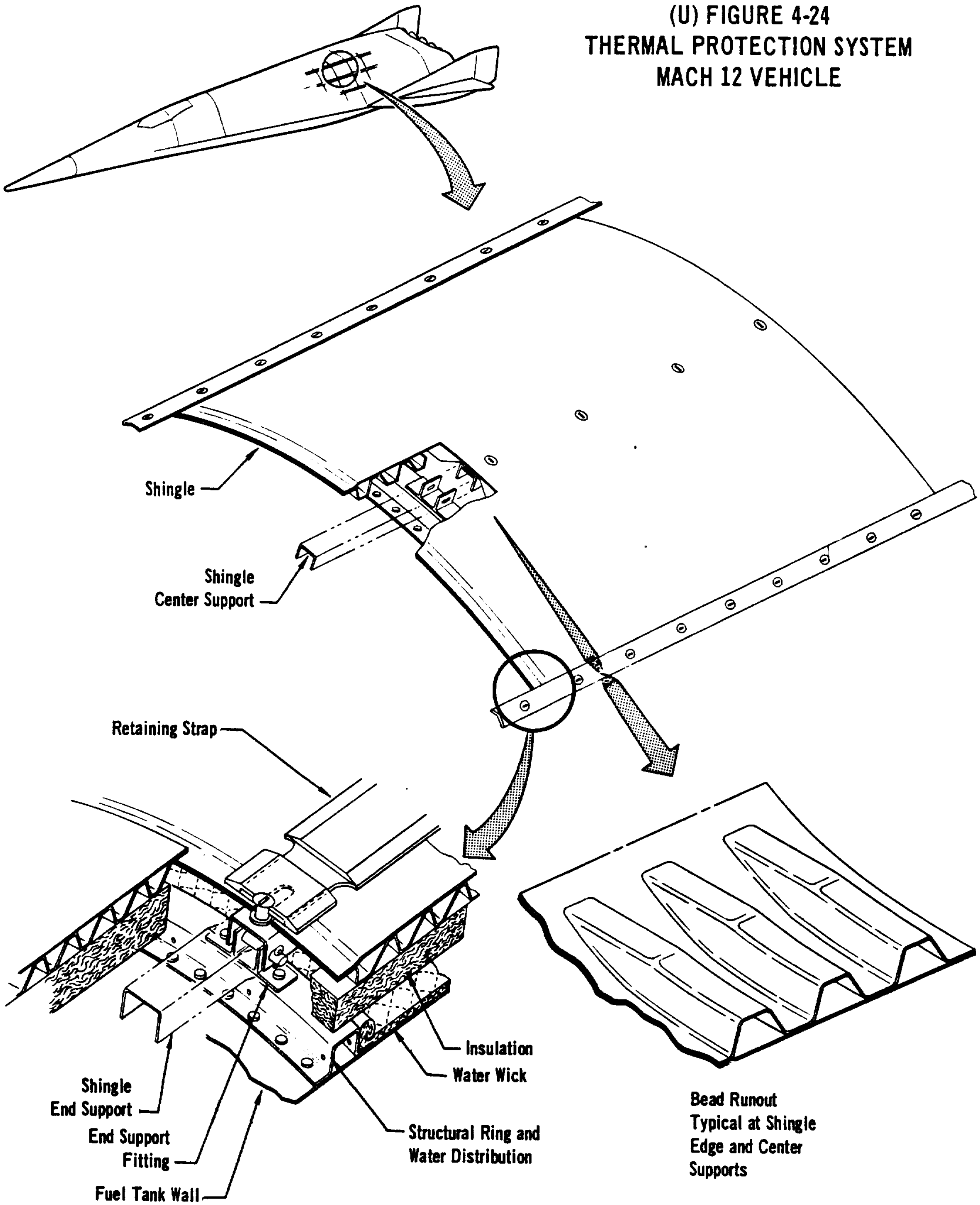
(U) FIGURE 4-23
MACH 12 RESEARCH AIRCRAFT STRUCTURAL ARRANGEMENT



(U) FIGURE 4-24
THERMAL PROTECTION SYSTEM
MACH 12 VEHICLE



(U) Landing Gear - The main landing gear consists of a cantilever air-oil shock strut and a landing skid. The gear retracts into the fuselage with the skid flush with the external surface. The shock strut is designed to restrict landing loads to no more than 3 g at maximum design sink speed. Material used in the strut and support structure is a high strength steel while the skid is a superalloy, T.D. NiCr. The skid heats up during flight to temperatures approaching the adjacent shingle temperatures. However, just prior to landing the temperature of the skid is at a maximum of 700°F (645°K).

(U) Control Surfaces - The vertical tail, rudder, and movable tip controls on the research aircraft are constructed of structural beams covered with thin skin or shingles for local air load support. This hot structure concept was selected based on previous MCAIR studies on "operational" hypersonic aircraft which indicated that uninsulated control surfaces provide a combination of weight and drag that give the best vehicle performance. The vertical tails are made of T.D. NiCr and the rudders and movable tips are constructed of coated columbium alloy. However, several options are available when selecting the type of construction and the materials to be used on these surfaces. Figure 4-23 shows the general structural arrangement of one concept of the tips. The movable tip controls provide an excellent test bed for research and demonstration of high temperature structure, materials, coatings, and construction combinations. Redundant load paths for the majority of the structure provides the opportunity to do structural testing with high confidence in aircraft safety.

(U) Nose Cone - The nose tip temperature of 5100°F (2540°K), based on a nose tip radius of 2.5 inch (6.35 cm), requires that it be fabricated out of a ceramic material or a regeneratively cooled metal structure. There are no available structural metals that have a reusable capability above tantalum, which is limited to 3200-3400°F (2030-2140°K). One applicable concept is a ceramic cap made of a maze of zirconia rods, blocks, and tubes held in place with zirconia binder. This concept was developed for the ASSET program and is flight proven. The other option is the "Q-Ball" type nose that is designed to be used as an attitude sensor. For the Mach 12 aircraft the Q-Ball is regeneratively cooled to a temperature that will allow use of superalloy metal. The difference in weight or cost of these two available concepts has only a negligible effect on the aircraft. The Q-Ball concept is incorporated in the research aircraft because of its attitude sensing capability.

4.2.2.2 (U) Structural Modifications (Research Options) - The arrangement of the major structural elements and load paths of the basic aircraft have been selected considering the modifications that would be adapted to the aircraft. Location of major bulkheads and major longerons have been established to provide a sound structure in the basic aircraft and to provide for ease of modification for incorporating the research options.

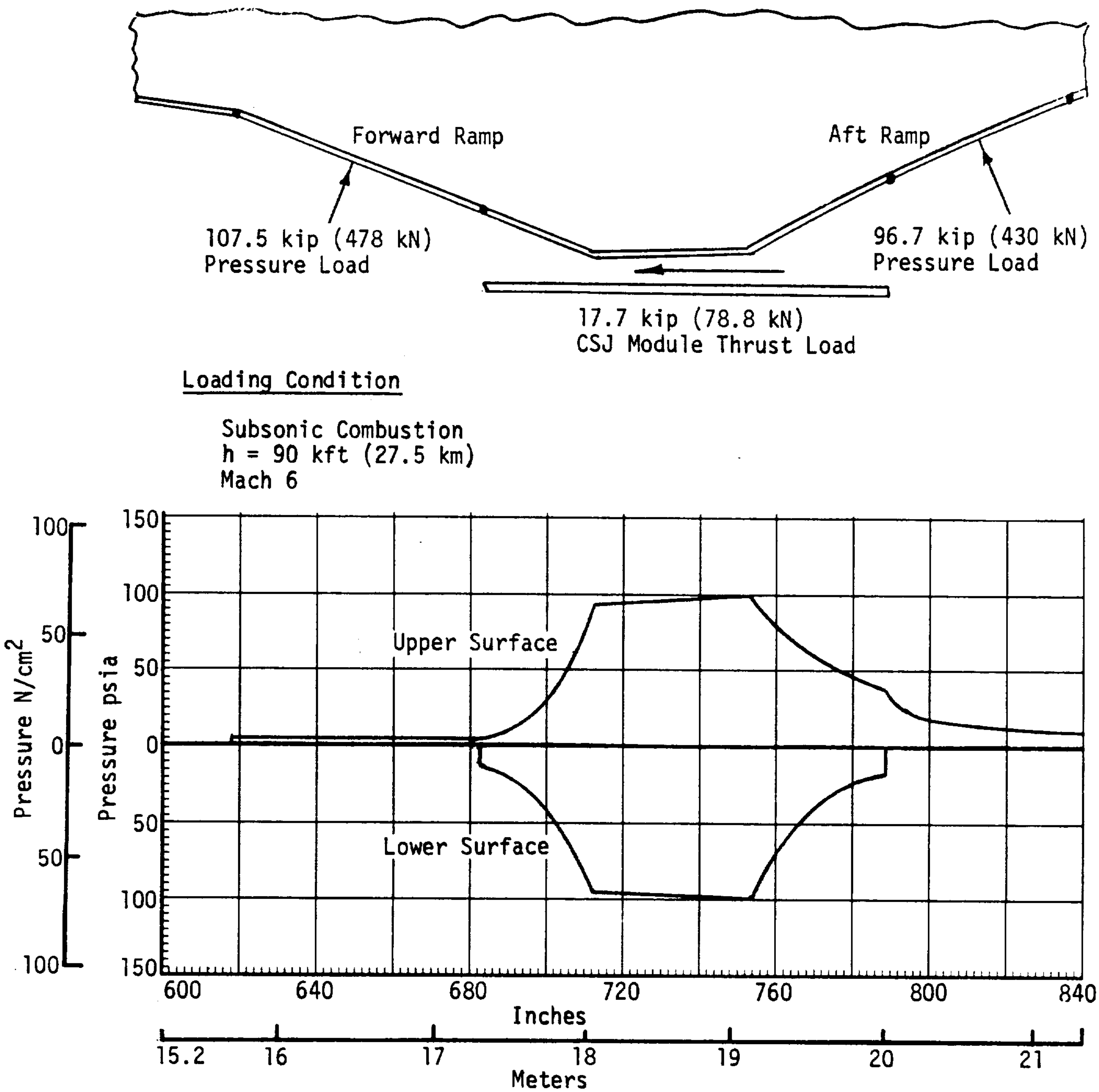
(U) Horizontal Takeoff (HTO) Option - The primary structural modifications for this option are in the fuselage bulkheads,to which the dolly gear is attached, and in the nose gear. Additional loading conditions for taxi and take-off require local strengthening of the gear support. The 2.0 g taxi loads in combination with gear reaction resulting from rocket gimballing yields a main gear load of about 175,000 lb (79,400 kg) which is greater than the landing load, should the aircraft have to land with the dolly attached. The shift in gear load to a bulkhead further forward requires a major redesign of the bulkhead. Local fittings in the fuselage structure for attachment of the main gear dolly will be designed to withstand surface temperatures. The water wick thermal protection system will be modified in the local area to compensate for the increased direct heat path. The design loads for the nose gear and the nose gear support structure resulting from 2.0 g taxi and horizontal takeoff at maximum design gross weight are less than the design loads for landing. Thus, no weight increase is required to the nose gear support structure for the HTO option. However, the strut length is increased to increase the aircraft angle of attack for HTO and this does impose a strut weight penalty.

(U) Vertical Takeoff (VTO) Option - Local modification of the fuselage structure made for the horizontal takeoff dolly will suffice for the launch cart support for the vertical takeoff option. The aft end of the aircraft will have to be modified locally to support the aircraft while in the vertical position. A forward support for the launch cart will also be added and will require redesign of a bulkhead to support a design normal load of 20,000 lb (88,900 N).

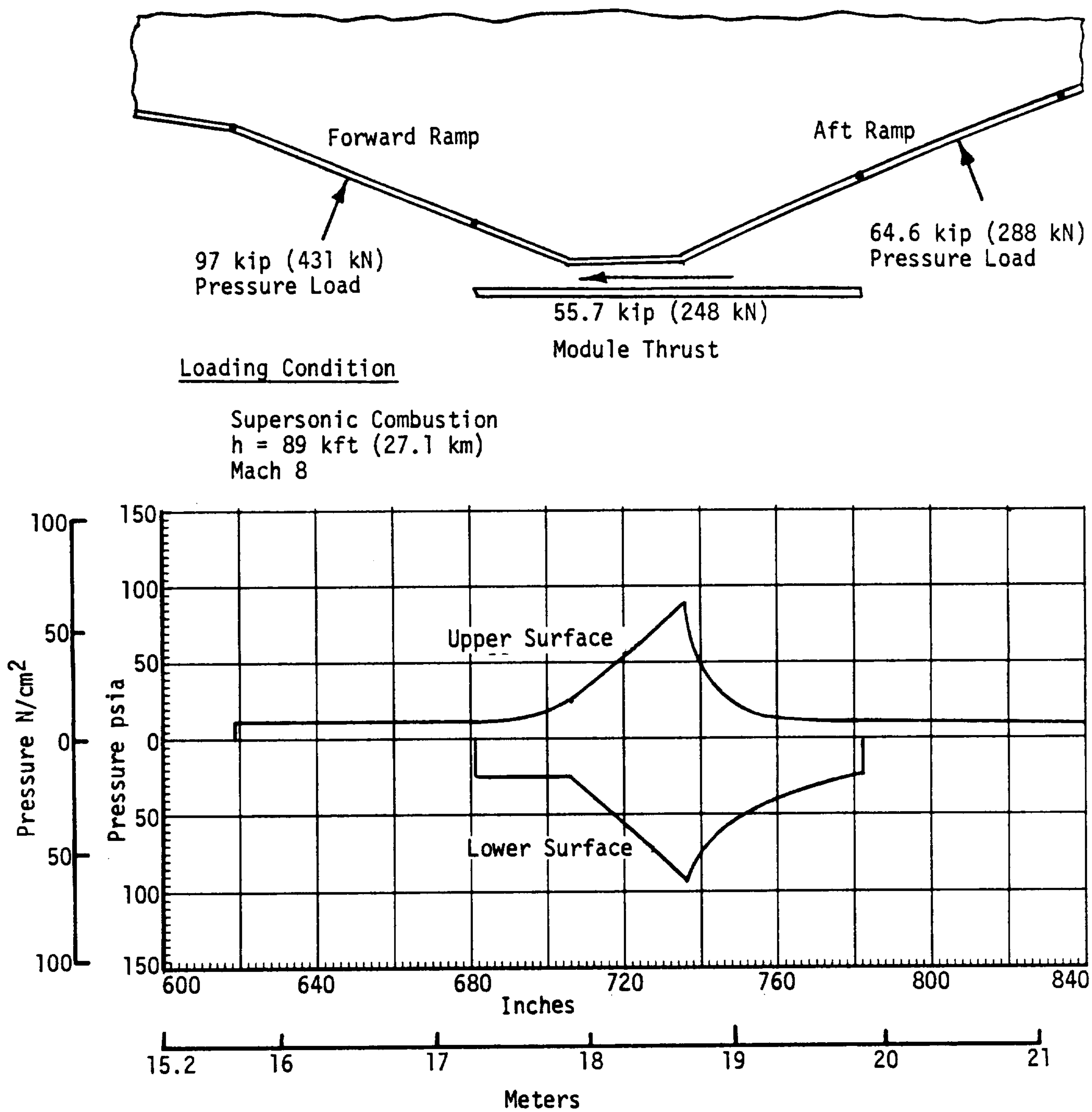
(U) Convertible Scramjet (CSJ) Option - Structural modification of the basic aircraft is confined to the lower surface between the forward edge of the third ramp and the aft edge of the movable part of the nozzle. Study of the flight profile showed that the critical design condition (i.e., highest pressure) occurs during a 5.0 g maneuver at Mach 6, at an altitude of 90,000 ft (27.4 km). This condition results in the loads and pressure distribution shown in Figure 4-25. The CSJ module design is based on the results of the analysis and design studies conducted for a CSJ powered cruise aircraft reported in Reference 8. The basic concept is regeneratively cooled structure with both the coolant tubes and the structural frames made of T.D. NiCr. An alternate concept, in which the T.D. NiCr frames are replaced by titanium frames and water wick cooling system, weighs nearly the same for the research aircraft mission requirements. However, the alternate concept would have a lower weight if the design pressure requirements were increased. The module is completely supported by links, hinges, and actuators which allow relative expansion between the module and fuselage structure while allowing vertical movement of the module and restraining fore and aft and side inertial loads. A schematic is shown in Figure 4-13. The foreward and aft ramps are hinged and actuated to cooled T.D. NiCr structures which have been shown to be the lowest weight (Reference (9)). There are, however, other available options since the surface temperature is within the limits of refractory metals (tantalum alloys). Also applicable are ceramic and ablative coatings on various metal substructures. The loads from the ramps and module are transmitted to four major bulkheads which will require redesign for concentrated loads.

(U) Scramjet (SJ) Option - Structural modifications to the basic aircraft for the SJ option are very similar to those required for adaptation of the CSJ. The load paths are the same; however, the magnitude and distribution of the pressure loads are somewhat different. The maximum or critical pressure condition occurs at Mach 8 at an altitude of 89,000 feet (27.1 km) during a 5.0 g maneuver. The resulting loads and distribution are shown in Figure 4-26; however, the temperatures occurring during this condition are somewhat less than the maximum design temperatures. Structure of the SJ module and the movable ramps are conceptually the same as the CSJ module.

(U) FIGURE 4-25
CONVERTIBLE SCRAMJET PRESSURE LOADS



(U) FIGURE 4-26
SCRAMJET PRESSURE LOADS



(U) Thermal Protection System (TPS) Option - The section of the fuselage provided for research on thermal protection systems can be used for a wide variety of TPS concepts with no revisions to the primary structure because sufficient space was provided between the moldline heat shield and the fuselage shell in the basic fuselage design. Were this space not provided,a major revision in structure or a moldline extension would need to be made for a thermal protection system with a thickness greater than 1 to 1-1/2 inches (2.54-3.8 cm).

(U) Armament (ARM)Option - Modification of the primary structure for addition of the armament package requires minor redesign of the major bulkheads for the missile tube attachment. Primary considerations in modifying the structure of the basic aircraft are the inertial loads of the missile tubes, missile ejection thrust reaction, and internal temperatures. The inertial load is approximately 18,000 lb (80,000 N) and the thrust reaction from ejection of the missiles is 14,000 lb (62,200 N) which results in a missile/aircraft relative velocity of 100 fps (30.5 m/sec) at separation. This reaction is transmitted into the major longerons through intercostals between major bulkheads.

(U) Staging (STG) Option - Modification to the primary fuselage structure will not be required for the adaptation of a second stage. The major second stage loads will be transmitted through the hardpoints provided for attachment of the research aircraft to the C-5A launch vehicle. Separation forces at these hard points are much less than the 160,000 lb (712,000 N) design load that results from the C-5A taxi condition.

(U) Subsonic Turbojet (TJ) Option - Structural modifications of the basic aircraft include redesign of the main bulkheads for the engine and inlet mounting supports with two bulkheads requiring major changes. A revision of the lower surface structure will also be required for the 30,000 lb (133,500 N) thrust reaction.

(U) The JP fuel used in the TJ engine is stored in the refurbished LO_2 tank which does not require any structural change since the density of the JP and its required pressure are lower than those of LO_2.

(U) The modification made to provide horizontal takeoff capability will also be required since the launch cart is also used for this option.

(U) Ramjet (RJ) Option - The ramjet thrust and inertial loads are carried through the same supports and mounts that are provided in the SJ modification. The loads are low and the modifications are considered structurally minor.

4.3 AERODYNAMICS

(U) The aerodynamic characteristics for the basic Mach 12 vehicle and several configuration options are presented in the following sections. Aerodynamic efforts have been primarily directed toward accomplishing three Phase III objectives:

- o Substantiation of the lift and drag values employed in vehicle performance analyses
- o Determination of the basic vehicle static stability and control characteristics
- o Analysis of the unaugmented handling qualities of the basic vehicle.

(U) The lift and drag characteristics employed in determining the performance data presented in Section 4.5 are shown to be well substantiated by two additional prediction techniques (Gentry and Harris Programs) introduced in the Phase III analyses. Static stability and control characteristics for the basic vehicle, both longitudinal and lateral-directional, are defined throughout the Mach number regime. They show the aircraft to be statically stable and aerodynamically controllable in all modes. A redundant three-axis control augmentation system will of course be required to tailor handling qualities to the desired levels. However, the unaugmented handling qualities for the bare airframe are sufficiently good in most instances to assure a safe termination of the mission in spite of an inoperative augmentation system.

4.3.1 (U) LIFT AND DRAG CHARACTERISTICS - The lift and drag coefficients employed in the performance analyses of the Mach 12 rocket vehicles are defined as

$$C_L = C_{L_\alpha}\alpha$$

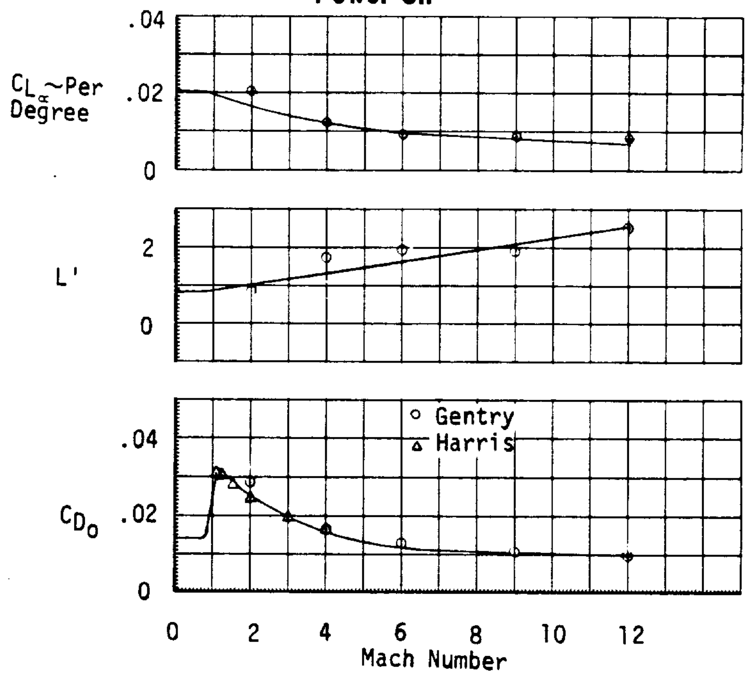
$$C_D = C_{D_o} + L'C_L^2$$

where the lift curve slope (C_{L_α}), the zero-lift drag coefficient (C_{D_o}), and the induced drag factor (L') are determined as described in Section 3.2.1.

4.3.1.1 (U) Basic Vehicle - The values of L' and C_{L_α} utilized in the basic vehicle performance analysis are shown in Figure 4-27, where they are compared with the values obtained from the Gentry Arbitrary Body Program. The agreement obtained substantiates the Phase II methods of predicting the values of L' and C_{L_α} which are employed in determining the Phase III vehicle performance.

(U) The values of C_{D_o} employed in the performance analysis of the basic vehicle are also shown in Figure 4-27. Also shown are the results obtained from the Gentry and Harris computer programs. The Harris C_{D_o} values shown include the same skin friction, base, and vertical tail toe-in drag contributions as employed in the basic Phase II method of analysis since only wave drag is computed in the program. The fuselage geometry employed as input data to the Harris program is terminated at 96% body length to account for flow separation effects as described in Section 3.2.1.

(U) FIGURE 4-27
MACH 12 ROCKET
LIFT AND DRAG CHARACTERISTICS
Power On

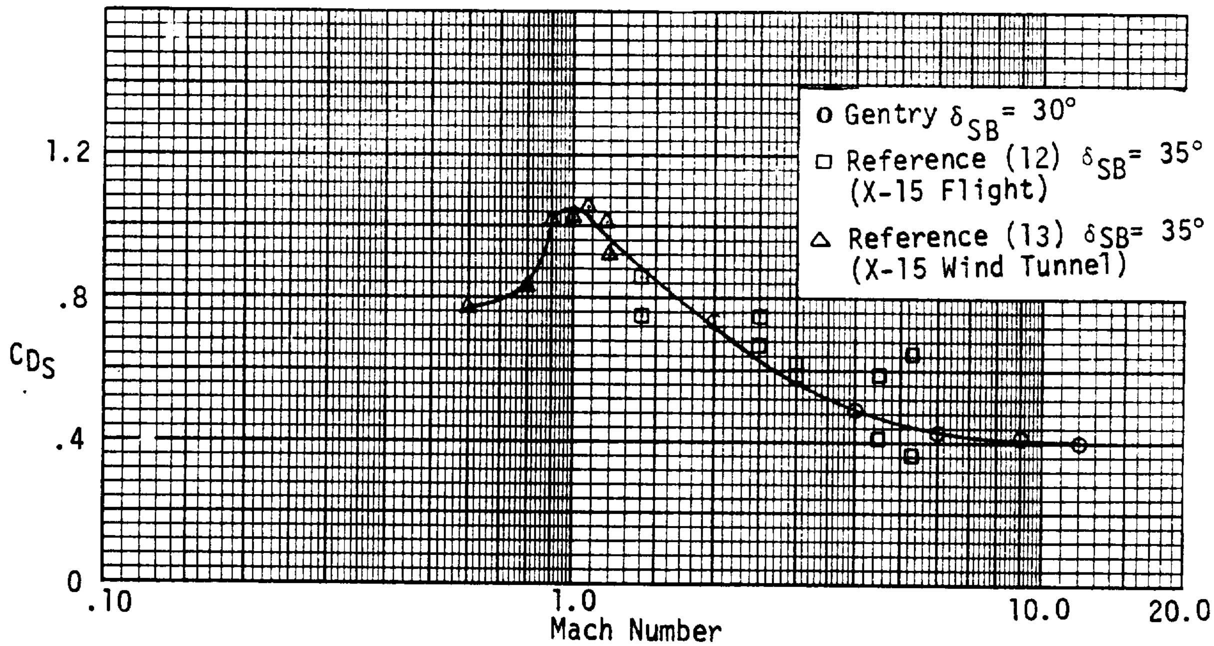


(U) The C_{D_O} values for the Gentry program include the same base and protuberance drag contributions as employed in the basic Phase II method of analysis since these are unaccounted for in the program computations. Although there are obvious differences in the three methods of computing C_{D_O}, the results indicate that the Phase II drag method employed in the basic vehicle performance analysis yields estimates substantially consistent with other accepted prediction techniques.

(U) Figure 4-28 presents the effect of speed brake deflection in terms of C_{D_S} as predicted by the Gentry Program together with measured X-15 test data obtained from References (12) and (13). A maximum 30° symmetrical outboard deflection of the rudder panels is employed for speed brakes on the vehicle. The predicted values compare favorably with the measured X-15 data. The speed brake effectiveness employed in the performance analysis is shown in Figure 4-29 together with the Gentry predictions for the maximum 30° deflection. To achieve the performed values used in the mission studies described in Section 6, a deflection of approximately 15° is required.

4.3.1.2 (U) Configuration Options - Using the same methods as employed for the basic vehicle, estimated values of C_{D_O} were obtained for the various modifications and are presented in Figure 4-30. These values are employed in determining the variation in vehicle performance to be expected when incorporating each of the configuration options studied. The values of L' and C_{L_α} are assumed to be unchanged by the addition of the various options to the basic configuration, since the vehicle aspect ratio is unchanged, and are therefore the same as presented in Figure 4-27. Performance changes due to aerodynamic effects are reflected solely by variation

(U) FIGURE 4-28
MACH 12 ROCKET
SPEEDBRAKE EFFECTIVENESS



in C_{D_O}. Because of the small values of lift required in performing the design missions, the effects of induced drag are also small and the foregoing assumption is valid.

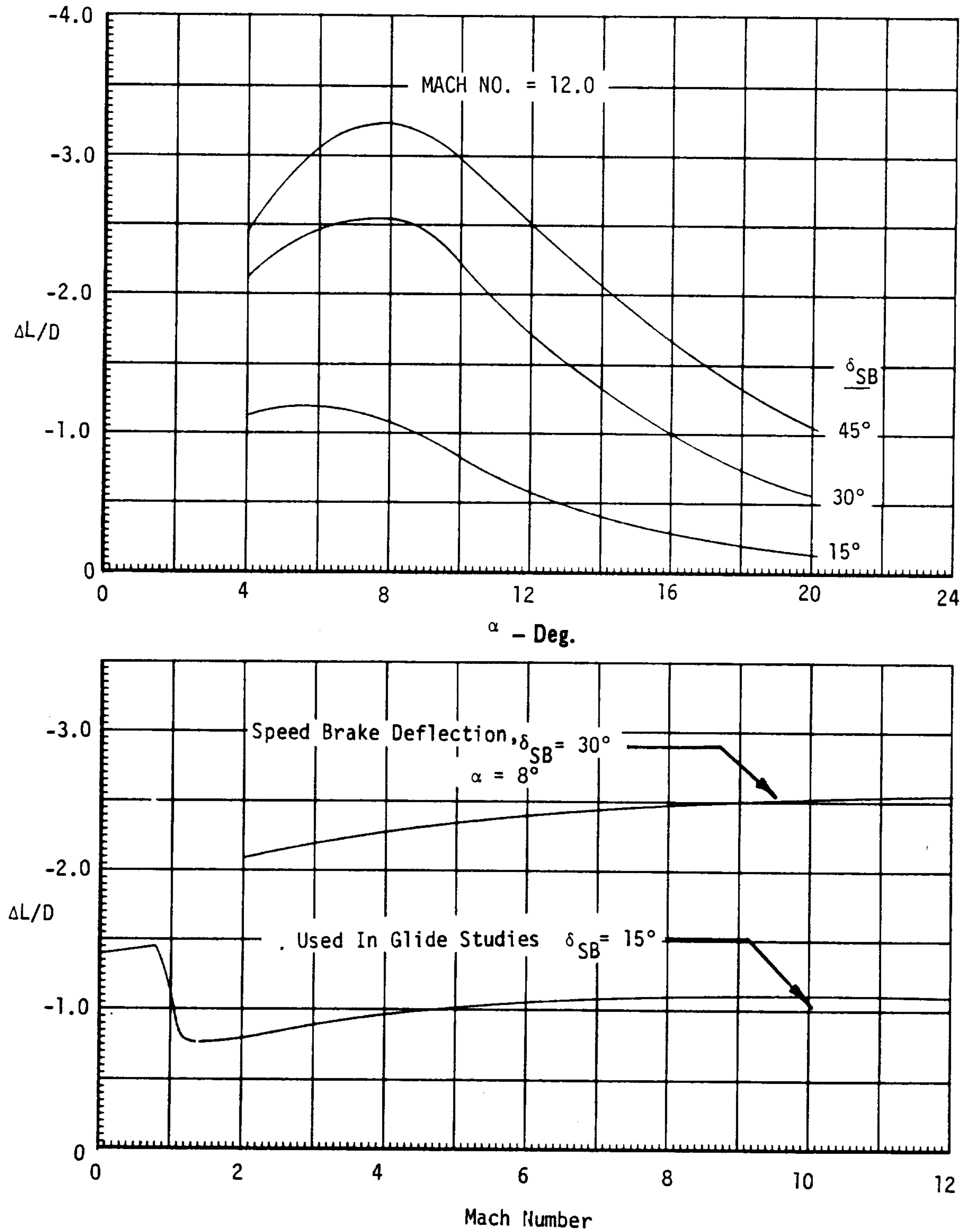
4.3.2 (U) STABILITY AND CONTROL CHARACTERISTICS - The longitudinal and lateral-directional static stability characteristics of the basic vehicle were obtained at Mach numbers of 2.0, 4.0, 6.0, 9.0, and 12.0 using the Gentry program. The longitudinal static stability characteristics are presented in Figures 4-31 through 4-35. The reference moment center employed is located at 53.4% of the body length. Also shown on these plots are selected climb and glide trim points corresponding to an anticipated aft center of gravity location of 64% length (see Figure 4-102 for cg envelope). The delta tip controls are shown to be quite effective in providing pitch control.

(U) Figure 4-36 presents a tabulation of the pertinent longitudinal stability derivatives for the trim points illustrated in Figures 4-31 through 4-35. They were obtained from the Gentry analysis using a body axis system. They are utilized in the dynamic stability analysis discussed in Section 4.3.3.

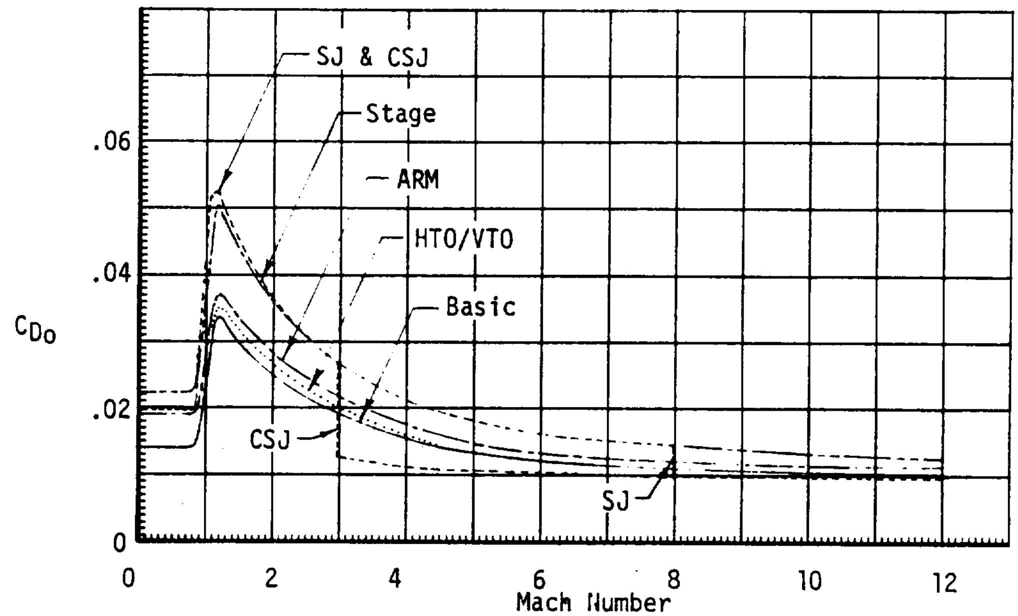
(U) The variation of neutral point with Mach number for the trimmed vehicle is shown in Figure 4-37 for the design mission. It will be noted that minimum static stability, approximately 1% of vehicle length, occurs during the climb at a Mach number of 9.0 for the presently configured vehicle.

(U) The effect on the longitudinal static stability of symmetrically deflecting the rudders for use as speed brakes is shown in Figures 4-38 and 4-39. An Orthomat Plotter drawing of the vehicle with the speed brakes deflected 30° is shown in Figure 4-40. The analysis indicates that the use of the rudders as speed brakes is quite feasible inasmuch as the basic vehicle stability is not seriously

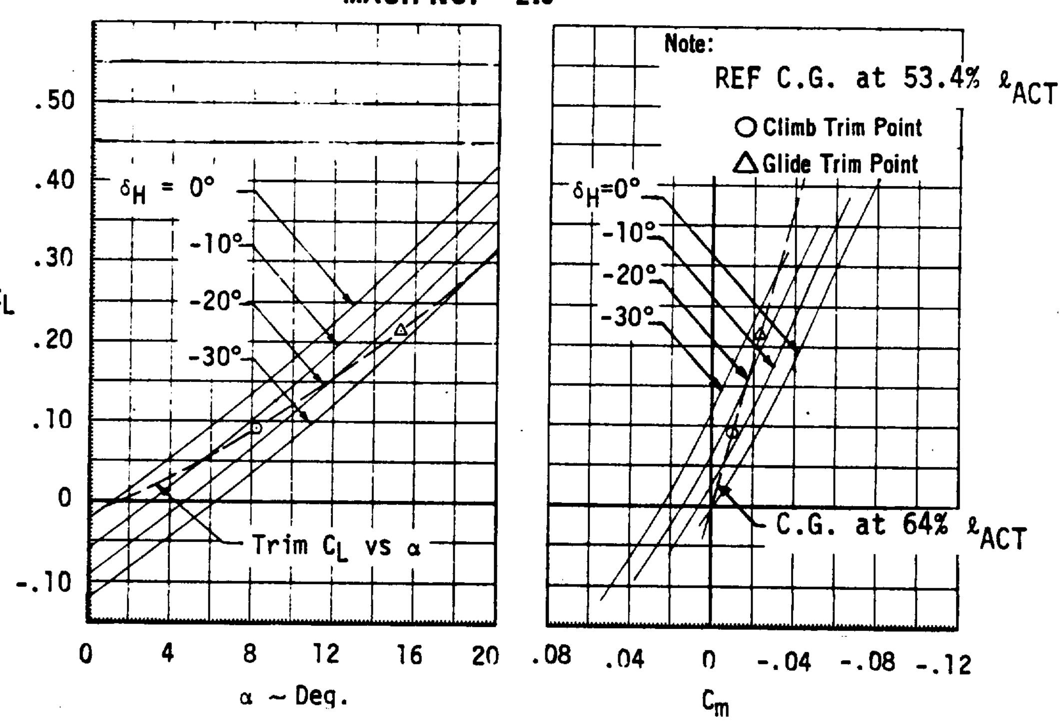
(U) FIGURE 4-29
EFFECT OF SPEED BRAKES
Gentry Data



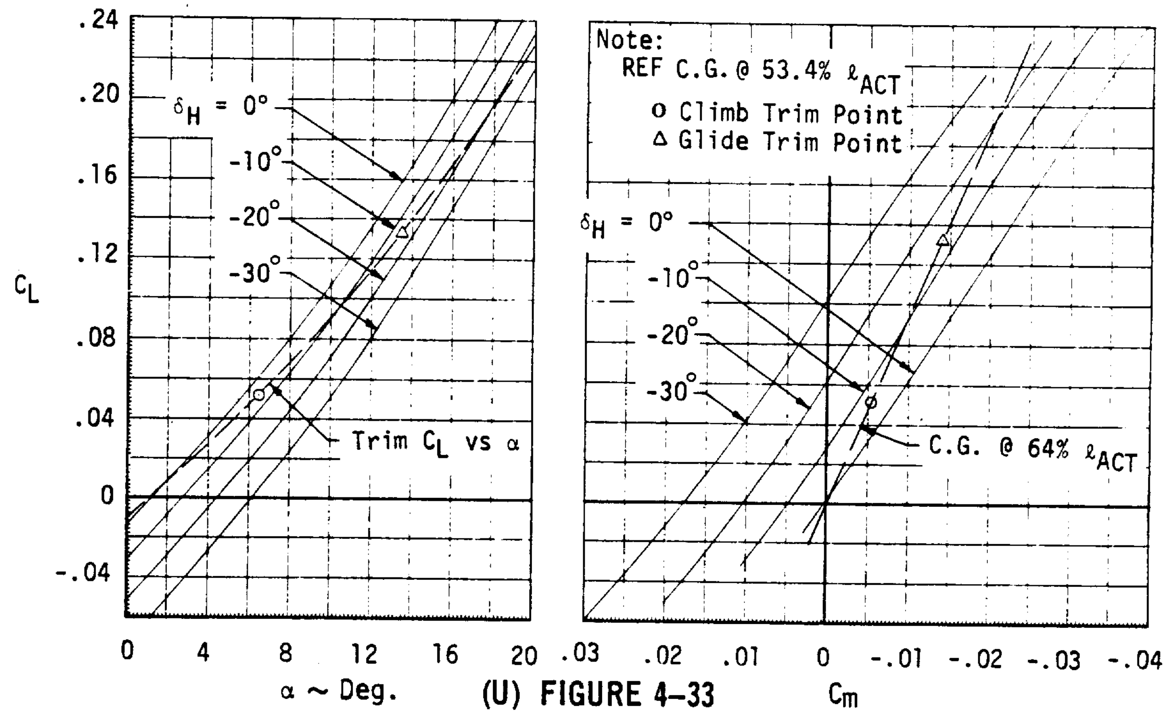
(U) FIGURE 4-30
MACH 12 VEHICLE
ZERO-LIFT DRAG, CONFIGURATION OPTIONS, POWER ON



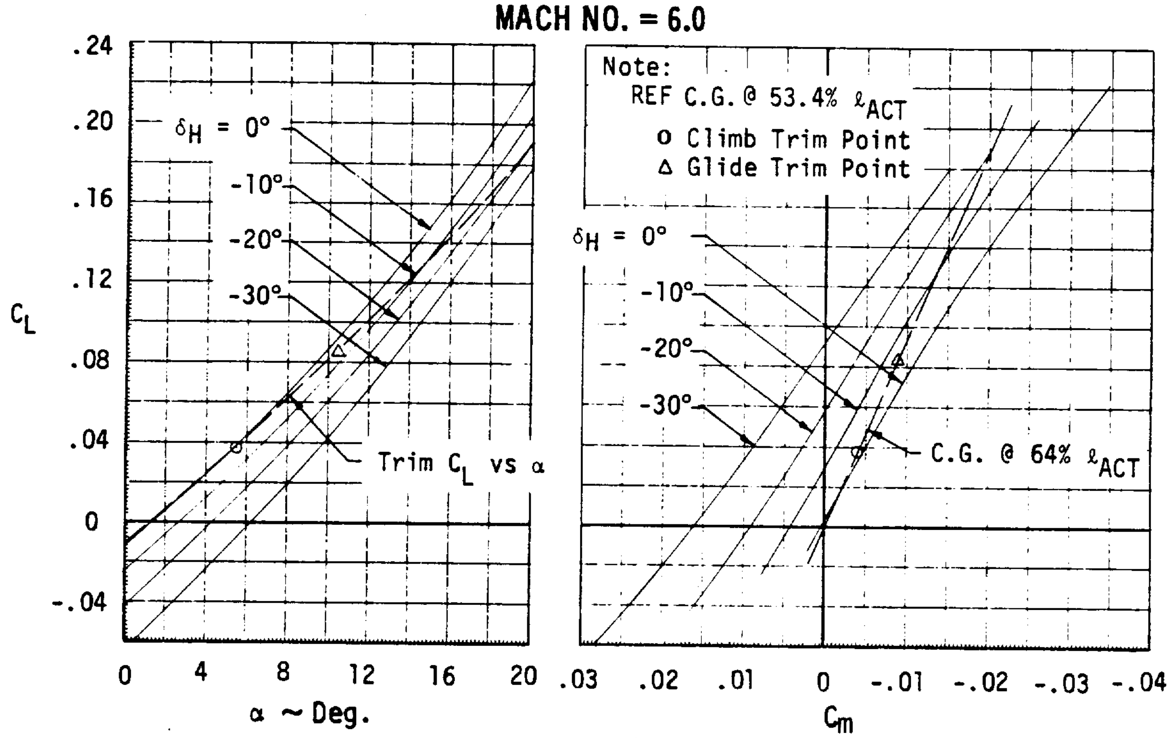
(U) FIGURE 4-31
MACH 12 ROCKET
LONGITUDINAL STATIC STABILITY AND CONTROL CHARACTERISTICS
MACH NO. = 2.0



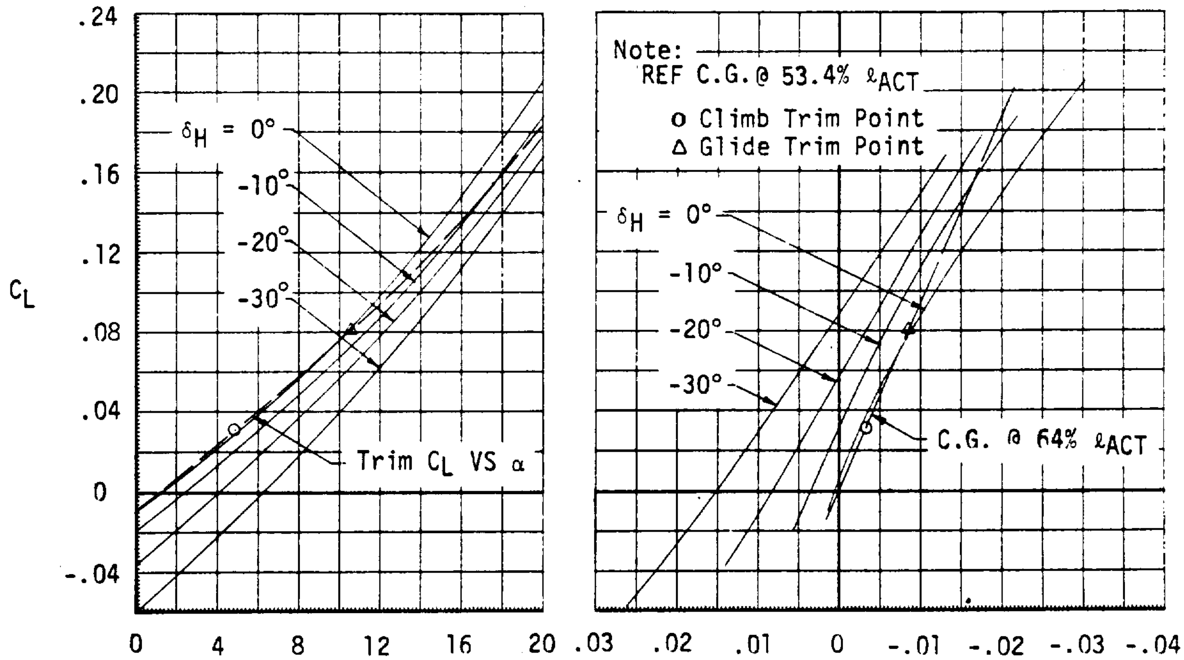
(U) FIGURE 4-32
MACH 12 ROCKET
LONGITUDINAL STATIC STABILITY AND CONTROL CHARACTERISTICS
MACH NO. = 4.0



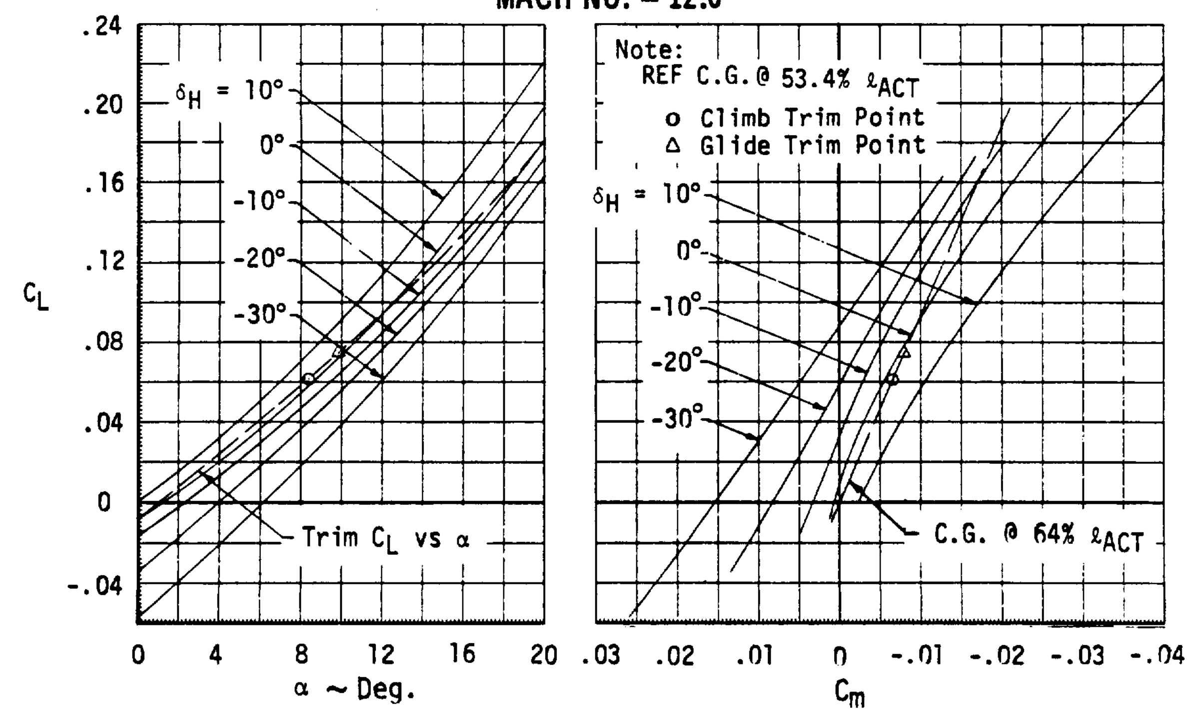
(U) FIGURE 4-33
MACH 12 ROCKET
LONGITUDINAL STATIC STABILITY AND CONTROL CHARACTERISTICS
MACH NO. = 6.0



(U) FIGURE 4–34
MACH 12 ROCKET
LONGITUDINAL STATIC STABILITY AND CONTROL CHARACTERISTICS
MACH NO. = 9.0



(U) FIGURE 4–35
MACH 12 ROCKET
LONGITUDINAL STATIC STABILITY AND CONTROL CHARACTERISTICS
MACH NO. = 12.0

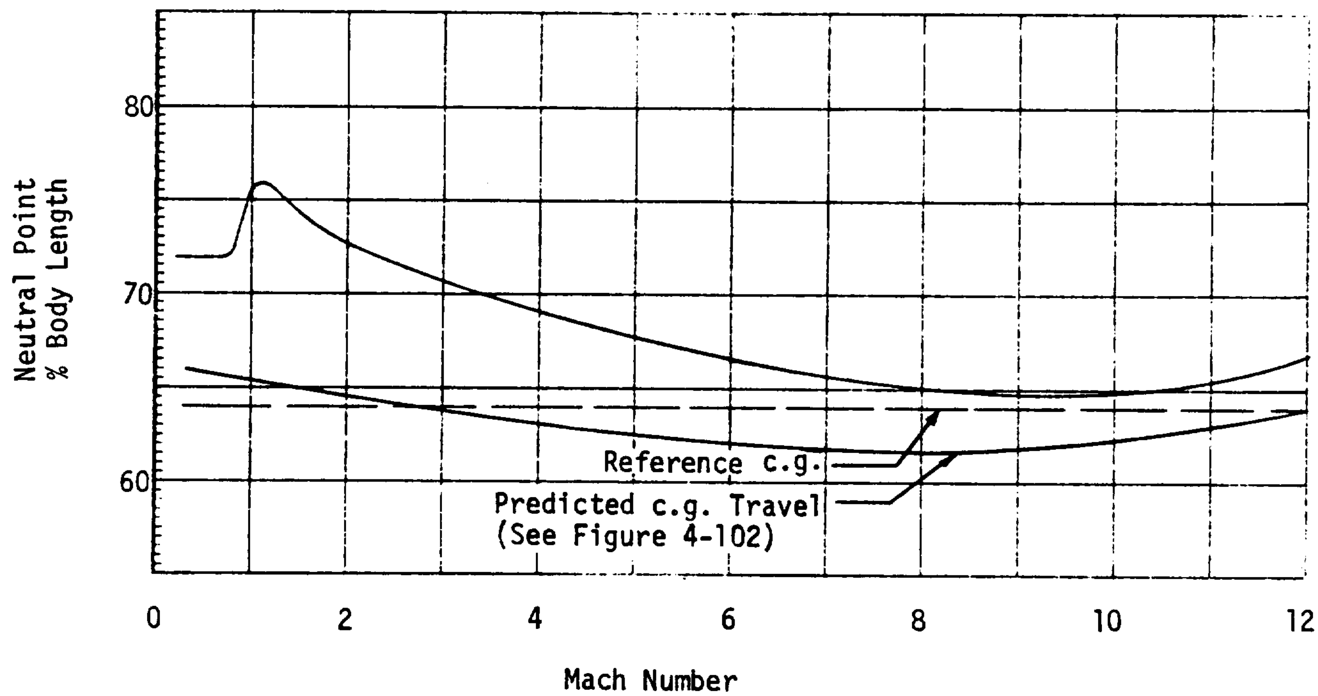


(U) FIGURE 4-36
MACH 12 ROCKET
LONGITUDINAL DERIVATIVES AT TRIM FLIGHT CONDITIONS

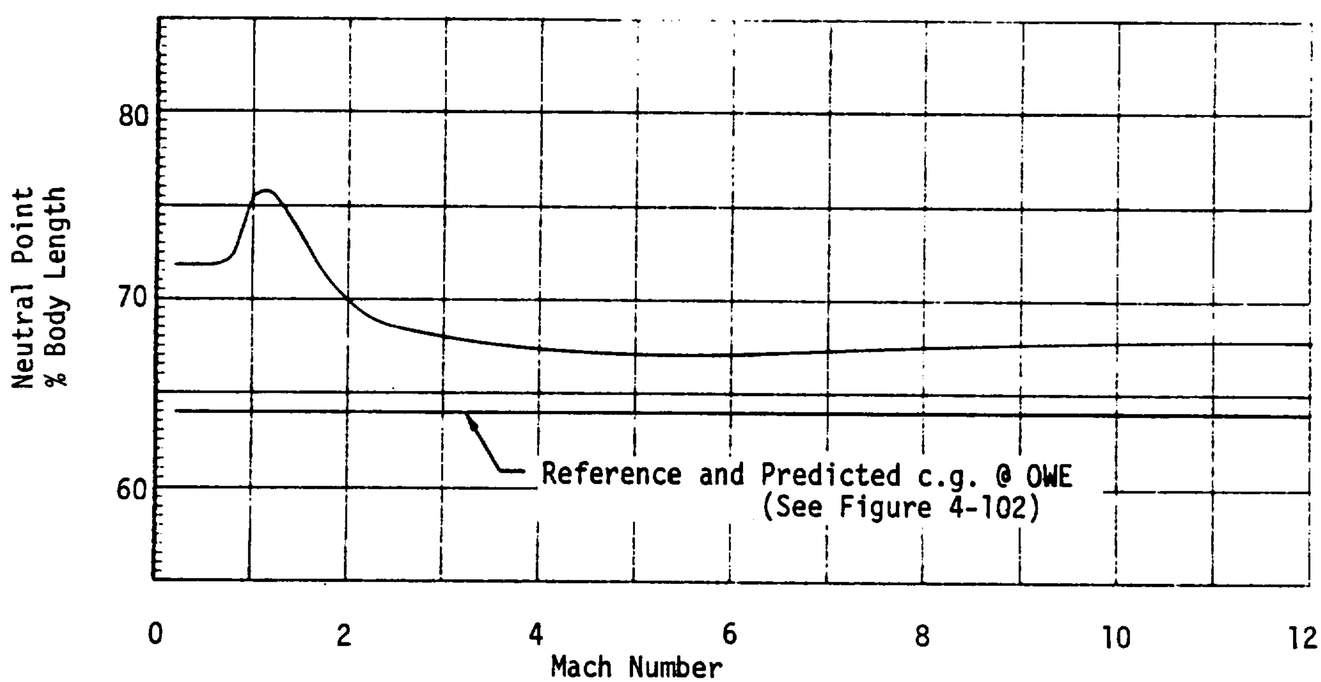
MACH NO.	2.0	4.0	6.0	9.0	12.0
			CLIMB		
PITCH DER.					
C_{A_α} /DEG	-.00033	.00004	-.00003	.00000	.00019
$C_{L\alpha}$ /DEG	.02213	.01241	.00990	.00885	.00957
$C_{N\alpha}$ /DEG	.02294	.01280	.01017	.00908	.01005
$C_{m\alpha}$ /DEG	-.00191	-.00064	-.00026	-.00006	-.00027
C_{mq} /RAD	-.13464	-.08136	-.06674	-.06094	-.06828
C_{A_q} /RAD	-.01807	-.01206	-.00642	-.00450	-.00846
C_{N_q} /RAD	.17693	.04020	.00608	-.00688	.01498
			GLIDE		
PITCH DER.					
C_{A_α} /DEG	-.00045	.00004	.00006	.00016	.00021
$C_{L\alpha}$ /DEG	.02459	.01560	.01156	.01084	.01037
$C_{N\alpha}$ /DEG	.02691	.01679	.01221	.01151	.01100
$C_{m\alpha}$ /DEG	-.00146	-.00053	-.00036	-.00039	-.00039
C_{mq} /RAD	-.18377	-.10967	-.07935	-.07690	-.07412
C_{A_q} /RAD	-.01396	-.01630	-.01230	-.01213	-.01129
C_{N_q} /RAD	.12260	.02986	.02414	.02932	.03022

(U) FIGURE 4-37
MACH 12 ROCKET
NEUTRAL POINT vs MACH NUMBER
TRIM FLIGHT CONDITIONS

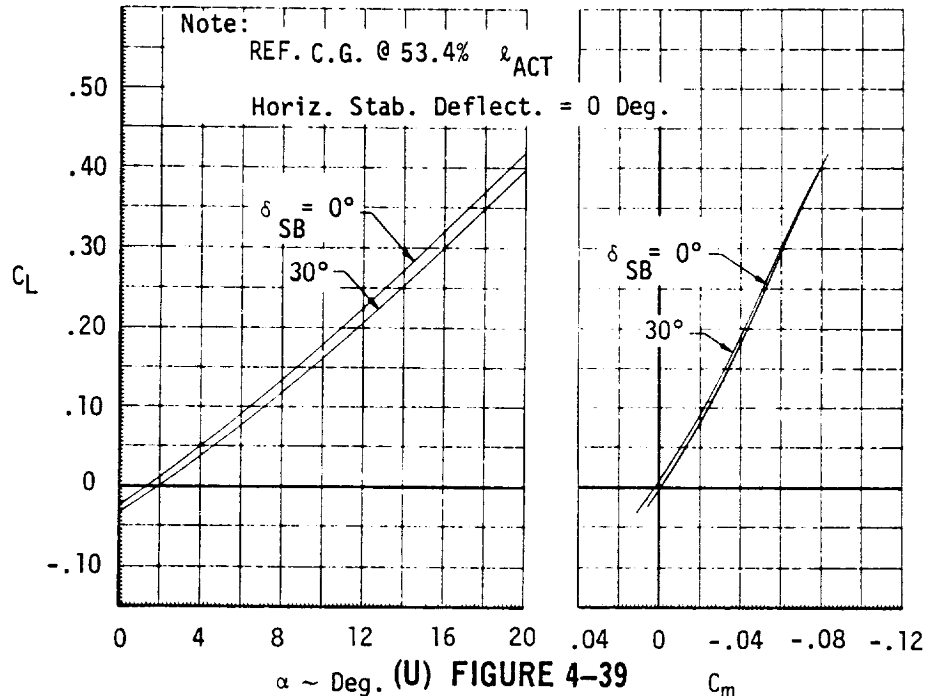
Climb



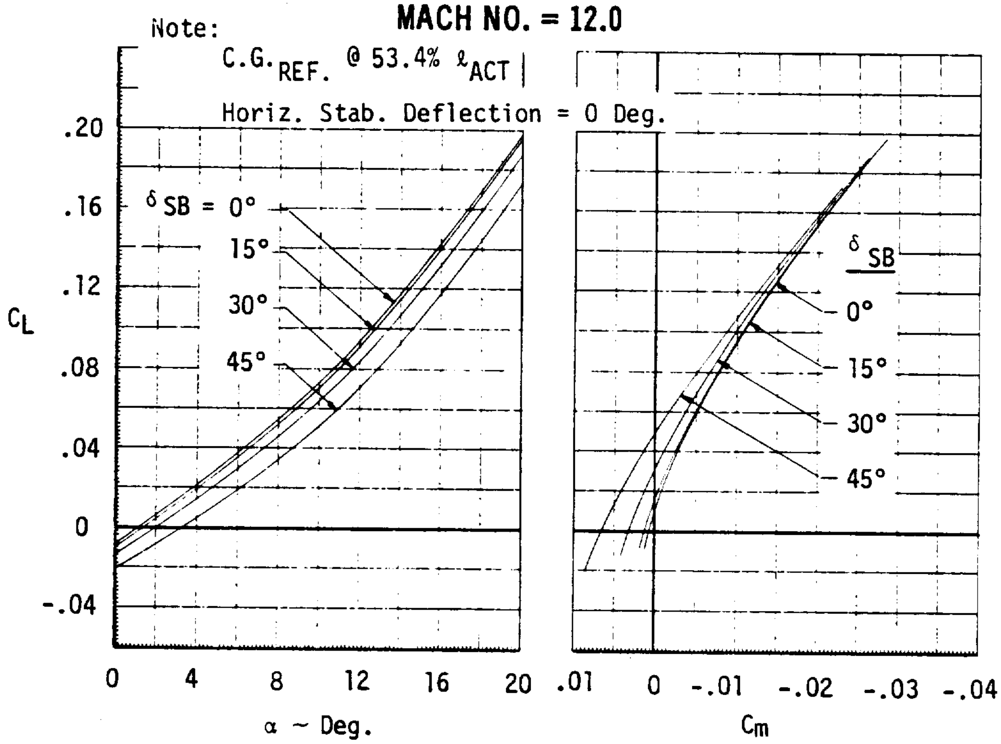
Glide



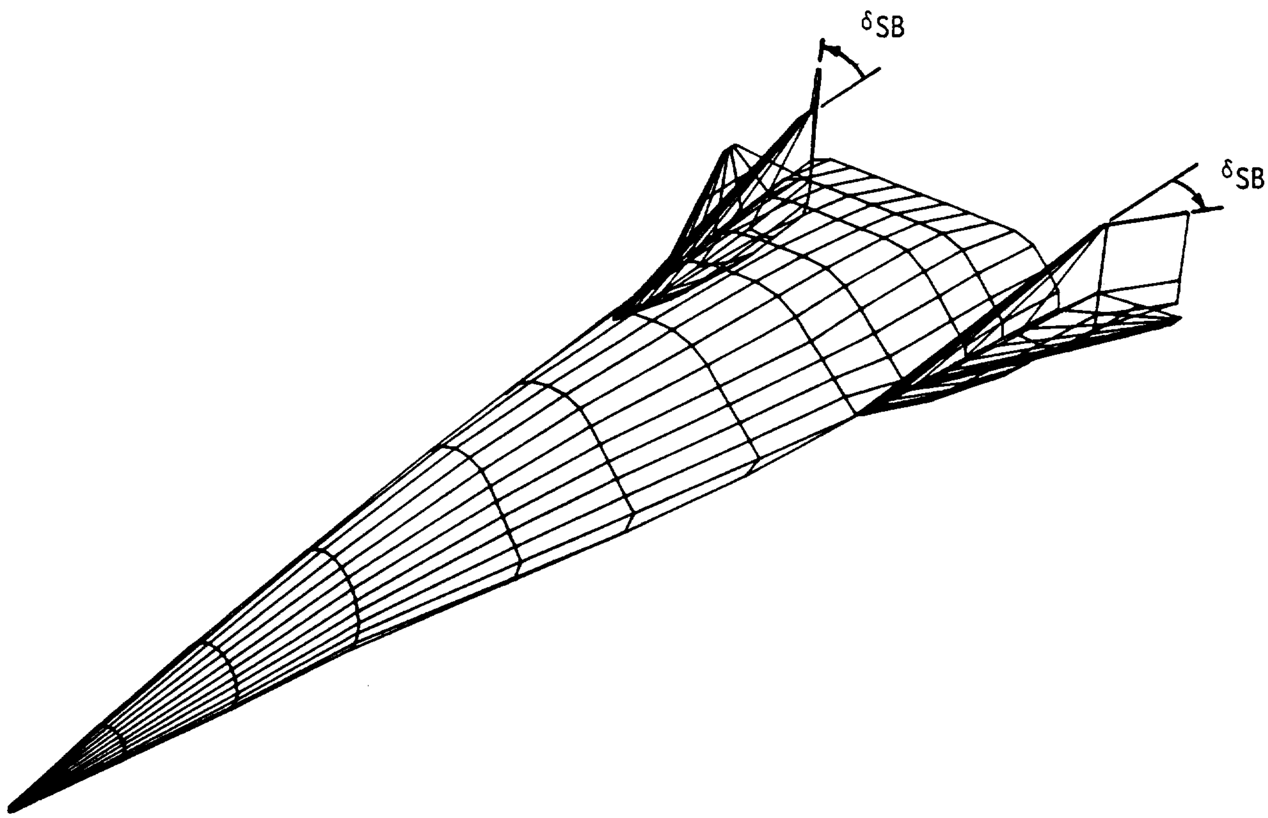
(U) FIGURE 4-38
MACH 12 ROCKET
EFFECT OF SPEED BRAKE DEFLECTION
MACH NO. = 2.0



(U) FIGURE 4-39
MACH 12 ROCKET
EFFECT OF SPEED BRAKE DEFLECTION
MACH NO. = 12.0



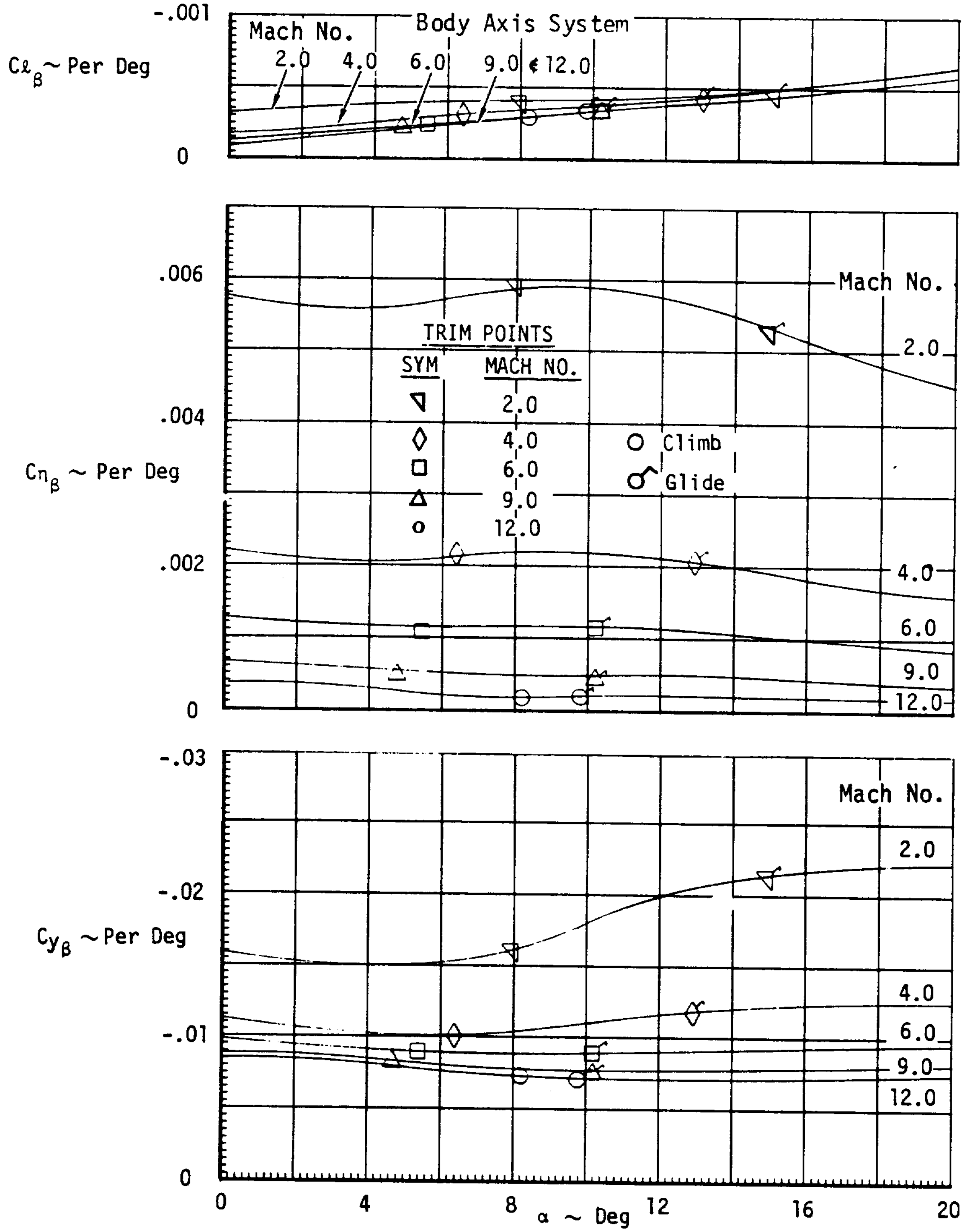
(U) FIGURE 4-40
MACH 12 ROCKET
Speed Brakes Deflected 30°



affected and the rudders are quite effective in reducing the vehicle L/D as was shown in Figure 4-29.

(U) Figure 4-41 shows the lateral-directional static stability characteristics of the basic vehicle. The symbols indicate the selected climb and glide trim points previously discussed. These data indicate that the vehicle is essentially neutrally stable directionally at Mach 12.0. The static directional stability can be increased by increasing the size of the vertical tails or by increasing the vertical fin toe-in angles. The design toe-in angle is 5 degrees. Figure 4-42 presents a tabulation of the pertinent lateral-directional stability derivatives for the selected trim points which are employed in the handling qualities analysis. These were obtained from the Gentry program.

(U) FIGURE 4-41
MACH 12 ROCKET
LATERAL-DIRECTIONAL STATIC STABILITY
HORIZONTAL STABILATOR DEFLECTION = 0°
C.G. at 64% ℓ_{ACT}



(U) FIGURE 4-42
MACH 12 ROCKET
LATERAL-DIRECTIONAL DERIVATIVES AT TRIM FLIGHT CONDITIONS

MACH NO.	2.0	4.0	6.0	9.0	12.0
-CLIMB-					
Cy_r/RAD.	.23432	.07336	.03320	.00987	-.00165
Cn_r/RAD.	-.98320	-.59696	-.51184	-.45680	-.40657
Cl_r/RAD.	.00018	.00478	.00119	.00040	.00500
Cy_{δ_r}/RAD.	.1033	.0696	.0634	.0615	.0573
Cn_{δ_r}/RAD.	-.1008	-.0686	-.0628	-.0614	-.0575
Cl_{δ_a}/RAD.	.0763	.0377	.0286	.0264	.0344
Cl_p/RAD.	-.0790	-.0407	-.0314	-.0287	-.0354
-GLIDE-					
Cy_r/RAD.	.21212	.07714	.04199	.01241	.00002
Cn_r/RAD.	-1.20724	-.67874	-.51374	-.43558	-.40218
Cl_r/RAD.	-.01268	.00262	.00505	.00695	.00694
Cy_{δ_r}/RAD.	.0924	.0624	.0585	.0561	.0559
Cn_{δ_r}/RAD.	-.0931	-.0629	-.0590	-.0566	-.0563
Cl_{δ_a}/RAD.	.0765	.0379	.0306	.0349	.0370
Cl_p/RAD.	-.0838	-.0454	-.0351	-.0374	-.0383

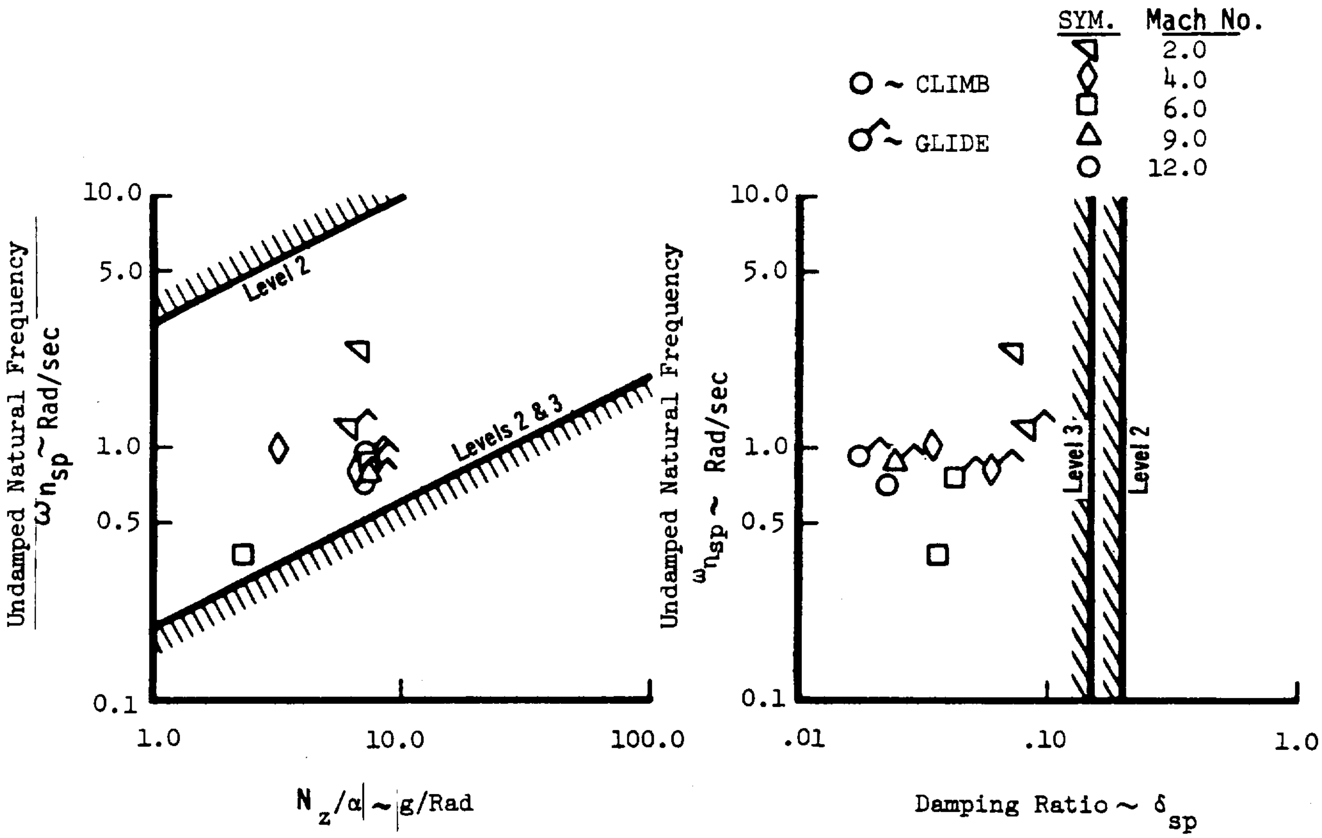
4.3.3 (U) HANDLING QUALITIES - The bare airframe handling qualities of the basic vehicle were examined briefly to provide some insight as to the behavior of the unaugmented short period characteristics. These conditions are for the bare airframe and are therefore representative of a totally inoperative augmentation system. The trim flight conditions indicated in the preceding section were utilized in the analysis. The results are compared to the requirements of the current military flying qualities specification, Reference (7). Although this specification is not directly applicable to a research aircraft, the Level 2 and Level 3 requirements of Categories B and C for a Class IV aircraft, as stated therein, are employed to provide a frame of reference. Briefly, the class, category, and levels referred to are defined as follows:

Class IV	High maneuverability airplanes such as: Fighter/Interceptor Attack Tactical Reconnaissance Observation
Category B	Nonterminal Flight Phases that are normally accomplished using gradual maneuvers and without precision tracking, although accurate flight-path control may be required. Included in this category are: Climb Cruise Descent Emergency Descent
Category C	Terminal Flight Phases (takeoff and landing)
Level 2	Flying qualities adequate to accomplish mission flight phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists.
Level 3	Flying qualities such that the airplane can be controlled safely but pilot workload is excessive, or mission effectiveness is inadequate, or both. Category B and C Flight Phases can be completed.

(U) Figure 4-43 shows the longitudinal short period dynamic stability characteristics for the unaugmented vehicle. The undamped short period natural frequency, $\omega_{n_{sp}}$, is plotted versus the normal load factor per unit angle of attack, N_Z/α, and the short period damping ratio, ζ_{sp}. While the frequencies are within acceptable limits throughout the mission profile it will be noted that the damping ratio is less than the Reference (7) standards at all flight conditions. Artificial damping will of course be required. This is typical and expected for high speed flight in the upper regions of the atmosphere.

(U) Figures 4-44 and 4-45 show the unaugmented lateral-directional characteristics of the basic vehicle. The Dutch Roll characteristics are presented in Figure 4-44 where the undamped natural frequency, ω_{nd}, is plotted versus the damping ratio, ζd. The results indicate that the dutch roll damping for all but the Mach 2 conditions is less than that required for Level 3 and will of course require augmentation. Since the roll-sideslip coupling (ϕ/β) is small, the product of frequency and damping ratio, $\zeta_d\omega_{nd}$, need only be greater than zero to meet the Level 3 requirement. This is accomplished throughout the Mach range with the Mach 2 climb and glide points meeting Level 2 requirements.

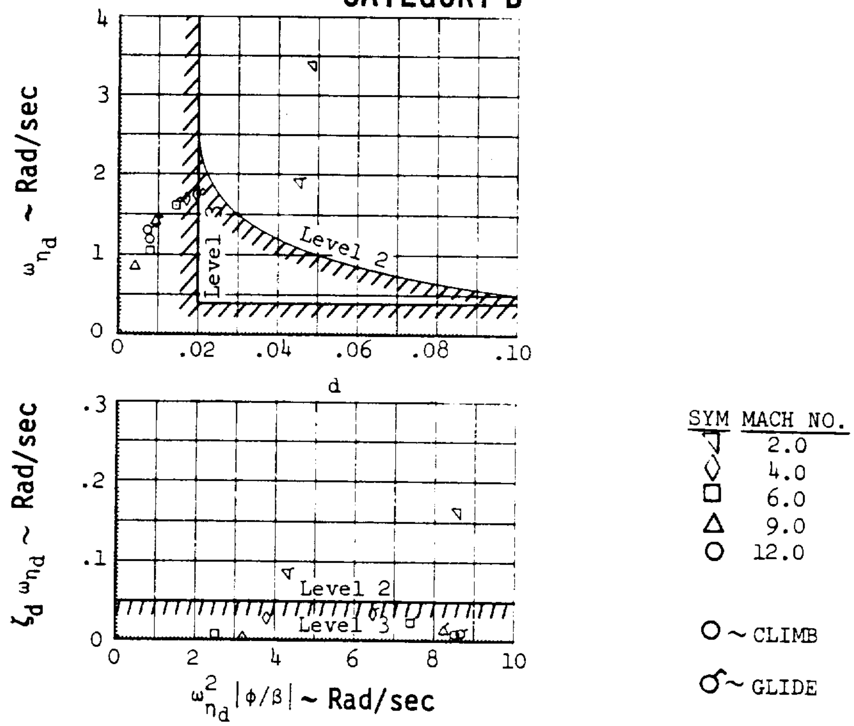
(U) FIGURE 4-43
MACH 12 ROCKET
LONGITUDINAL SHORT PERIOD DYNAMIC STABILITY
UNAUGMENTED
CATEGORY B



(U) The roll-sideslip coupling characteristics produced in rolling maneuvers are examined in Figure 4-45. These data indicate that the sideslip excursions, $\Delta\beta_{max}$, are within the Level 2 requirements and that the roll-rate oscillation, P_{osc}/P_{AV}, exceeds the Level 2 limit only for the initiation of glide at Mach 12.

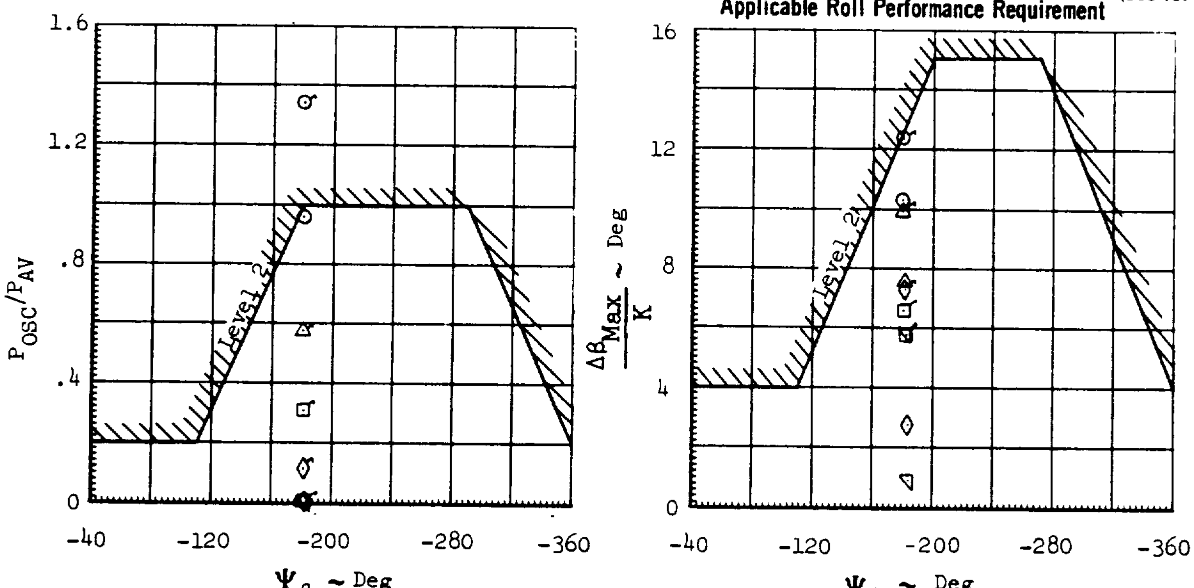
4.3.4 (U) TAKEOFF AND LANDING CHARACTERISTICS - The longitudinal static stability and control characteristics at takeoff and landing speeds are shown in Figure 4-46. The longitudinal short period dynamic stability characteristics are shown in Figure 4-47. Although the damping ratio is less than Reference (7) standards at takeoff, it is quite acceptable for landing.

(U) FIGURE 4-44
MACH 12 ROCKET
DUTCH ROLL CHARACTERISTICS
UNAUGMENTED
CATEGORY B

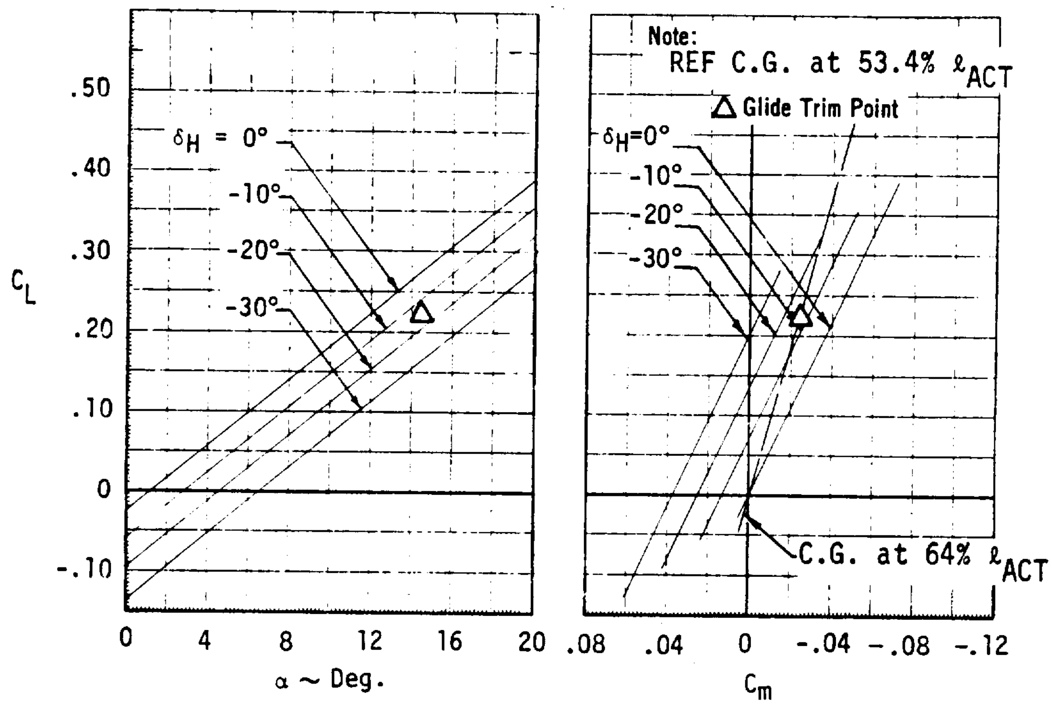


(U) FIGURE 4-45 MACH 12 ROCKET
ROLL-SIDESLIP COUPLING UNAUGMENTED
CATEGORY B

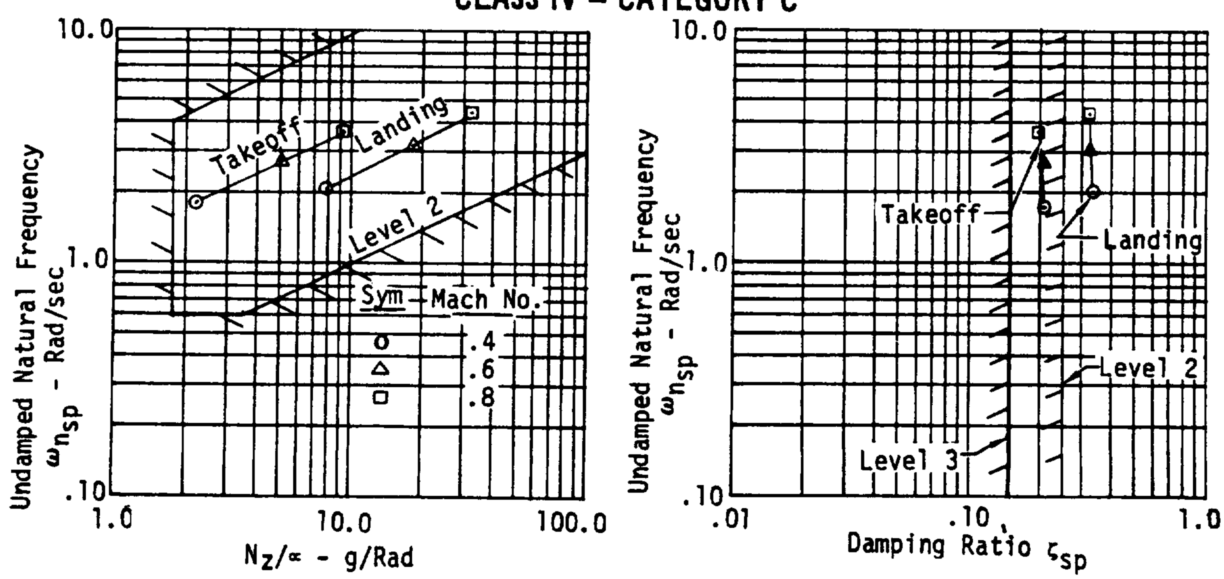
$$K = \frac{\text{Commanded Roll Performance}}{\text{Applicable Roll Performance Requirement}}$$ (see ref 7)



(U) FIGURE 4–46
MACH 12 ROCKET
LONGITUDINAL STATIC STABILITY AND CONTROL CHARACTERISTICS
MACH NO. ≤ 0.80

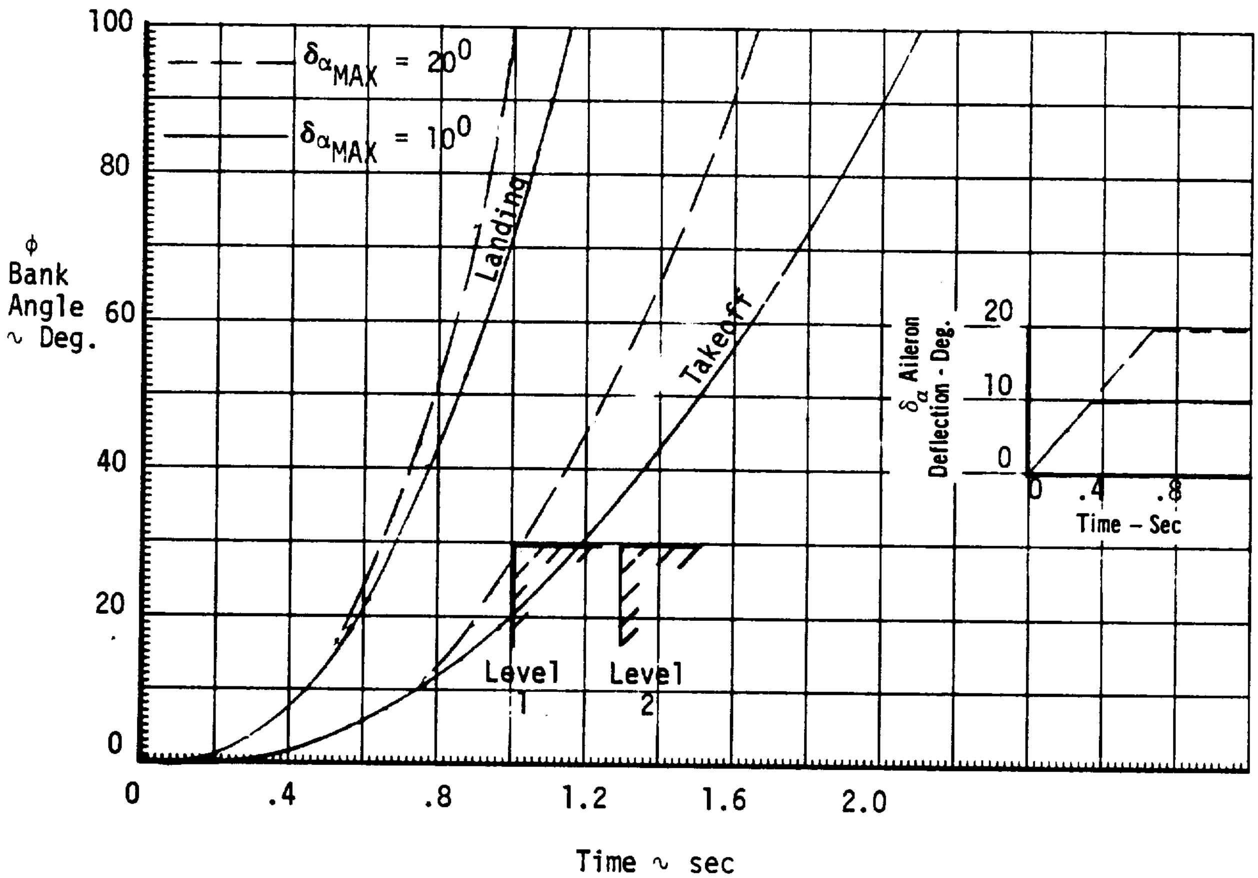


(U) FIGURE 4–47
MACH 12 ROCKET
LONGITUDINAL SHORT PERIOD DYNAMIC STABILITY
UNAUGMENTED AIRFRAME
CLASS IV – CATEGORY C



(U) The lateral control characteristics at takeoff and landing speeds are shown in Figure 4-48. The delta tip controls are shown to be quite effective when deflected differentially to provide roll control. Since the tip controls are also used for pitch control, when they are operated collectively, the amount of differential deflection available for lateral control is a function of the pitch control requirements. This sharing is most critical during takeoff and landing when large control deflections may be required in both control modes. Two differential deflections, 10° and 20°, are shown in Figure 4-48. These will provide ample lateral control power and at the same time reserve sufficient pitch control deflections, -20° and -10° respectively, to trim out the lift required for takeoff and landing ($\alpha \approx 15°$). See Figure 4-46.

(U) FIGURE 4-48
MACH 12 ROCKET
BANK ANGLE vs TIME
TAKEOFF AND LANDING

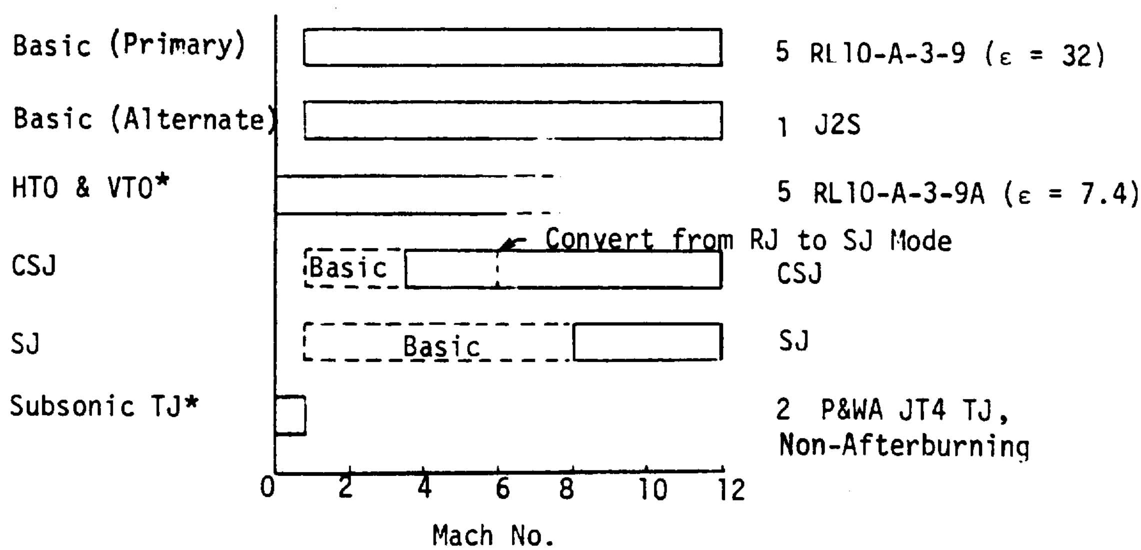


4.4 PROPULSION

(U) In the course of the study it was determined that it was feasible to use near-term engines developed from existing hardware for the Mach 12 vehicle. Use of such engines to achieve the high performance desired, results in costs considerably less than those required for new engines. Thus the vehicle performance objectives were satisfied at high confidence and low cost with readily available engines. Two valid propulsion systems were identified. One system would use five advanced P&WA RL10-A-3-9 rocket engines, the other would use a single advanced Rocketdyne J2S rocket engine. For the basic vehicle the system with five RL10-A-3-9 engines was selected, in order to acquire the multiple-engine reliability, to facilitate throttling to low thrust levels for cruise, and to permit roll control without adding a separate reaction system.

(U) Five propulsion system options were investigated to extend the research capability of the basic vehicle. For all of these options, the basic rocket propulsion system is retained. Modifications to the basic vehicle permit incorporating the options and accomplishing further significant research for moderate additional costs. In the following paragraphs the propulsion system of the basic vehicle is described, followed by description of the propulsion system research options added to the basic vehicle. Figure 4-49 summarizes the operating regimes of the various propulsion systems.

(U) FIGURE 4-49
PROPULSION SYSTEMS AND OPERATING REGIMES
Mach 12 Vehicle



*Ground take-off; all others air launched.

4.4.1 (U) BASIC VEHICLE - The basic Mach 12 vehicle uses five LO_2/LH_2 P&WA RL10-A-3-9 rocket engines, an advanced version of the RL10-A-3-3 used on the Centaur vehicle, with chamber pressure of 450 psi and mixture ratio of 6.0. The engines are installed spanwise at the trailing edge of the vehicle in a horizontal arrangement. The basic propulsion system is compatible with each of four vehicle options; namely the Thermal Protection System, Armament, Staging and Ramjet options.

(U) Two candidate off-the-shelf rocket engines were available for this vehicle: the advanced P&WA RL10-A-3-9 and the Rocketdyne J2S. Both engines were investigated in Phase II and are described in Volume III. Either five RL10-A-3-9 engines or one J2S engine could be used to perform the nominal flight profile. However, factors other than performance were considered, as summarized in Figure 4-50. Operational aspects such as throttling, vehicle stability/control, reliability, and weight advantages point to the five engine, RL10-A-3-9 arrangement which was chosen for the basic vehicle. The engine is identical in all physical aspects to the engine described in Volume III: length, diameter, weight, chamber pressure. The Phase III engine, however, operates at O/F = 6.0 to improve vehicle performance as discussed in Section 2.4. The O/F ratio is held constant with varying throttle settings.

(U) Thrust and fuel consumption characteristics of the RL10-A-3-9 rocket engine are presented in Figure 4-51. The minimum operating altitude increases as the engine is throttled, because throttling is accompanied by reduced chamber pressure and thus reduced nozzle exit pressure.

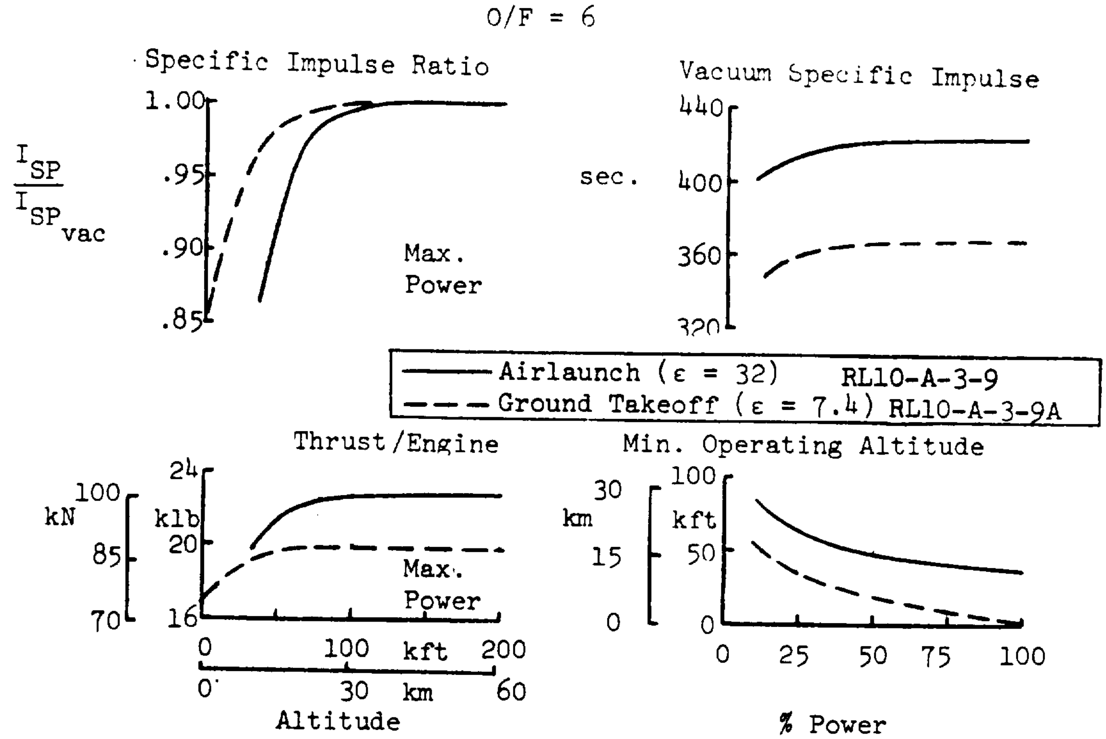
(U) FIGURE 4-50
COMPARISON OF ENGINE ALTERNATES

Candidate Propulsion System	Cost	Technology Status	Interface	Total Wt. lb (kg)	I_{sp} sec	Reliability	
		Throttling to match thrust to drag at cruise	(1) Roll control (2) HTO/VTO option			Expected number of flights between *failure	Expected no. of flights between secondary mission **
P&WA RL10-A-3-9 (5 Engines)	\$24M RDT&E \$ 2M Investment	Shutdown four engines, throttle one to 31%	(1) Differential gimballing of outboard engines provides roll control (2) Short-nozzle (ε=7.4) version of engine needed	1455 (660)	424 (O/F=6.0)	20,000 At least 3 engines lightoff	3333 At least 4 engines lightoff; 40 All 5 engines lightoff
Rocketdyne J2S (1 Engine)	\$10M RDT&E \$ 3M Investment	Throttle to 3%; this technology not demonstrated	(1) Requires addition of a reaction control system (2) Basic engine okay	4050 (1840)	431 (O/F=5.9)	220	

* Failure means possible inability to reach a landing site due to lack of engine ignition or lack of full thrust.

** Secondary mission means enforced choice of alternate flight profile due to lack of engine ignition or lack of full thrust.

(U) FIGURE 4-51
ROCKET ENGINE PERFORMANCE



4.4.2 (U) PROPULSION OPTIONS - Five research options to the basic vehicle propulsion system were investigated, to extend the research capability of the Mach 12 vehicle. All of these modifications retain the basic rocket propulsion system. One of the options involves a change in the nozzle expansion ratio of the engines, to permit sea level operation. The other modifications do not change the engines, but add various airbreathing engines as additions to the basic vehicle. Figure 4-52 summarizes the characteristics of the propulsion systems for these vehicle options, and Figures 4-53 through 4-56 show the propulsion system configuration details and the installed engine performances. Figures 4-9 through 4-15 and Figure 4-21 depict the overall vehicle arrangements of these configuration options.

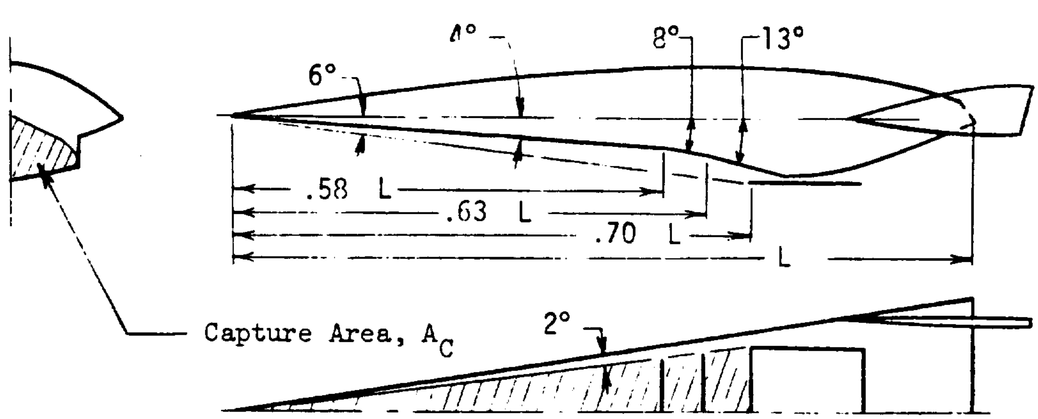
4.4.3 (U) RESEARCH VERSATILITY - The Mach 12 vehicle as evolved has extensive potential for accomplishing useful research associated with hypersonic airbreathing engines. This potential is beyond that associated with the specific modifications just discussed. The excess thrust available from the basic RL10-A-3-9 rocket engines permits research testing of a wide variety of advanced airbreathers. Potential research falls into two categories:

- o Component research and development (inlets, burners, nozzles)
- o System test and evaluation Mach 12 flying test bed

(U) FIGURE 4-52
PROPULSION MODIFICATIONS TO MACH 12 VEHICLE

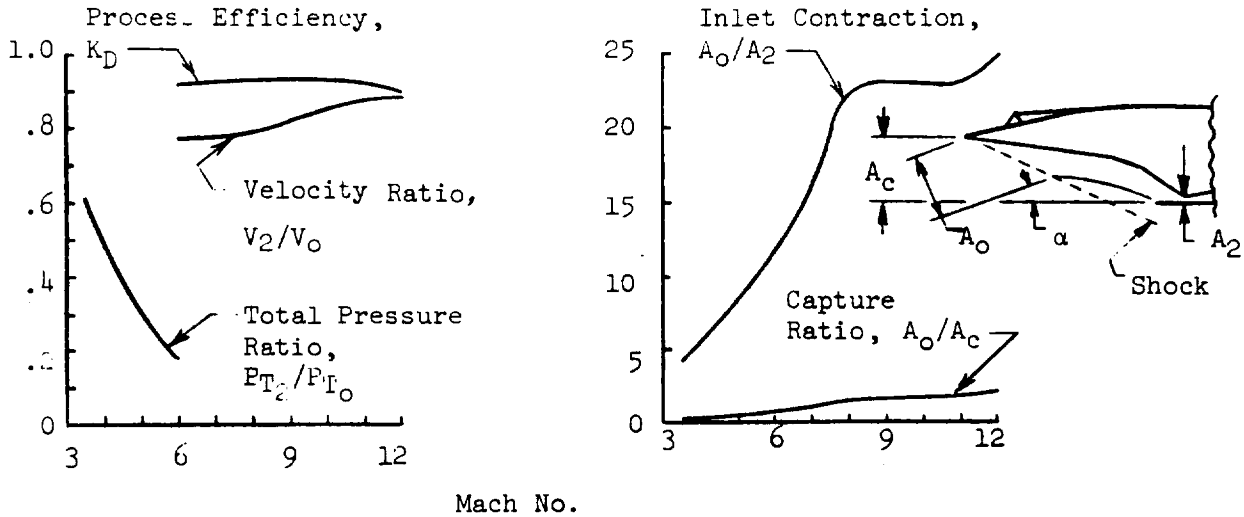
Type of Modification	Status of Basic RL10-A-3-9 Rocket System	Characteristics of Added Propulsion System: Operating Speed Range	Thrust Desired	Configuration Source	Installed Performance Presented	Remarks
HTO/VTO	Replaced with identical engines except ε = 7.4 (RL10-A-3-9A)	M=0 to fuel depletion	VTO: T/W>1	N/A	Fig. 4-51	
CSJ	Maintained, used to accel. to Mach 3.5	M=3.5-12	At M=3.5 a/g >.5	Inlet, nozzle per Ref. 9; combustor per Ref. 14 (45", 114 cm)	Inlet, Fig. 4-54 Instl. engine performance, Fig. 4-55	Inlet config. in Fig. 4-53 bleed & bypass drag are zero
SJ	Maintained, used to accel. to Mach 8	M=8-12	At M=8 a/g >.5	Inlet, nozzle same as CSJ; combustor per Phase II (30",76cm)	Same as CSJ	Same as CSJ
TJ	Maintained, not operated	M=0-.8	Approx. 50 klb (220 kn)	Inlet typical of those used with engine in current appl.; engine and nozzle per engine spec.	Figure 4-56	Bleed & bypass drag are zero
RJ	Maintained, used as primary propulsion	M=2-6	Test Bed	Tactical missle RJ typical of studies, Ref. 15; subscale manned-vehicle RJ per Ref. 16	Not conducted	

(U) FIGURE 4-53
INLET CONFIGURATION FOR THE CSJ/SJ OPTIONS



(U) FIGURE 4-54
CSJ/SJ INLET EFFICIENCY AND CAPTURE CHARACTERISTICS

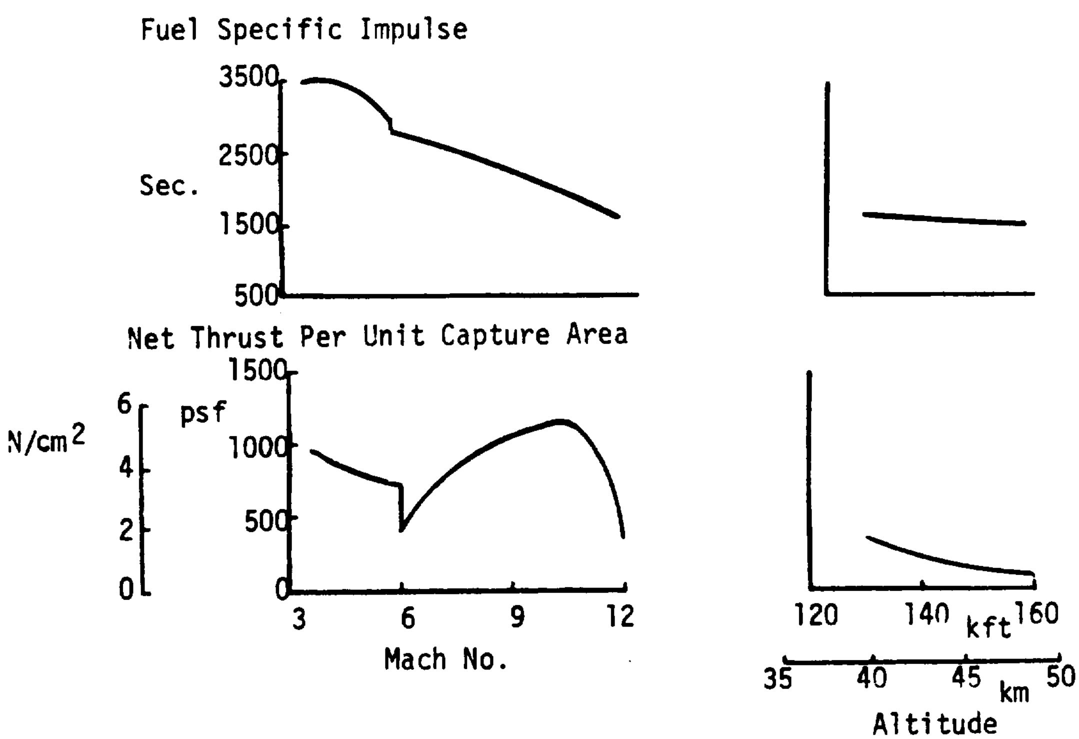
Based On Inviscid, Perfect Gas Analysis



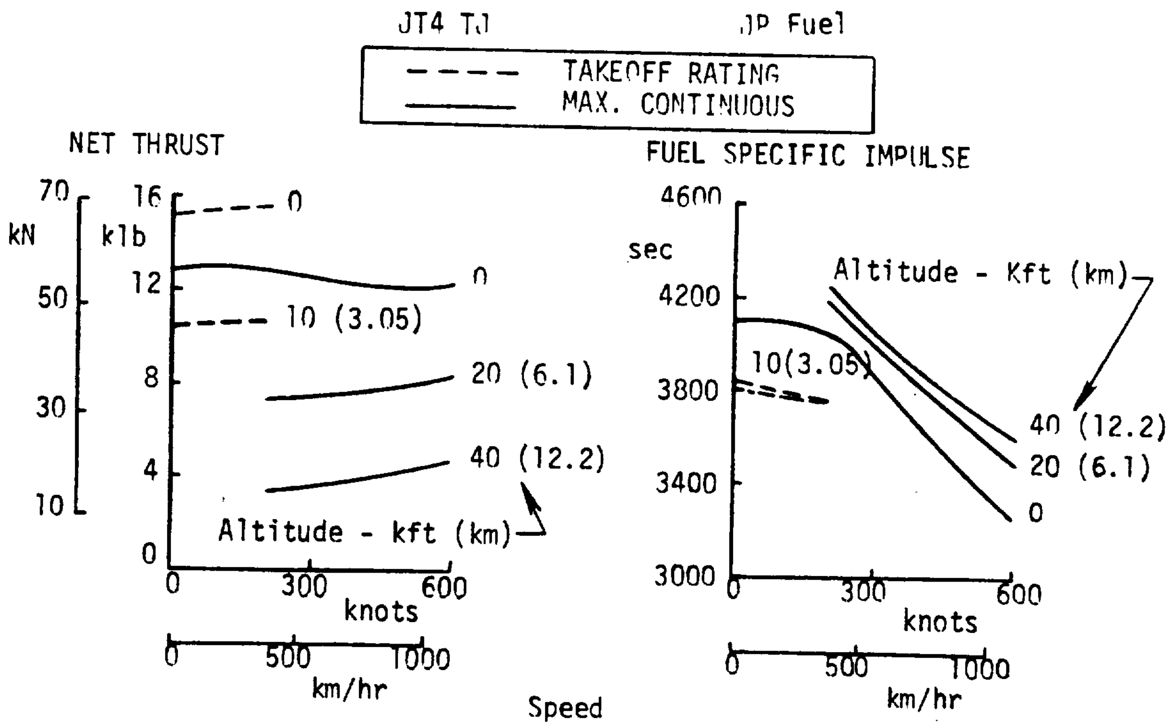
(C) FIGURE 4-55
INSTALLED PERFORMANCE OF THE CSJ/SJ OPTIONS

Stoichiometric (ϕ = 1)

Acceleration (Along Flight Path of Figure 4-64) | Cruise (Mach 12)



(U) FIGURE 4-56
INSTALLED PERFORMANCE OF THE SUBSONIC TJ OPTION

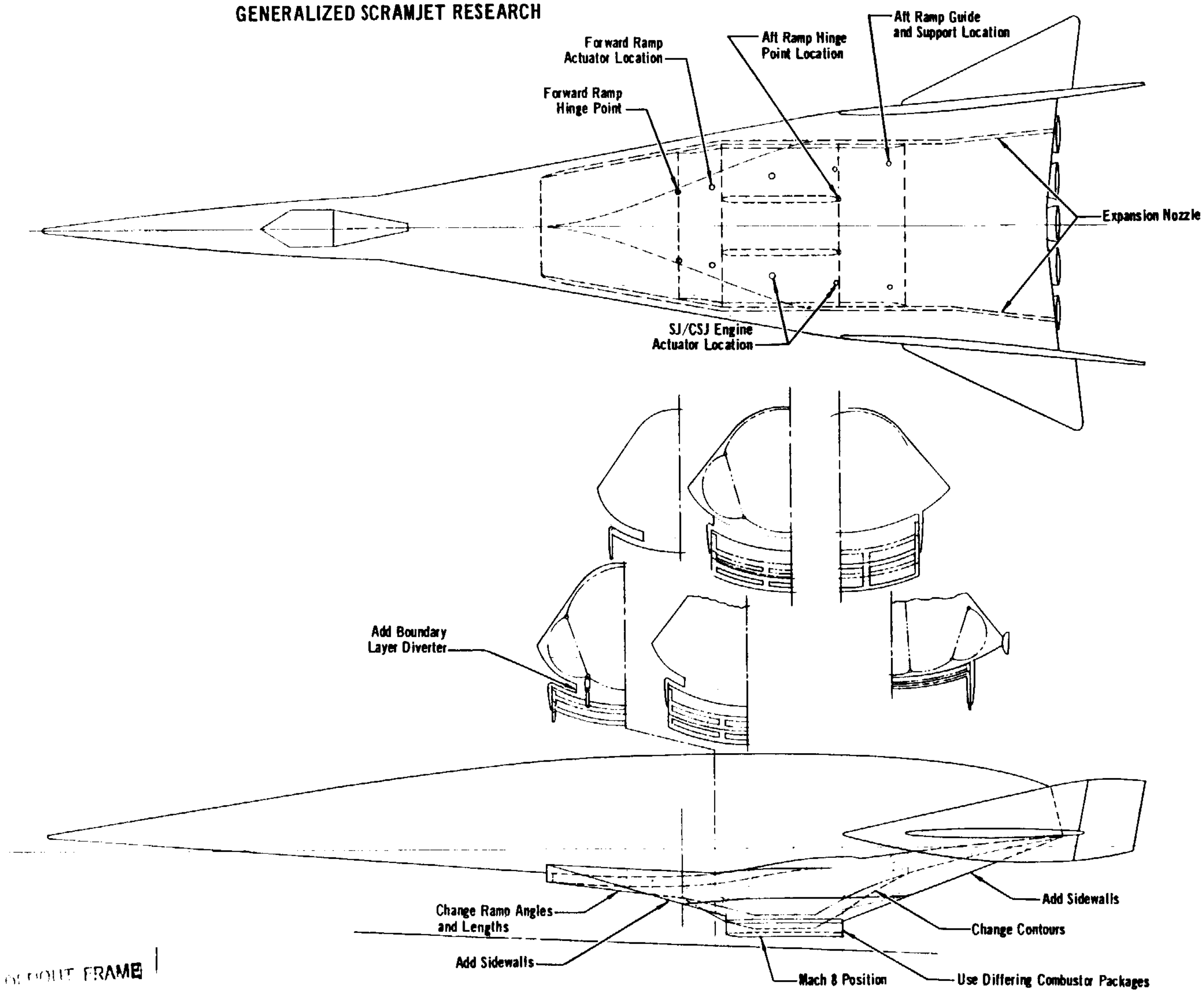


The component testing would follow exploratory development on the ground. A sequential program would be involved: inlet first, then use the inlet in burner development, finally use the inlet and burner in nozzle development. With successful flight verification of the components, the complete propulsion system would be assembled and proven in flight test. The capability of this vehicle to permit testing in a high dynamic pressure environment with very large air flow rates is unique. It is not available in any existing or currently-planned facility.

(U) For most of the specific engine sizes currently envisioned for potential operational systems, the vehicle could be used to test components and systems as large as approximately 50% of full scale. This size would be of significant benefit in providing high confidence for development of advanced propulsion system elements. For certain engines the vehicle could test full-scale systems, such as the tactical missile engines illustrated in Figure 4-22.

(U) Regarding the SJ and CSJ engines, considerable in-flight development is anticipated due to the lack of ground facilities. In this endeavor all of the defining geometry could be open to change, as shown in Figure 4-57. Inlet ramp angles and ramp lengths could be varied to effect performance trade-offs, as could nozzle contours. The benefit of sidewalls on the inlet and nozzle, and of boundary layer diverters, could be assessed. Likewise different combustor internal geometries could be investigated. The result of this parametric effort would be the integrated scramjet engine best suited to the research vehicle mission plus a significant data base for operational systems.

(U) FIGURE 4-57
GENERALIZED SCRAMJET RESEARCH



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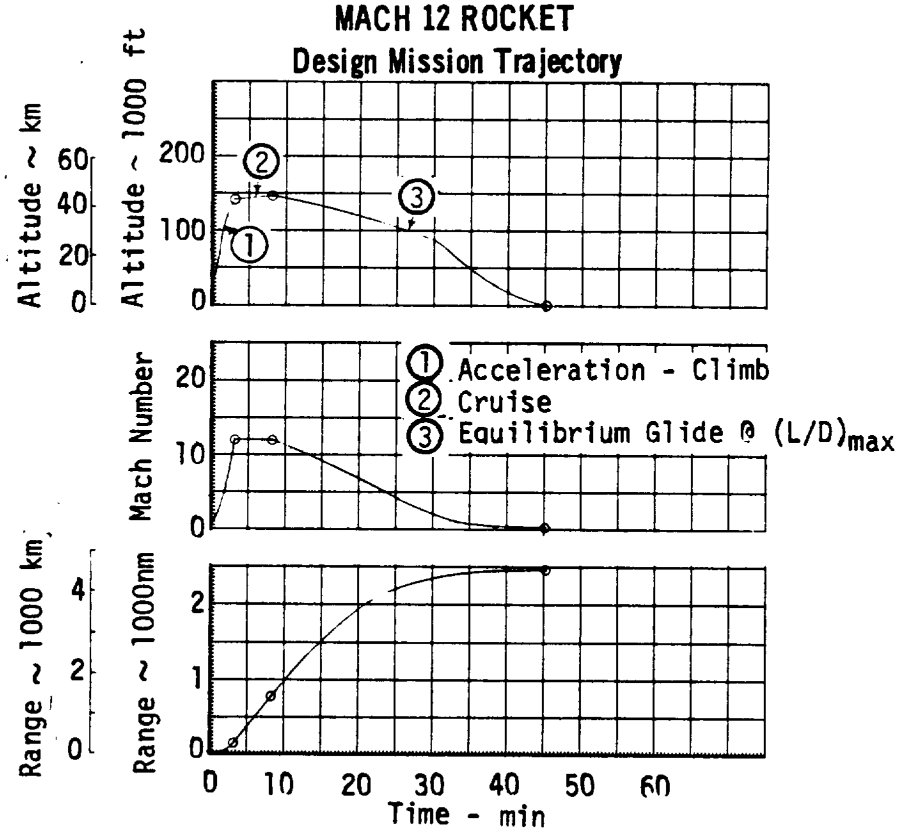
4.5 PERFORMANCE AND TRAJECTORIES

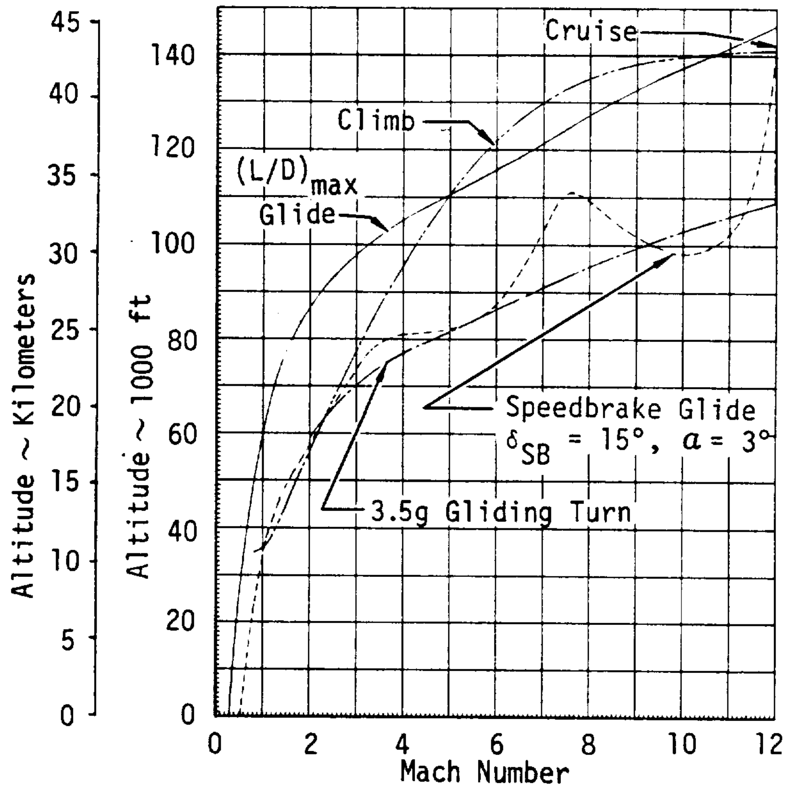
(U) The performance and trajectories for the basic vehicle and several configuration options are presented in the following sections.

4.5.1 (U) BASIC VEHICLE - The design mission trajectory for the basic vehicle, as defined in Section 2.3, is shown in Figures 4-58 and 4-59. The climb trajectory is optimized for minimum fuel usage. The end-of-boost constraints were to achieve Mach 12 at the equilibrium glide altitude corresponding to $(L/D)_{max}$ and a glide path angle of 0 degrees. Three modes of unpowered glide following the five-minute cruise are illustrated in Figure 4-59. Maximum range is provided by gliding at $(L/D)_{max}$. Minimum range will be obtained by deflecting the rudder speedbrakes ($\delta_{SB} \approx 15°$) and reducing the angle-of-attack to 3°. Lower angles-of-attack will result in a trajectory that exceeds the design dynamic pressure limitation of 2,000 lb/ft^2 (95,800 N/M^2). The pilot can modulate the use of the speedbrakes to provide a variation in glide range as required for energy management purposes. The third mode of descent shown is a gliding turn performed at a normal load factor of 3.5 g following the five-minute cruise. A descent to an altitude of 125,000 ft (38 km) is required to provide sufficient dynamic pressure to achieve 3.5 g.

(U)FIGURE 4-58
MACH 12 ROCKET
Design Mission Trajectory



(U) FIGURE 4-59
MACH 12 ROCKET
DESIGN MISSION PROFILE

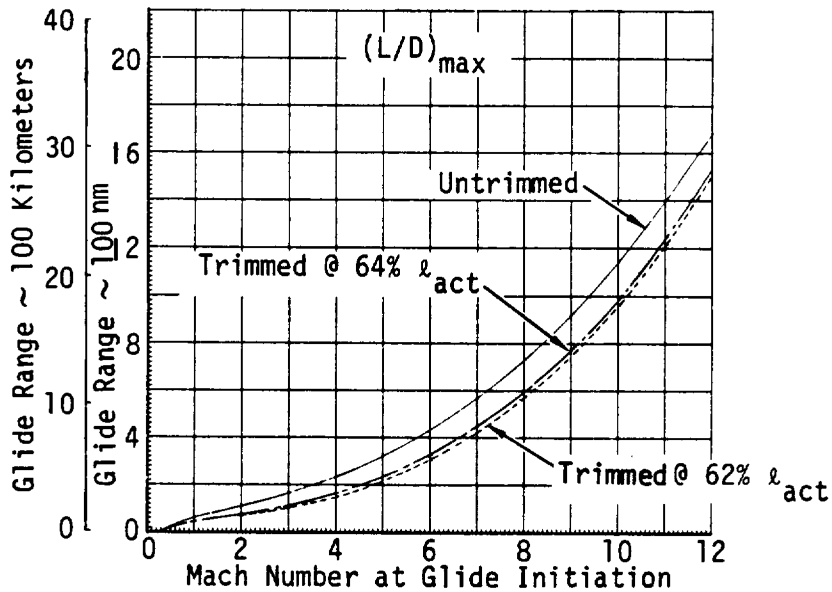


(U) The effect of trim stabilator deflections on glide range are shown in Figure 4-60 as a function of Mach number. The trimmed $(L/D)_{max}$ range capabilities, based on Gentry data and assumed center of gravity locations of 64% and 62% of body length, are compared with the glide range achieved for the untrimmed $(L/D)_{max}$ values determined by the basic Phase II methods. The more stable forward center of gravity location requires more stabilator deflection for trim and therefore results in a lower value of $(L/D)_{max}$ and less range. The small difference between the trimmed and untrimmed range values indicates that trim drag losses are small for this configuration.

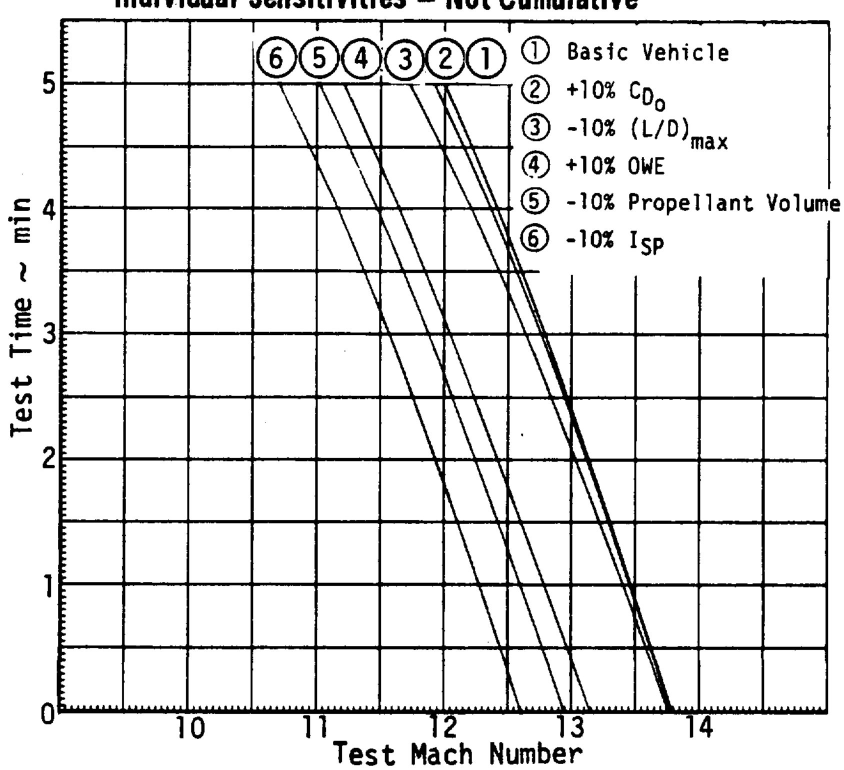
(U) The basic vehicle performance sensitivities to changes in vehicle physical characteristics and flight environment were evaluated and are shown in Figures 4-61 and 4-62. OWE, propellant weight, rocket engine specific impulse, $(L/D)_{max}$ and C_{D_o} are varied 10% in a direction that would reduce performance. The results are shown in Figure 4-61 in terms of the test time and test Mach number achievable. The sensitivity of each parameter on performance can be determined by comparing to the basic vehicle performance also shown in Figure 4-61.

(U) The effect of varying atmospheric conditions from the 1962 U.S. Standard Atmosphere was also investigated. A Mil Std. 210 Tropical atmosphere was used in the analysis. The changes in drag and pressure losses were found to be negligible and the effects are insignificant relative to the basic vehicle performance. If the vehicle were not air launched, these losses would be somewhat greater.

(U) FIGURE 4-60
MACH 12 ROCKET
GLIDE RANGE CAPABILITY

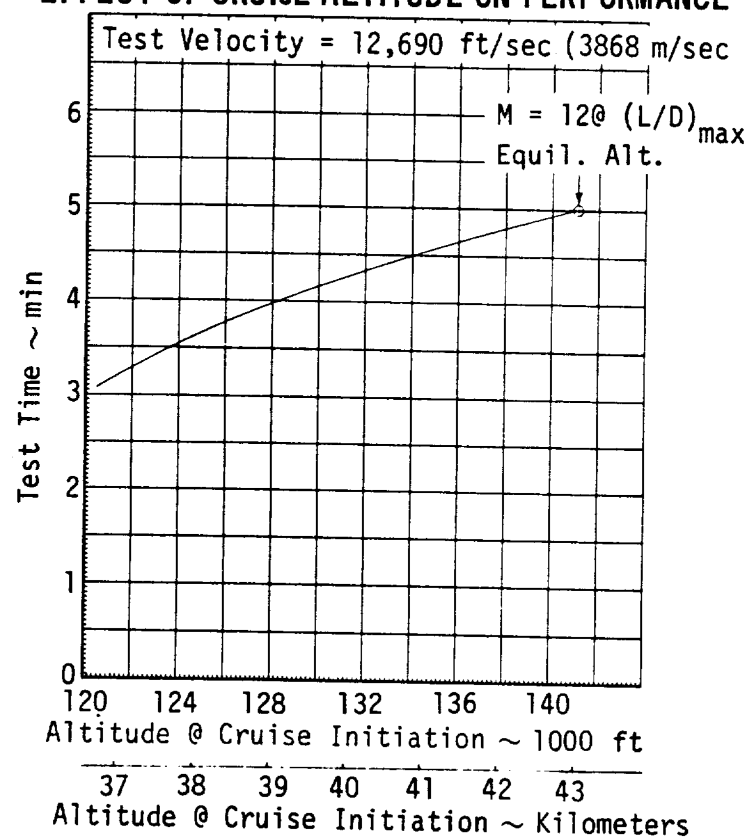


(U) FIGURE 4-61
MACH 12 ROCKET PERFORMANCE SENSITIVITIES
Individual Sensitivities – Not Cumulative



(U) The effect of varying cruise altitude is presented in Figure 4-62. In this analysis, test altitudes lower than that required for equilibrium Mach 12 cruise at $(L/D)_{max}$ were selected. The cruise velocity is assumed constant and equal to the velocity attained by the basic configurations upon attaining cruise altitude. The required boost trajectories were optimized for minimum fuel usage in each case. Figure 4-62 indicates the test time available when the cruise altitude is lowered. The loss in performance indicated is the result of higher drag losses associated with operation in the more dense lower atmosphere.

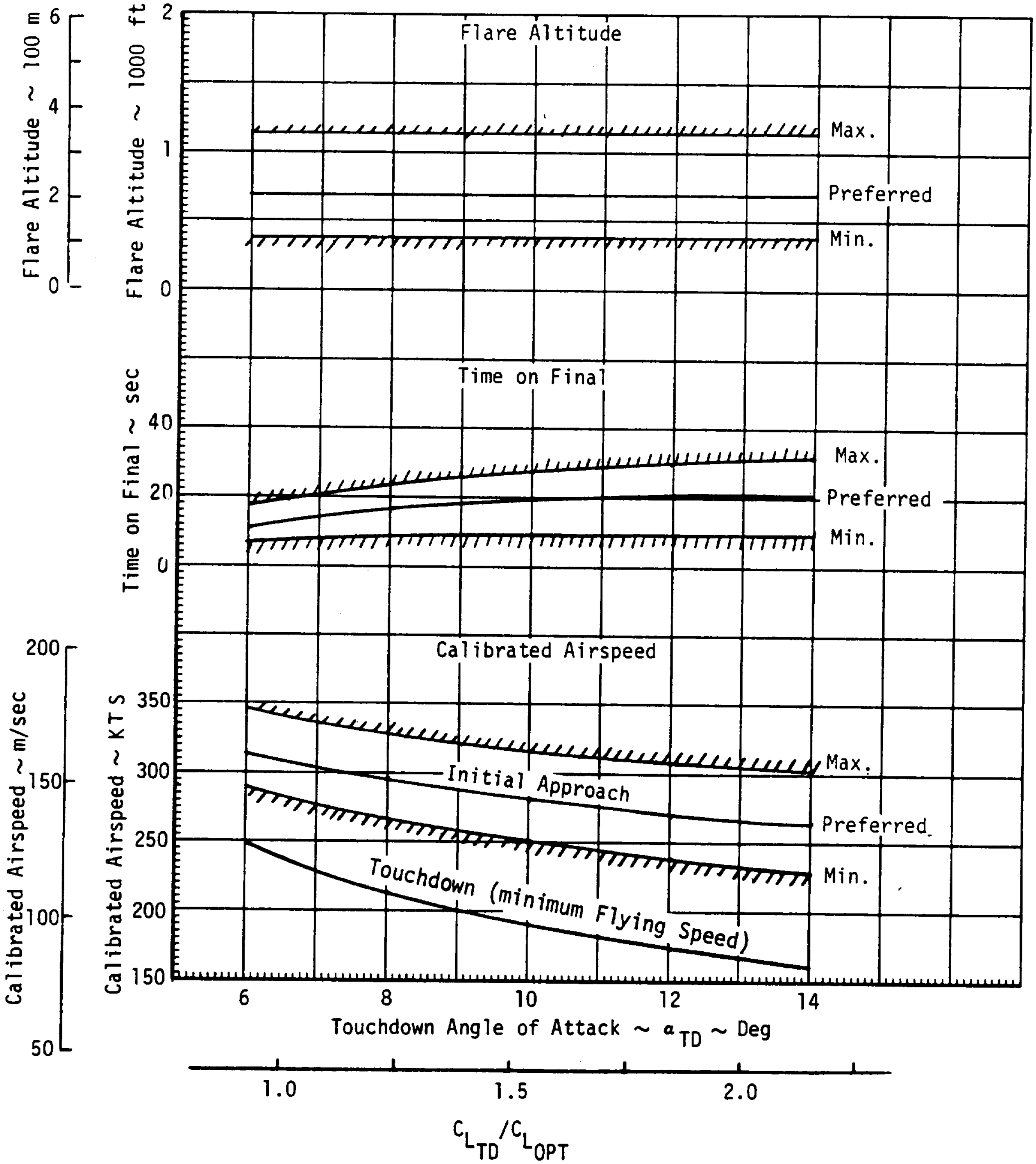
(U) FIGURE 4-62
MACH 12 ROCKET
EFFECT OF CRUISE ALTITUDE ON PERFORMANCE



(U) Typical power-off approach and landing characteristics for the basic Mach 12 vehicle are shown in Figure 4-63 in terms of flare altitude, time-on-final, pre-flare approach speed, and touchdown velocity. The basic Mach 12 vehicle can be operated within the boundaries shown which denote the envelope of satisfactory pilot ratings (Cooper rating of 3.5) obtained in a MCAIR landing simulation study (Reference (17)) of power-off approach techniques. The preferred conditions indicated correspond to the flight path for which the best pilot ratings were obtained.

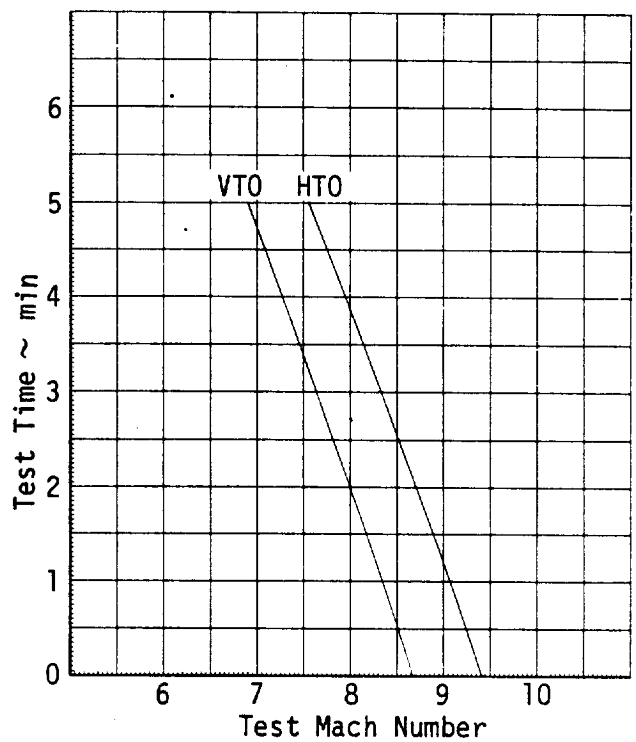
4.5.2 (U) CONFIGURATION OPTIONS - The performance of the basic vehicle with installed configuration options to provide for the testing of various research concepts is discussed in this section.

(U) FIGURE 4-63
MACH 12 ROCKET
APPROACH AND LANDING CHARACTERISTICS
POWER OFF



4.5.2.1 (U) Horizontal/Vertical Takeoff (HTO/VTO) Options - The performance capability of the basic configuration when it is modified for ground launch in either the horizontal or vertical takeoff mode is presented in Figure 4-64. Because of the large back pressure losses associated with ground launched vehicles, the exhaust nozzle expansion ratio employed for the 5 rocket engines is reduced from 32 to 7.4. Engine I_{sp} and vacuum thrust are therefore adjusted in the performance calculations to reflect these engine modifications. The performance results indicate that for five minutes of test time, test Mach numbers of 7.6 and 6.9 can be achieved with the basic vehicle when modified to operate in the horizontal and vertical takeoff modes, respectively. In the vertical takeoff mode, vehicle control is provided by engine gimballing. The HTO option has an additional use in subsonic research described in Section 4.5.2.6.

(U) FIGURE 4-64
MACH 12 ROCKET HTO/VTO OPTIONS PERFORMANCE CAPABILITY



4.5.2.2 (U) Scramjet/Convertible Scramjet (SJ/CSJ) Options - The performance capabilities of the basic vehicle when modified to incorporate a scramjet or convertible scramjet configuration were investigated employing the three flight paths and four modes of engine utilization shown in Figure 4-65. In all cases the flight is initiated from an air launch at .8M/35,000 ft (10,700 m) and the test time corresponds to rocket-off cruise on the airbreather engine (SJ/CSJ) at $(L/D)_{max}$ equilibrium altitude. The modes of operation and climb profiles employed are as follows and correspond to the alphabetical identification shown in Figures 4-65 and 4-66.

(A) Rocket boost along the rocket flight path (see Figure 2-3) to test Mach number and $(L/D)_{max}$ equilibrium altitude.

(B) Rocket boost along the rocket flight path to an altitude of 100,000 ft (30,500 m), constant altitude rocket acceleration to Mach 8; then either

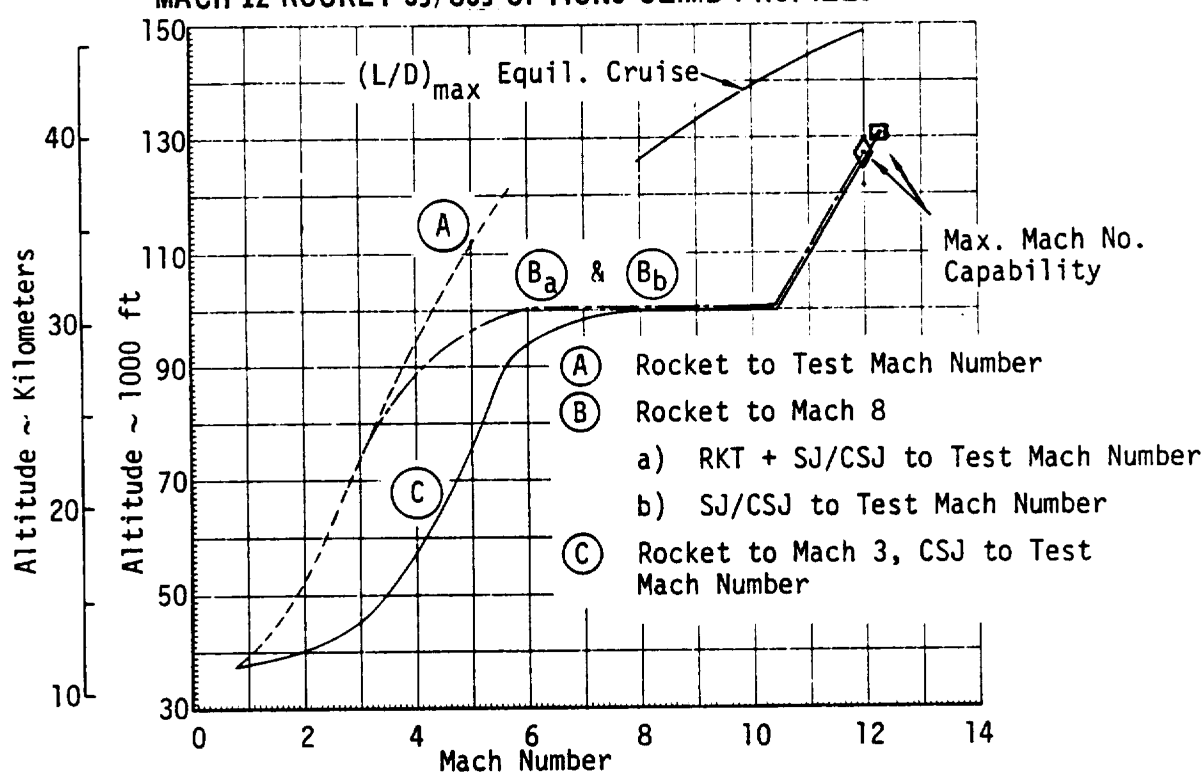
(a) rocket plus SJ/CSJ boost along the airbreather flight path (Ref. Volume III, Figure 4-30) to test Mach number followed by climb at test Mach number to $(L/D)_{max}$ equilibrium altitude.

or (b) SJ/CSJ boost (rocket-off) along the airbreather flight path to test Mach number followed by climb at test Mach number to $(L/D)_{max}$ equilibrium altitude.

(C) Rocket boost to airbreather flight path at Mach 3 (45,000 ft) (13,700 m), CSJ boost (rocket-off) along the airbreather flight path to test Mach number, climb at test Mach number to $(L/D)_{max}$ equilibrium altitude.

The available fuel volume for all versions is the same as that of the basic configuration. However, it was assumed that the volume could be proportioned as desired between LO_2 and LH_2. This would require a variable tankage arrangement within the vehicle.

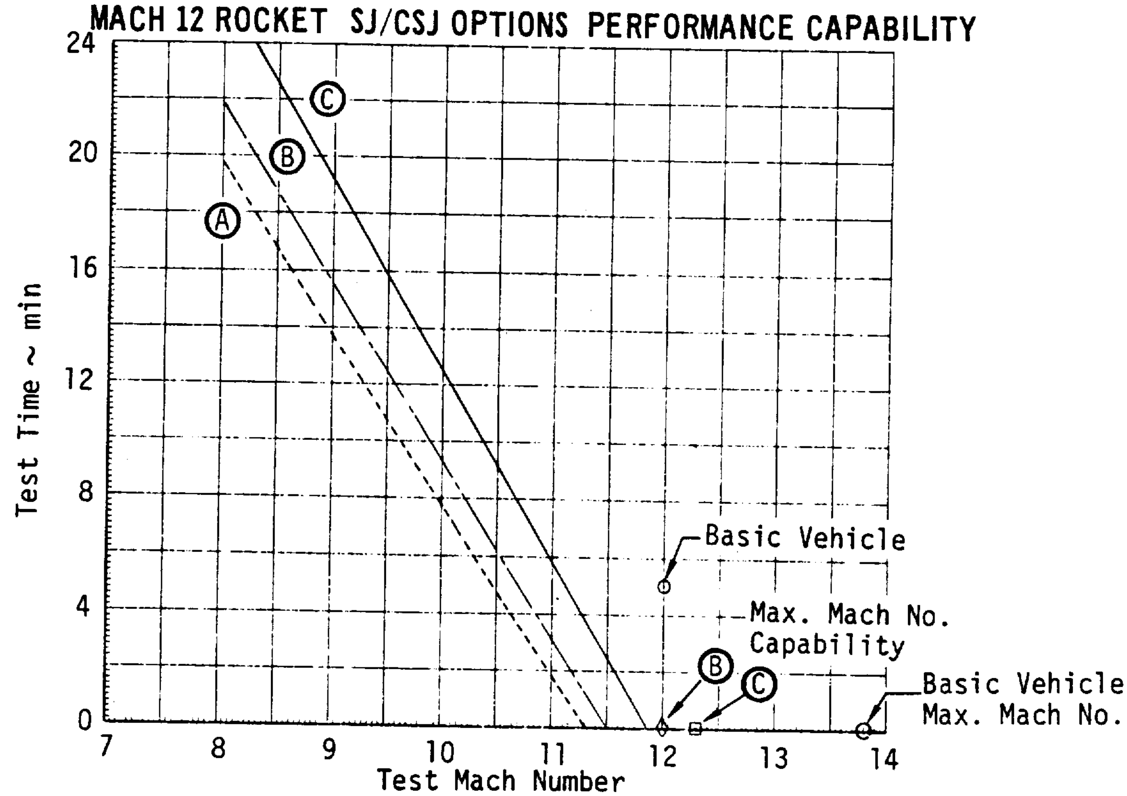
(U) FIGURE 4-65
MACH 12 ROCKET SJ/CSJ OPTIONS CLIMB PROFILES



(U) The performance capabilities are presented in Figure 4-66. For the modes of operation investigated, the best performance is achieved with the CSJ providing the acceleration and climb following a rocket boost to Mach 3. (Condition C) Operating in this mode, the vehicle can provide nearly 26 minutes of cruise at a Mach number of 8, 5 minutes of cruise at a Mach number in excess of 11, and 0 test time, just reaching $(L/D)_{max}$ equilibrium altitude, at a Mach number of 11.8. By continuing to accelerate along the airbreather flight path, rather than climbing to equilibrium altitude, a Mach number of 12.3 can be achieved.

(U) FIGURE 4-66

MACH 12 ROCKET SJ/CSJ OPTIONS PERFORMANCE CAPABILITY



(U) The next best performance is achieved with the SJ/CSJ providing the acceleration and climb with, or without, rocket augmentation following a rocket boost to Mach 8. (Condition B) In either case, i.e. with the rocket continuing to operate throughout the climb, or shut down at Mach 8, results are essentially identical and only one level of performance is shown. Approximately, 22 minutes of cruise time is available at a test Mach number of 8, 5 minutes at a Mach number of 10.7, and $(L/D)_{max}$ equilibrium altitude will just be achieved with 0 test time available at a Mach number of 11.5. If acceleration is continued along the airbreather flight path, rather than climbing to equilibrium altitude, a Mach number of 12 can be obtained.

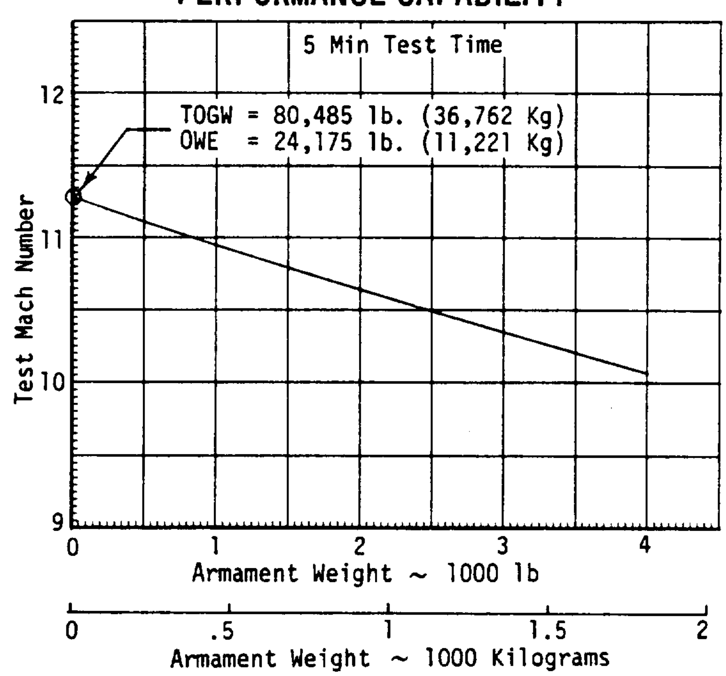
(U) A third level of performance is obtained when the complete boost is accomplished by the rocket along the rocket climb profile and the SJ/CSJ is employed only for cruise. (Condition A) Operating in this mode the vehicle is capable of providing about 20 minutes of cruise at Mach 8, 5 minutes at a Mach number of 10.5 and 0 test time when equilibrium altitude is attained at Mach 11.3.

(U) The performance levels achieved for the various modes of operation give indication of the expanded test capabilities available to the Mach 12 vehicle. The research potential of the basic aircraft will be significantly enhanced by the incorporation of these airbreathing configuration options.

4.5.2.3 (U) Thermal Protection System (TPS) Option - Modifications to provide for the testing of thermal protection system options affect the basic vehicle performance very slightly. No external configuration change is required and therefore, the drag remains unchanged. Only the change in vehicle weight produces a variation in vehicle performance. The effect on the basic vehicle weight due to this installation is an incremental increase of 273 pounds, (124 kg), or about 1% of OWE. Figure 4-61 (OWE weight sensitivity) indicates that for a constant 5 min. of test, a 6.6% decrease in test Mach number results for a 10% increase in OWE. The performance degradation for a 1% increase in OWE is therefore less than 1% and therefore negligible.

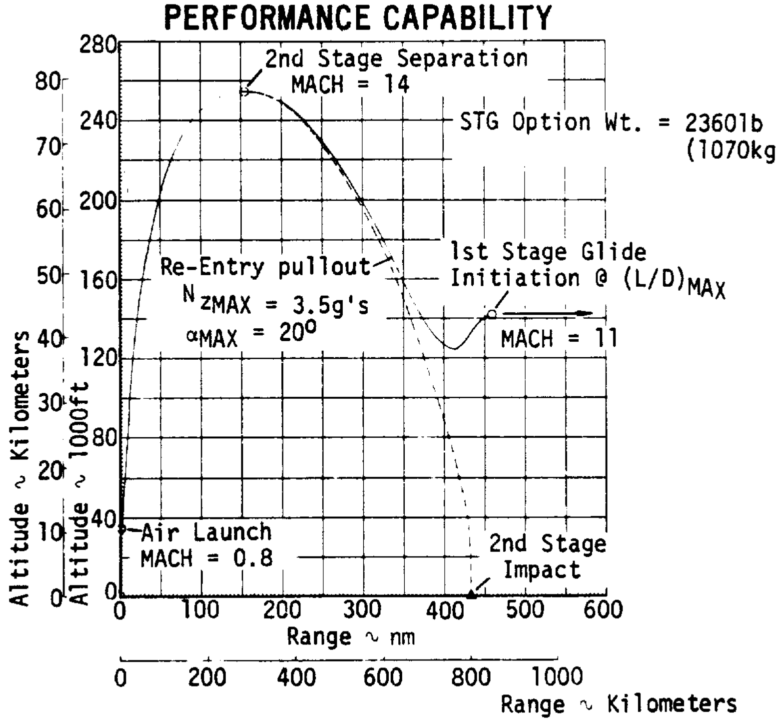
4.5.2.4 (U) Armament (ARM) Option - The performance capability of the vehicle when configured in the armament option is presented in Figure 4-67. This option requires a change to the upper fuselage moldline to provide an armament bay. The performance effects resulting from the increased drag and weight associated with this modification are shown in terms of test Mach number for a constant 5 min test time in Figure 4-67. The Mach number reduction from basic vehicle performance is only 6% for the option alone (no missiles included). The effect of varying missile weight is also shown.

(U) FIGURE 4-67
MACH 12 ROCKET
ARMAMENT OPTION
PERFORMANCE CAPABILITY

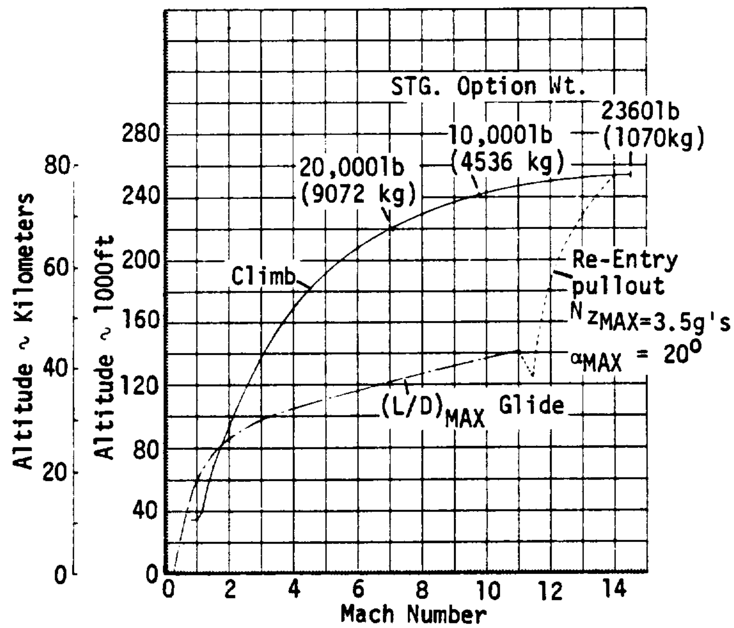


4.5.2.5 (U) Stage (STG) Option - Stage separation techniques similar to those required for space transportation systems and recoverable booster operations can be investigated by modifying the basic vehicle to accommodate an instrumented upper stage. The performance capabilities of the STG option were investigated with regard to attaining flight conditions at separation approximating those currently anticipated for the space transportation system; 250,000 ft (76,000 m) and 11,500 ft/sec (3,500 m/sec). The maximum altitude trajectory that can be obtained for separation studies without exceeding the basic Mach 12 vehicle thermal or structural constraints during the recovery pullout is shown in Figures 4-68 and 4-69. Vehicle angles of attack were arbitrarily limited to a maximum of 20° when the thermal load factor of 3.5 was not constraining. The conditions shown in Figure 4-68 are for the minimum STG option weight of 2,360 lb (1170 kg). For simulation purposes, it is desirable to vary the STG option weight and mass distribution. The effects of varying the STG option weight are illustrated in Figure 4-69 where the reduced altitude and velocity at burnout are shown as a function of the upper stage weight.

(U) FIGURE 4-68
MACH 12 ROCKET
STAGE OPTION
PERFORMANCE CAPABILITY



(U) FIGURE 4-69
MACH 12 ROCKET
STAGE OPTION
ALTITUDE – MACH NO. PROFILE



(U) To completely simulate the separation characteristics of a full-scale system, it is necessary to duplicate the full-scale Mach number, Reynolds number, and Froude number with the scaled models, which in this case are the basic Mach 12 vehicle and the STG option. It is not possible to achieve all three similarity conditions simultaneously. Of the three parameters, the Froude number, which relates the vehicle mass and inertia forces to the air loads, is deemed the most important insofar as separation characteristics are concerned.

$$\text{Froude number} = \sqrt{\frac{V^2}{\ell g}}$$

$$\frac{V^2_{MODEL}}{V^2_{FULL-SCALE}} = \frac{V^2_M}{V^2_{FS}} = \frac{\ell_M}{\ell_{FS}} = \text{SCALE FACTOR} = \text{SF}$$

If the centrifugal relief effects are neglected for simplification, the following scaling laws apply:

$$V_M = V_{FS}(SF)^{1/2}$$

$$W_M = W_{FS}\,\frac{\rho_M}{\rho_{FS}}\,(SF)^3 \text{ since } V^2 \approx \frac{W}{SC_L}\;\frac{2}{\rho}$$

$$I_M = I_{FS}\,\frac{\rho_M}{\rho_{FS}}\,(SF)^5 \text{ since } I = k^2W$$

(k = radius of gyration)

These scaling relationships can be illustrated by an example. For instance, a typical variation of the ratio of upper stage weight to lower stage weight (W_2/W_1) is shown for a representative space shuttle concept during first stage boost in Figure 4-70. The actual full-scale first and second stage weights are shown in Figure 4-71 as a function of W_2/W_1. A similar weight relationship can be provided by the Mach 12 vehicle with the STG option as illustrated in Figure 4-71. Here it is assumed that fuel will be off-loaded from the basic Mach 12 vehicle to achieve the desired separation flight conditions at burnout and that the weight of the STG option is adjusted to the value required as illustrated in Figure 4-71. The weight scaling between the model and full-scale articles is also shown in Figure 4-71. Using the numerical subscripts 1 and 2 to denote first and second stages respectively, the weight scaling shown is based on the following:

$$\frac{W_M}{W_{FS}} = \frac{(W_1 + W_2)_M}{(W_1 + W_2)_{FS}}$$

$$= \frac{W_{M_1}}{W_{FS_2}}\left[\frac{1 + (W_2/W_1)_M}{(W_1/W_2)_{FS} + 1}\right]$$

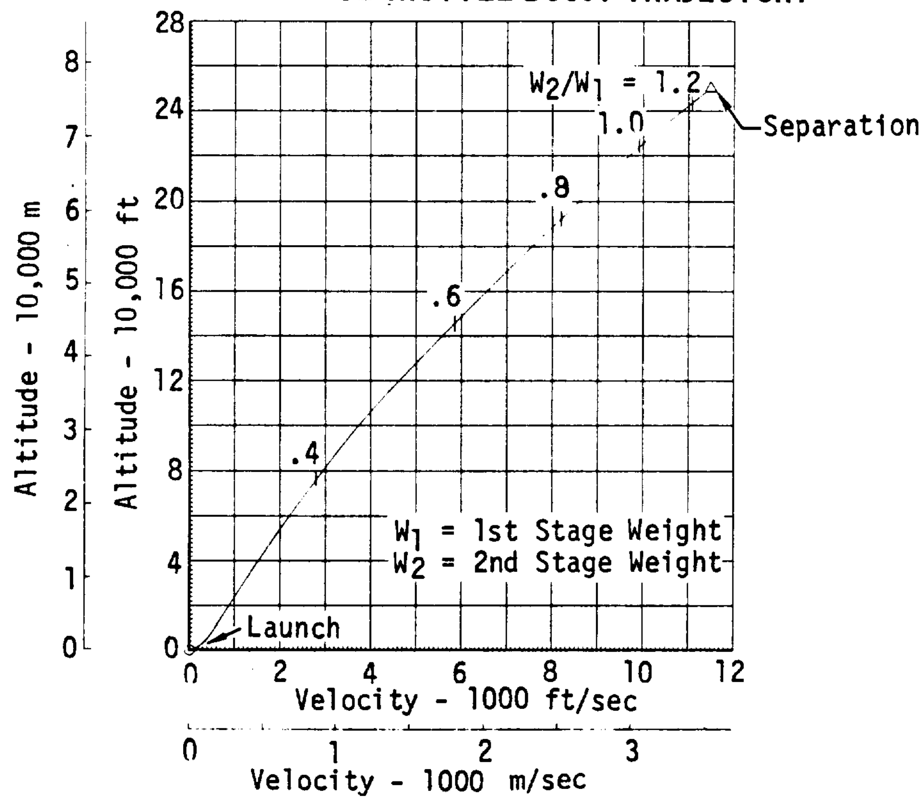
$$= \frac{24,538}{650,000}\left[\frac{1 + (W_2/W_1)_M}{1 + \frac{1}{(W_2/W_1)_{FS}}}\right]$$

$$= .0377\,\frac{1 + (W_2/W_1)_M}{(W_2/W_1)_{FS} + 1}\;\left(\frac{W_2}{W_1}\right)_{FS}$$

$$= .0377\left(\frac{W_2}{W_1}\right) \text{ since } \left(\frac{W_2}{W_1}\right)_{FS} = \left(\frac{W_2}{W_1}\right)_M$$

Knowing the geometric scale factor (SF) involved, the ratio of upper to lower stage weights (W_2/W_1) at the separation point of interest, and the corresponding full-scale velocity and altitude (Figure 4-70) permits the determination of the velocity and altitude requirements for simulation. These are shown in Figure 4-72 for assumed geometric scale factors of 25%, 33%, and 40%. The recovery ceiling shown represents the maximum altitude from which the basic vehicle can negotiate a pull-out without exceeding a 20° angle of attack, 3.5 g normal load factor, thermal constraints, and design dynamic pressure limitations. The maximum velocity capability shown in Figure 4-72 is computed on the basis of a full fuel load.

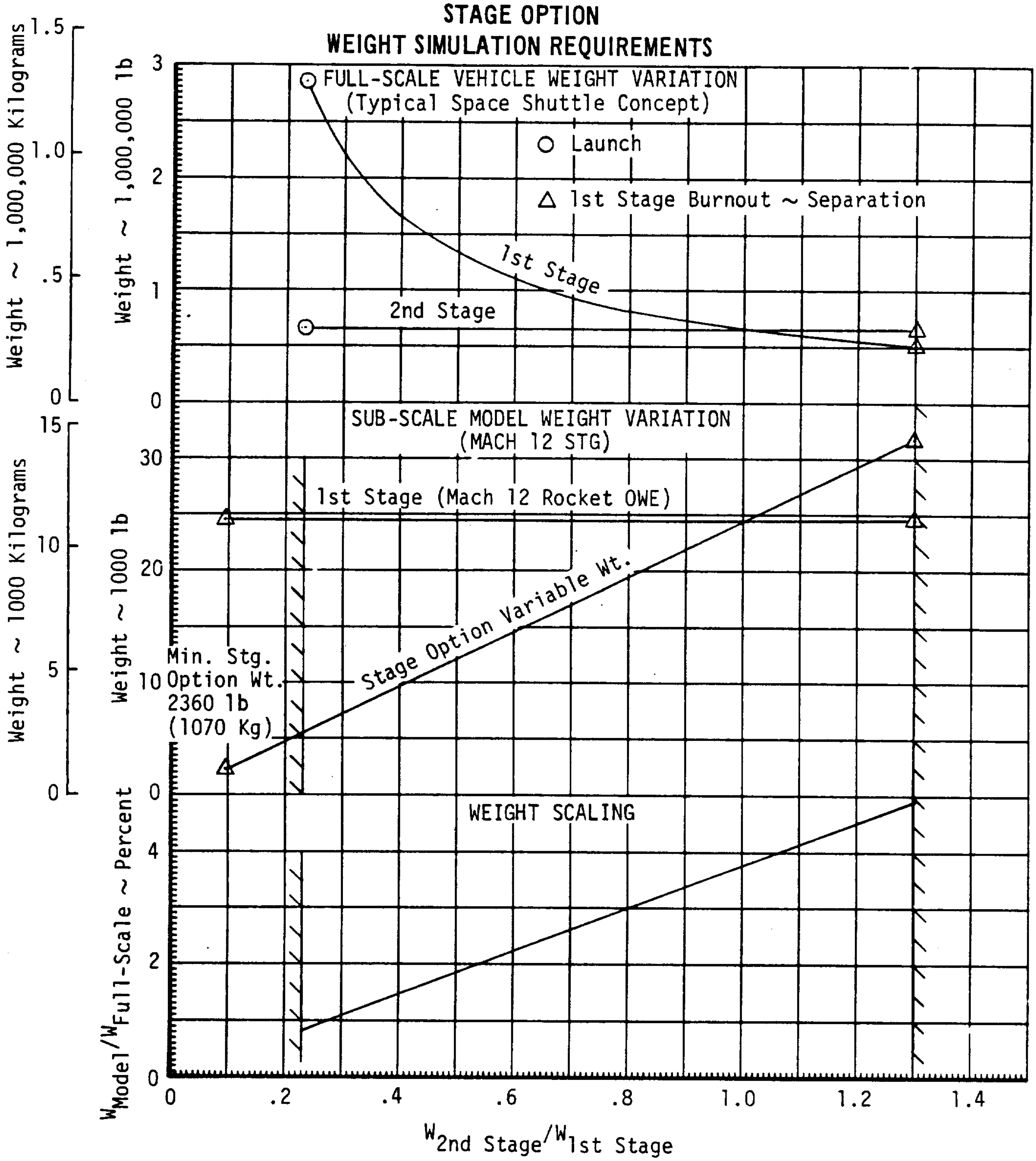
(U) FIGURE 4-70
TYPICAL SPACE SHUTTLE BOOST TRAJECTORY



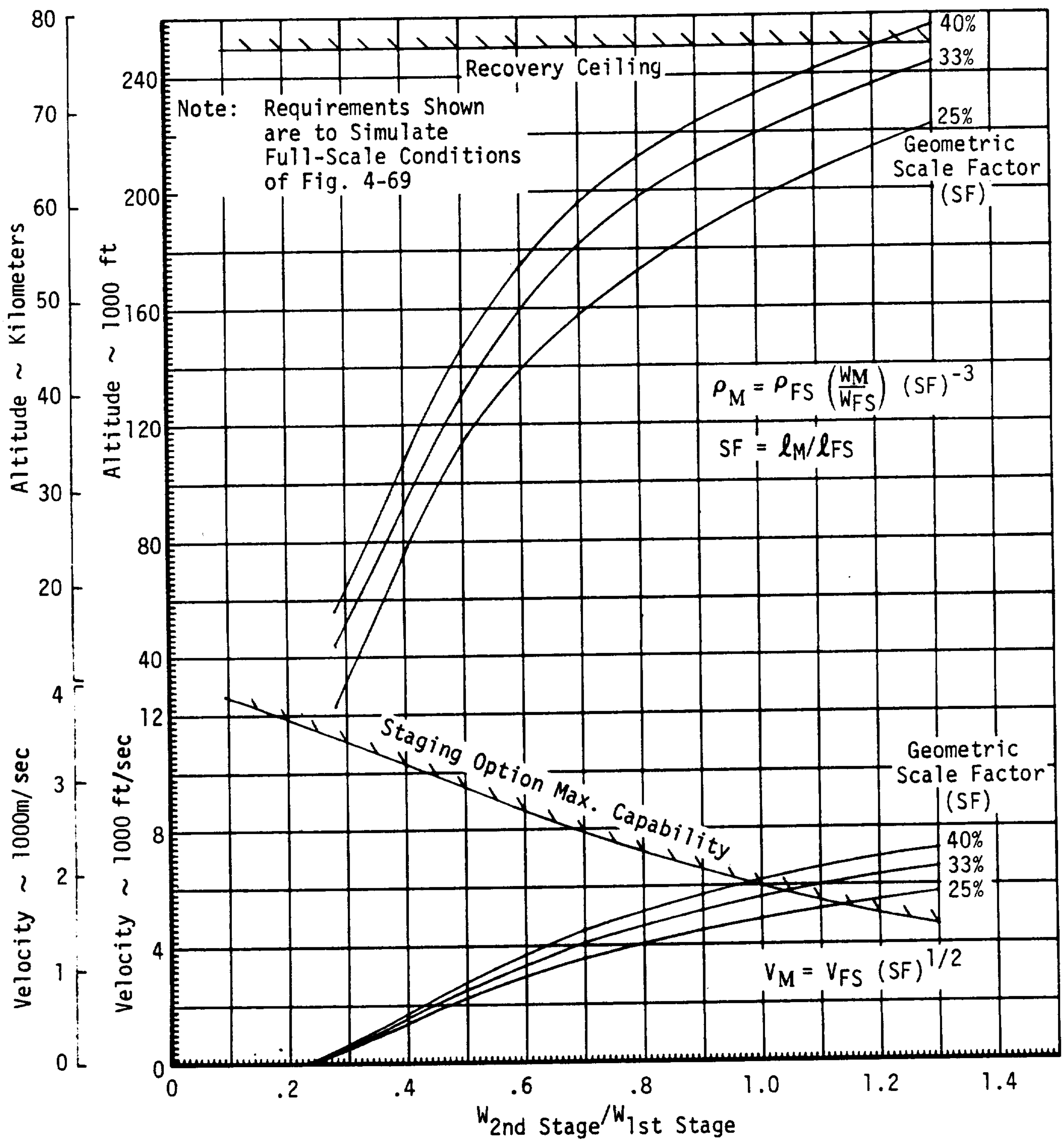
(U) The mass distribution for the Mach 12 vehicle and the STG option must also be the same as that corresponding to their full-scale counterparts.

4.5.2.6 (U) Subsonic Turbojet Research (TJ) Option - The capability of the basic vehicle to operate in a subsonic research mode is desirable to provide for studies involving low speed handling qualities relative to takeoff and landing performance. Two concepts were considered. The first concept called for the addition of two under carriage aft mounted turbojet engines (Figure 4-21). These were to be utilized in place of rocket power for subsonic investigations. However, because of the rather poor simulation of the basic vehicle characteristics, high cost, and the overall complexity of this design, an alternate concept appears much more attractive. This alternate concept involves incorporating the horizontal take off option discussed in Section 4.5.2.1. Operation in this mode will provide the takeoff and landing capability and low speed research potential required.

(U) FIGURE 4-71
MACH 12 ROCKET
STAGE OPTION
WEIGHT SIMULATION REQUIREMENTS

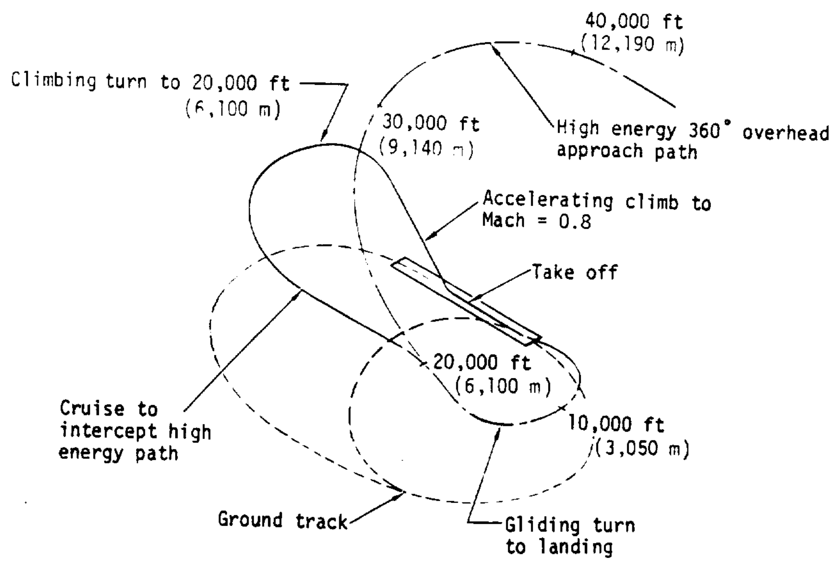


(U) FIGURE 4-72
MACH 12 ROCKET
STAGE OPTION
ALTITUDE AND VELOCITY REQUIRED



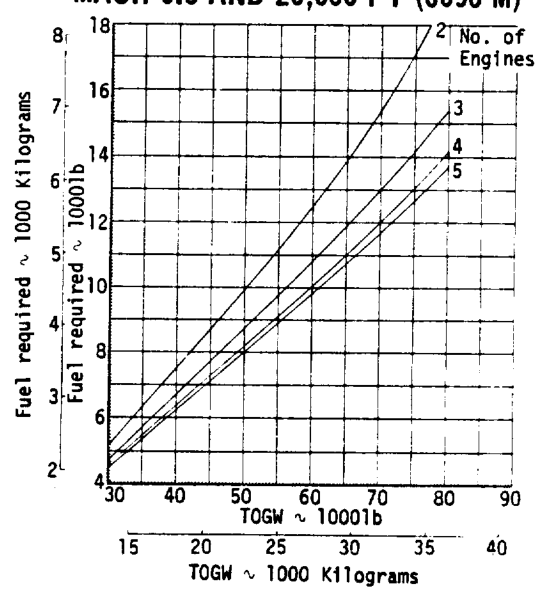
(U) Although normally air launched, the Mach 12 vehicle can perform horizontal takeoffs and operate briefly at subsonic speeds when the HTO option is incorporated. This configuration, when fully loaded with fuel, has the capability of performing three rocket powered takeoffs and unpowered landings. The mission is accomplished by employing a maximum performance takeoff, followed by an acceleration and climb to Mach .8 at 20,000 ft (6100 m). During the climb a 180° turn is executed as illustrated in Figure 4-73 to intercept the downwind portion of a 360° high energy overhead approach pattern. The engines are then shut down or throttled back to minimum idle for the power-off glide to touchdown. A touch and go landing is performed and the entire maneuver is repeated. This maneuver was investigated employing 2, 3, and 4 engines, as well as all 5 available. Figure 4-74 presents a plot of the fuel required to reach Mach .8 and 20,000 ft (6100m) as a function of the number of engines employed and the TOGW. The use of all 5 rocket engines results in the least fuel required to achieve the selected conditions.

(U) FIGURE 4-73
MACH 12 ROCKET
SUBSONIC RESEARCH MISSION



4.5.2.8 (U) Alternate Engine (J2S) Option - The J2S rocket engine is an existing alternate propulsion system that can be employed to power the basic vehicle. It was not selected as the primary propulsion system because the 5 RL10-A-3-9 rocket engine arrangement appeared more attractive from several standpoints. Among these are lateral control and longitudinal acceleration during boost, and throttling capability for efficient steady state testing during cruise. The multiple engine installation of the RL10-A-3-9 system will provide for lateral control when aerodynamic control is insufficient, as for example during VTO operations. The large thrust-to-weight ratio of the J2S will result in high longitudinal accelerations whereas the selected engine configuration will provide a more favorable acceleration environment for the pilot. During cruise, 4 of the 5 engines can be shut down with the remaining engine throttled back to approximately 30% of full thrust providing more efficient operation than that available with the J2S.

(U) FIGURE 4-74
MACH 12 ROCKET
SUBSONIC RESEARCH OPTION
FUEL REQUIRED TO ACCELERATE AND CLIMB TO MACH 0.8 AND 20,000 FT (6096 M)



4.6 THERMODYNAMICS

(U) An assessment of the local flow field has shown that the external flow over the surface of the Mach 12 research vehicle is predominantly turbulent (transition based upon a value of 150 for the ratio of momentum Reynolds number to local Mach number). However, aerodynamic heating rates are sufficiently low to allow radiation cooling of external surfaces with extensive use of conventional and superalloy materials and a small amount of coated columbium and tantalum shingles. These material distributions enhance the performance capability of the aircraft, allowing a normal 3.5 g maneuver at Mach 12, as well as recovery from a staging altitude of 255,000 feet (78,000 m) at Mach 14.5 in performing stage separation research applicable to current space transportation system concepts.

(U) The interior of the aircraft (crew, equipment, fuel, and structure) is isolated from the external thermal environment by an active thermal protection system (TPS). A water wick system which makes use of the integrated performance of radiation cooled shingles, high temperature insulation, radiation barrier, water evaporation, and cryogenic tankage insulation was selected on the basis of its low unit weight, high volumetric efficiency and confidence in the attainment and reliability of such a system.

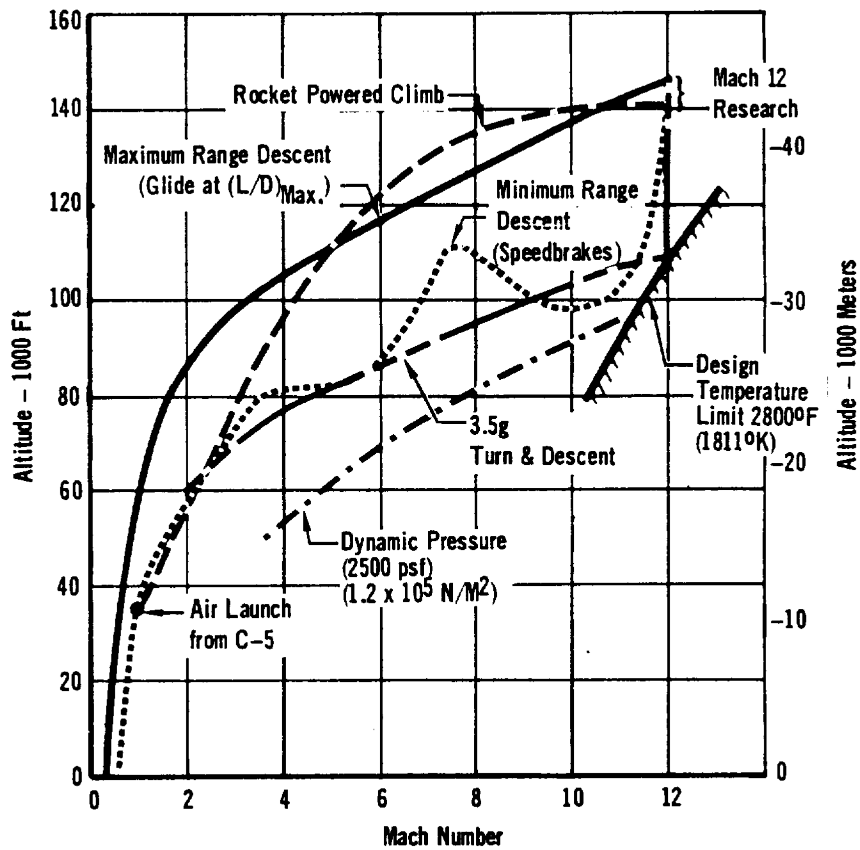
(U) A ranking of research objectives by the scientific community has clearly indicated the importance of re-usable TPS to future acquisition of operational hypersonic systems. In addition, a number of specialists were contacted in order to identify other potential thermal protection systems of interest and their ideas on how the Mach 12 vehicle could be used in conducting research for such systems. As a result, the selected approach provides a 5 ft (1.53 m) long, 6 inch (15.2 cm) deep bay around the complete periphery of the aircraft plus adequate volume for storage of associated equipment, expendables and special instrumentation. This option provides nearly unlimited flexibility to perform long term development and demonstration tests with full scale hardware. Both passive and active, cryogenic and non-fuel systems can be subjected to the thermal and mechanical environment characteristic of operational systems.

(U) Regenerative cooling studies show that the test engine sizes selected for performing airbreathing propulsive research can be efficiently cooled (equivalence ratio of one or less), using the liquid hydrogen fuel. Thus, flight test of research vehicle engines will be a valid qualification for prototype and/or first generation operational use.

4.6.1 BASIC VEHICLE

(U) Vehicle Temperatures - Moldline materials (shingles and insulation) for the basic Mach 12 vehicle have been selected to be compatible with maximum temperatures anticipated for the flight profiles presented in Figure 4-75. No significant change in moldline materials is required when the basic vehicle is reconfigured to perform the various research options (TPS, propulsion, armament, staging, etc) considered during this study.

(U) FIGURE 4-75
FLIGHT PROFILES FOR MACH 12 RESEARCH
(Basic Mach 12 Rocket Aircraft)



(U) Maximum temperatures and corresponding shingle materials are presented in Figure 4-76. As shown by this figure, 79% of the surface experiences maximum temperatures less than 2400°F (1590°K) permitting extensive use of conventional and superalloy materials. 20% of the remaining surface area reaches temperatures that are compatible with the use of coated columbium, and only 1% of the area requires the use of higher temperature materials such as tantalum.

(U) Normal flight operations do not require maneuvering flight thus external surface temperatures will be significantly less than the maximum design values presented herein. For example during a nominal Mach 12 test run the surface temperatures are 400°F (476°K) to 600°F (590°K) less than maximum design temperatures.

(U) Maximum surface temperatures (radiation equilibrium values with an emissivity of 0.8) for non-stagnation regions were determined based upon the turbulent heating correlation of Spalding and Chi. Upper surface and speedbrakes achieve their maximum temperature during the minimum range descent at Mach 11.4 and 107,000 feet (32,620 m). Lower surface and delta tip temperatures are a maximum during the 3.5 g turn, specifically at the Mach 12, 110,000 feet (33,550 m) and 9 degree angle of attack condition. Nose tip and leading edge heating rates, as predicted by the Fay and Riddell correlation, are also a maximum at this 3.5 g turn condition.

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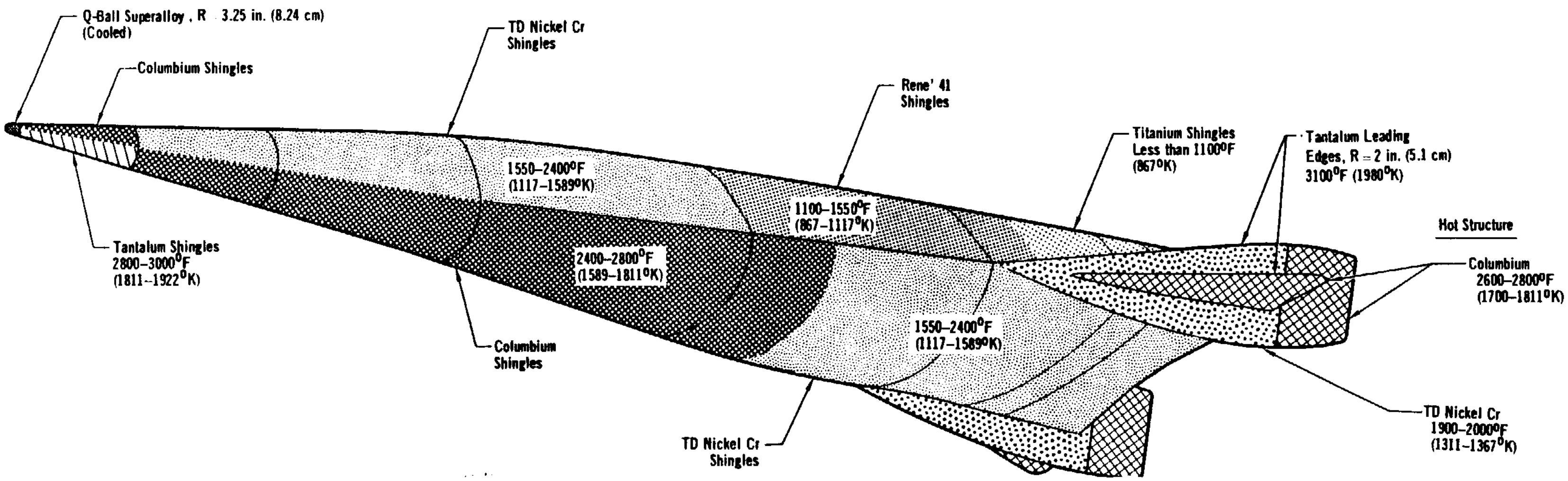
(U) FIGURE 4-76
MAXIMUM SURFACE TEMPERATURES
(Basic Mach 12 Rocket Aircraft)



Surface Material Distribution

Material	% of Surface Area
Titanium	4
Rene' 41	21
TD Nickel Cr	54
Columbium	20
Tantalum	1

FOLDOUT FRAME 1

FOLDOUT FRAME 2

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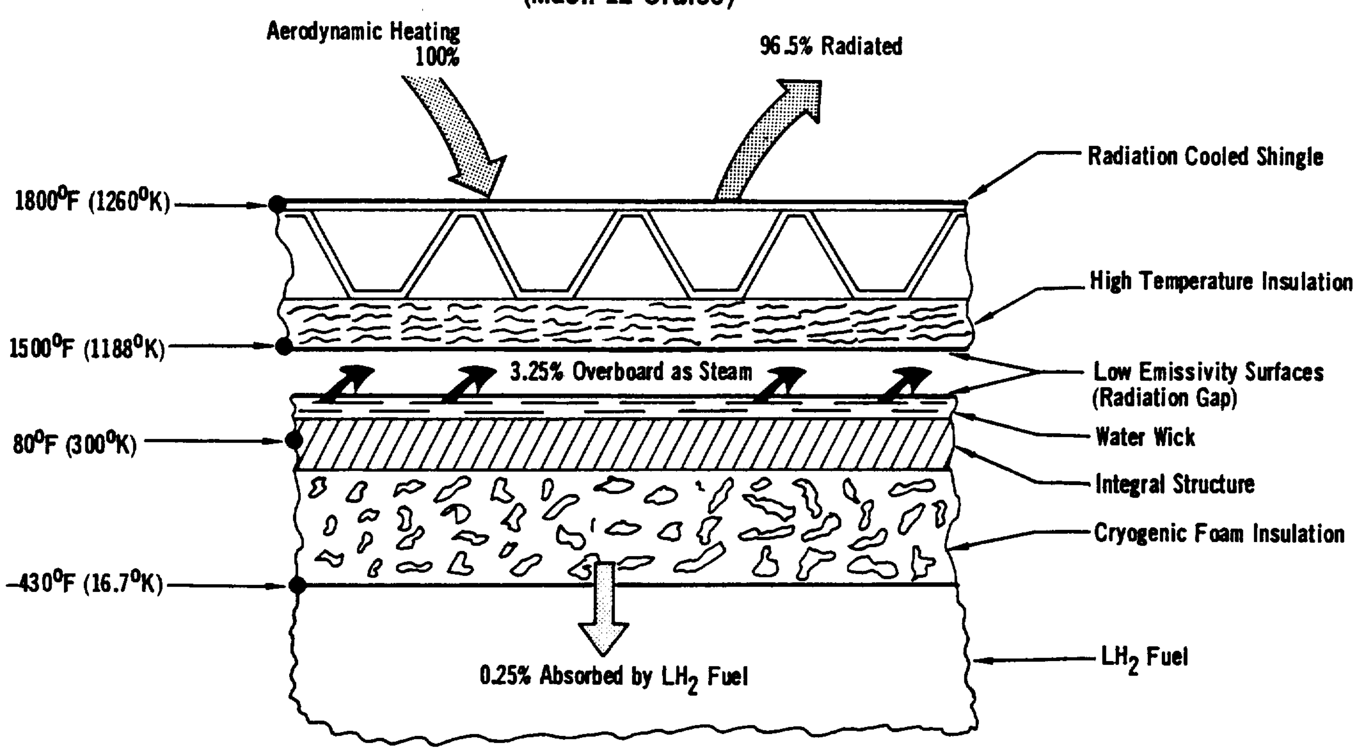
The nose tip is a cooled superalloy Q-Ball attitude sensor of the X-15 type. It is approximately 6.5 inches (16.5 cm) in diameter and will be subjected to a maximum aerodynamic heating rate of 110 Btu/ft^2 sec (1.25×10^6 watts/m^2) at a wall temperature of 1550°F (1120°K). Tantalum leading edges (2 inch, 5.1 cm, radius) on the vertical fins and delta tip controls will approach a maximum temperature of 3100°F (1980°K).

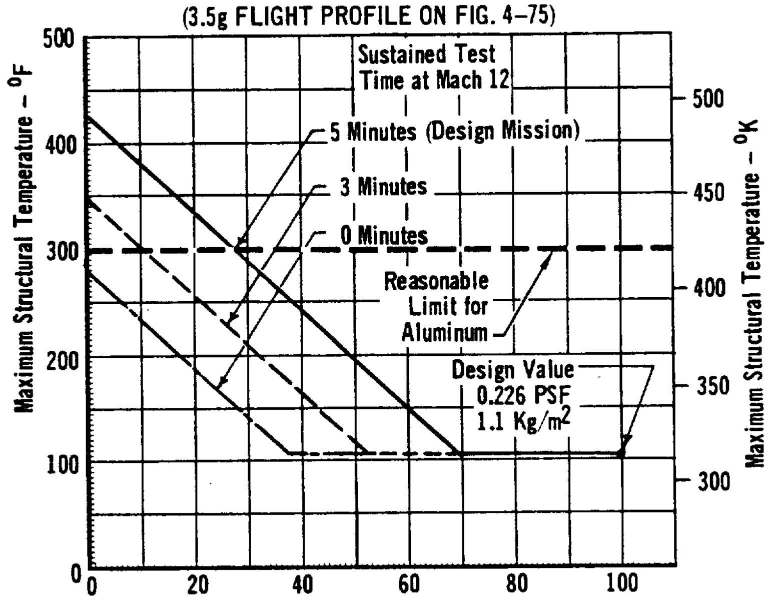
(U) Thermal Protection System - Phase II tradeoff studies resulted in the selection of an active TPS (water wick concept) to isolate the interior of the aircraft from the previously defined Mach 12 environment. Even though external surface temperatures experience large variations with vehicle location and flight conditions, aluminum structural temperatures are uniform throughout the aircraft and nearly constant (approximately room temperature) throughout the flight. The proposed system takes advantage of the high heat of vaporization of water as a heat sink and the judicious usage of surface coatings (high external surface emissivity and low internal surface emissivities on both sides of a radiation gap). The combination of high temperature insulation, radiation gap, and emissivity control significantly reduces the heat transfer to the water, thereby resulting in an active cooling system of low penalty in terms of weight and volume. The water wick system, is illustrated in Figure 4-77 and indicates the heat flux distribution during steady state Mach 12 cruise, for a typical lower surface, cryogenic fuel area. As indicated, utilization of a high emissivity (0.8) coating enables 96.5% of the convective heat flux to be radiated away. Of the 3.5% incident heat flux penetrating the high temperature insulation, 3.25% is absorbed in boiling off water at the prevailing vaporization temperature of approximately 80°F (300°K). Only 0.25% of the incident heat flux passes through the primary structure and foam insulation to be absorbed by the cryogenic fuel

(U) FIGURE 4-77
STEADY STATE THERMAL CHARACTERISTICS OF WATER WICK TPS
(Mach 12 Cruise)



(U) As can be seen in Figure 4-78, maximum structural temperatures (design value of about 100°F, 311°K) upon landing are only 20°F (11°K) higher than the steady state cruise temperature of 80°F (300°K). That is, even though the prevailing vaporization temperature steadily increases during descent (212°F, 375°K, upon landing), structural temperatures increase only slightly because of the compensating decrease in the external heating environment.

(U) FIGURE 4-78
EFFECT OF INITIAL WATER BLANKET SATURATION LEVEL ON MAXIMUM STRUCTURAL TEMPERATURES
(3.5g FLIGHT PROFILE ON FIG. 4-75)

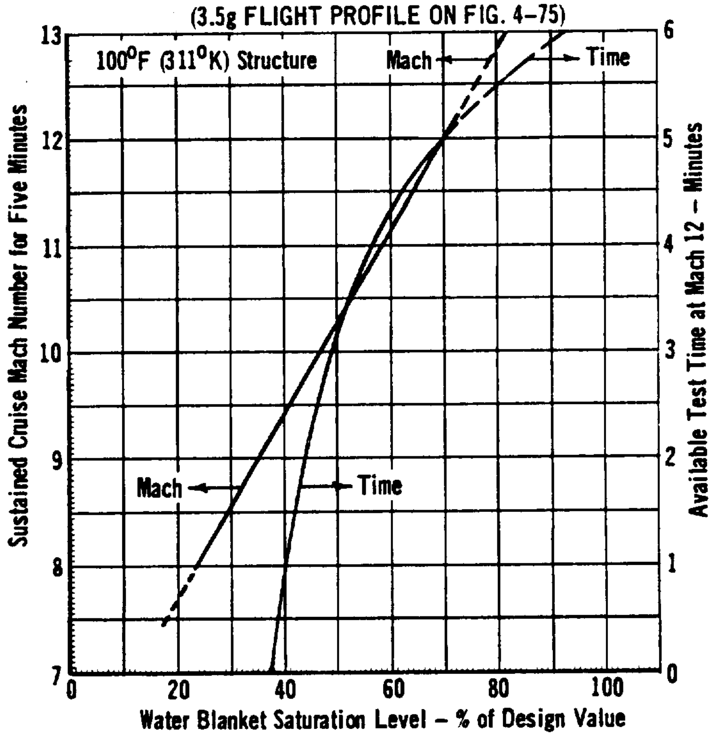


(U) Even though the Mach 12 research vehicle is configured with a water wick TPS, it can be used as a test bed for flight testing passive and active TPS concepts for other aircraft and spacecraft (see Section 4.6.2).

(U) A study was performed to determine the effect of initial water blanket saturation level on structural (or fuel tank wall) temperatures. The heat pulse imposed upon a representative lower surface location, during 5 minutes of Mach 12 cruise followed by a 3.5 g turn, was established. The system was sized to absorb this heat load with a 30% safety factor in water weight. A design value of water weight corresponding to 0.226 psf (1.1 kg/m^2) which represents a 100% blanket saturation level was determined. As shown in Figure 4-78, if the blanket is filled to 70% of the design value or greater, sufficient water is available to absorb the imposed heat load while holding the maximum structural temperature to about 100°F (311°K). Figure 4-78 shows the maximum structural temperatures that would be attained from exposure to the same heat pulse with the blanket initially filled to levels less than 70% of the design value. These temperatures were determined by accounting for total evaporation prior to the conclusion of the heat pulse while no consideration was given to heat transferred through the aluminum structure from the dry area to adjacent areas where excess water exists. This latter effect would minimize any locally sharp temperature increase. In the extreme situation, characterized by a local absence of water in the blanket initially and no lateral heat transfer by conduction through the aluminum structure to adjacent areas, the structure conceivably could reach a local maximum temperature of 425°F (491°K) resulting in permanent damage. If the blanket were initially filled to less than 38% of the design value, structural wall temperatures in excess of 250°F (394°K) result, which in fuel tank areas is sufficient to degrade the cryogenic foam insulation. While these effects are undesirable, isolated instances could be tolerated without catastrophic failure to the aircraft.

(U) Reduced test time effects at Mach 12 are reflected in Figures 4-78 and 4-79. Three minutes of test time with structural temperatures not significantly exceeding 100°F (311°K) can be achieved by filling the blanket to 52% of its design saturation level. An initial saturation level of 37% is sufficient to absorb the heat loads resulting from the boost, 3.5 g turn, and descent phases of the flight profile (no sustained test time at Mach 12). Therefore, a reasonable learning process associated with water refurbishment procedures could be tolerated, consistent with normal flight test planning of gradually increasing test speed and times to the design values. Figure 4-79 depicts water requirements for 5 minute test times at reduced speeds. Approximately 23% and 47% of the Mach 12 design value of water would be sufficient to absorb the heat load associated with 5 minutes of testing at Mach 8 and Mach 10, respectively. Since the basic system is 30% oversized, significant overshoots in speed could be tolerated. The 100% blanket saturation level also permits extension of the allowable test time to more than 6 minutes at Mach 12.

(U) FIGURE 4-79
RESEARCH CAPABILITY AS A FUNCTION OF INITIAL WATER BLANKET SATURATION LEVEL
(3.5g FLIGHT PROFILE ON FIG. 4-75)



(U) As mentioned previously, the above results are based upon the 3.5 g maneuvering (180 degree heading change) descent of Figure 4-75. For those cases when a head-on approach is possible (i.e., no 3.5 g turn), the descent will fall between the two extremes, namely, the maximum range, $(L/D)_{max}$ glide, and minimum range, speedbrake, descent defined in Figure 4-75. Following a minimum range descent path, upon completion of a 5 minute Mach 12 test, results in a less severe (shorter) heating profile and water blanket saturation level (see Figure 4-78) than the 3.5 g design case considered previously. The most probable head-on approach, that is, a nominal descent, results in maximum structural temperatures that are approximately 50°F (28°K) higher than the results of Figure 4-78 and requires an initial saturation of only 78% to limit the structure to 100°F (311°K). The most severe (longest) heating profile occurs when the maximum range, $(L/D)_{max}$ glide, descent is required. For this limiting condition, maximum structural temperatures are about 130°F (72°K) higher than the results of Figure 4-78 and a 93% water blanket saturation level is required to limit the structure to 100°F (311°K).

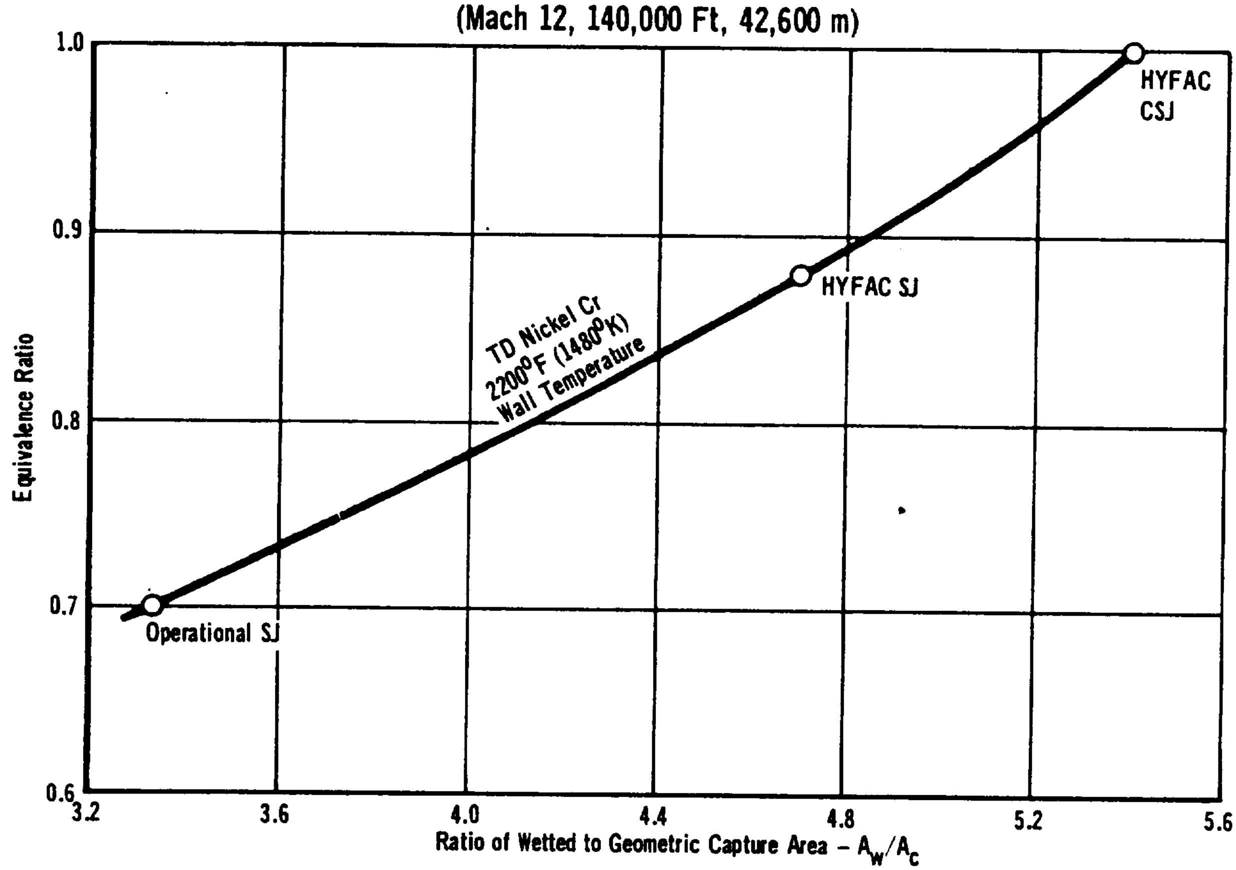
(U) In summary, the results of this study show that the water wick system can be oversized by 30% or more with a negligible effect upon vehicle performance. This margin of safety enhances the vehicle design by allowing for growth capability, unforseen emergencies, and incomplete filling of the water blankets. Concerning the latter, it was found that initial water wick saturation levels would have to be less than 30% of the design value to exceed a structural temperature of 300°F (422°K), which is not an unrealistic design temperature for aluminum.

4.6.2 (U) CONFIGURATION RESEARCH OPTIONS - The thermal aspects of reconfiguring the basic vehicle to extend its research capability are discussed below.

(U) Scramjet/Convertible Scramjet Options - Analytical studies were performed to determine regenerative cooling requirements for the scramjet and convertible scramjet test modules. The primary purpose of this study was to determine if the test modules, once developed, could be applied to operational systems. Past studies (Reference (18)) have shown that engine cooling has a significant impact upon the design and operation of airbreathing cruise aircraft and hence it had to be demonstrated analytically that the test modules could be efficiently cooled. For a regenerative cooling system, where all of the fuel used as a coolant is available for propulsion, the combustion equivalence ratio and the cooling equivalence ratio must be equal. Therefore, to determine the operating point for a given vehicle, it is necessary to match the coolant flow rate required to limit the temperature of the regeneratively cooled surfaces with the fuel flow rate required to achieve the desired thrust. Based upon results obtained from previous studies, for example, Reference (18), a TD NiCr plate-fin heat exchanger (0.10 inch, 0.254 cm, square core; 2200°F, 1480°K, maximum temperature) was selected as representative of the cooling panels which will be employed on Mach 12 scramjet or convertible scramjet engines. For such small core heat exchangers, overall cooling efficiencies of 80% or better can be achieved. A value of 82% was used in obtaining coolant flow rates during this study. As used herein, cooling efficiency is defined as the ratio of actual to ideal coolant temperature rise. The ideal or maximum possible temperature rise is that hypothetical case where the hydrogen temperature at the heat exchanger exit is equal to the wall temperature. Although the research vehicle test modules have more wetted area per unit capture area than some of the operational engines studied in the past, they can still be cooled without exceeding an equivalence ratio of 1.0 and a wall temperature of 2200°F (1480°K), as shown in Figure 4-80.

This, then, results in efficient cruise economy and simulates operational system usage. These equivalence ratios are for the 1 g cruise condition. The scramjet and convertible scramjet test modules are retracted and inoperative during the maneuver condition. The ramjet test module is not operated above Mach 6. Its equivalence ratio is 0.5.

(U) FIGURE 4-80
PROPULSIVE RESEARCH/ENGINE COOLING REQUIREMENTS
(Mach 12, 140,000 Ft, 42,600 m)



(U) The development of a regenerative cooling system will involve considerable testing activity. The two primary areas of attention are verification of the hot gas heat transfer and thermal/structural performance of the heat exchanger. Engine heating data would be obtained from scaled engine tests conducted in present and planned wind tunnel facilities. Heat exchanger characteristics would be determined by laboratory testing of full scale panels. Flight heating rates would be simulated by the use of quartz lamps or other improved methods. Development of a heat source capable of simulating a heating rate of 500 Btu/ft^2 sec (5.67×10^6 watts/m^2) has been initiated by McDonnell Aircraft. These panel tests would be satisfactory for investigating manufacturing techniques, flow distribution, controls, and performance.

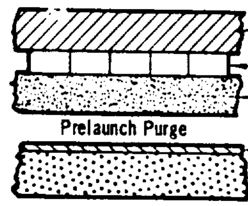
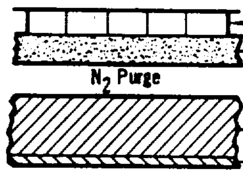
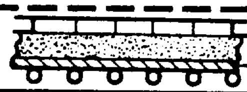
(U) Although tunnel and laboratory testing are a necessary part of the scramjet development cycle, present facilities cannot provide all of the necessary information required for a proper assessment of the technical problems associated with a complete scramjet engine. The inlet, combustor, and nozzle should be tested as a complete unit with actual hydrogen combustion to properly determine the complex flow within the engine. Continuous operating times are required. Current tunnels cannot provide simulation of the flow properties existing at high Mach numbers. The complex shock boundary layer interactions may not scale properly from small scale models to the full scale aircraft. Laboratory tests of full scale panels cannot account for the inter-dependence of the hot gas heat transfer and the coolant side heat transfer. The localized high heating rates due to shock interaction and combustion phenomena cannot be duplicated in the laboratory panel tests. Thus, to insure successful integration and operation of the complete engine cooling system, advanced facilities (see Section 7 of VOLUME IV, Part 2) and/or flight test of a representative engine module is required.

(U) Thermal Protection System (TPS) Option - At the initiation of the HYFAC program, and with the aid of the scientific community, research objectives were identified to denote the areas where additional R&D effort is required to pave the way for the acquisition of future operational hypersonic aircraft systems. A number of research objectives, dealing with the development of reusuable thermal protection systems were identified and were ranked quite high in importance. Although much of the research required to develop such systems can be and would be accomplished in ground facilities, final development and/or demonstration will require flight test.

(U) During Phase III, a number of specialists were surveyed and several thermal protection systems of current interest were identified as noted in Figure 4-81. While not a complete list it does represent a rather significant cross-section of different types of systems. The various vehicle modifications considered in order to provide additional TPS research are indicated in Figure 4-82. The recommended option, (A), reserves a 5 foot (1.53m) long test section around the complete periphery of the vehicle extending from 2 feet (0.61m) forward to 3 feet (0.92m) aft of the forward LH_2 fuel bulkhead. The structure and tank wall were relocated in this area to allow for a 6 inch (15.3cm) deep bay (external moldline to structure/tank wall) for conducting TPS research. In addition, approximately 25 ft^3 ($0.71m^3$) of volume is available on the upper surface immediately behind the test section for storage of equipment, expendables, special instrumentation and controls. Two hundred pounds (90.8kg) was considered an adequate weight allowance for the above provisions and/or any changes relative to the basic system in conducting TPS research. Considerations that lead to the recommendation of TPS option (A) are as follows:

- o TPS testing capability in both cryogenic and non-cryogenic areas.
- o Full scale testing with actual vehicle shapes, structure, loads, attachments, gradients, etc.
- o Exposure to actual variations in vehicle heating rates, i.e., lower surface, sides, and upper surface.
- o Capability of conducting TPS research throughout the program without restricting vehicle operations or interfering with other research options. This permits long term TPS tests of potential reusuable thermal protection systems under actual operational conditions.

(U) FIGURE 4-81
CONTEMPORARY TPS – RESEARCH APPLICATIONS

TPS Concept	Primary Interest	Proposed Application
 HCF *; Fiberglass Honeycomb; Insulation; Prelaunch Purge; Tank Wall; Internal Insulation	• NASA • MDC • Lockheed	Shuttle
 Shingle; Insulation; N_2 Purge; N_2 Purge; Small Pore Size Insulation; Tank Wall	NASA Langley	• LH_2 Cruise Aircraft • Shuttle
 Moldline Structure	Bell NASA Langley	• Mach 6 Transport
Convective Cooling  Insulated Structure	Martin Bell	• Shuttle

* Hardened Compacted Fibers

(U) Other alternate TPS options that were considered include (see Figure 4-82).

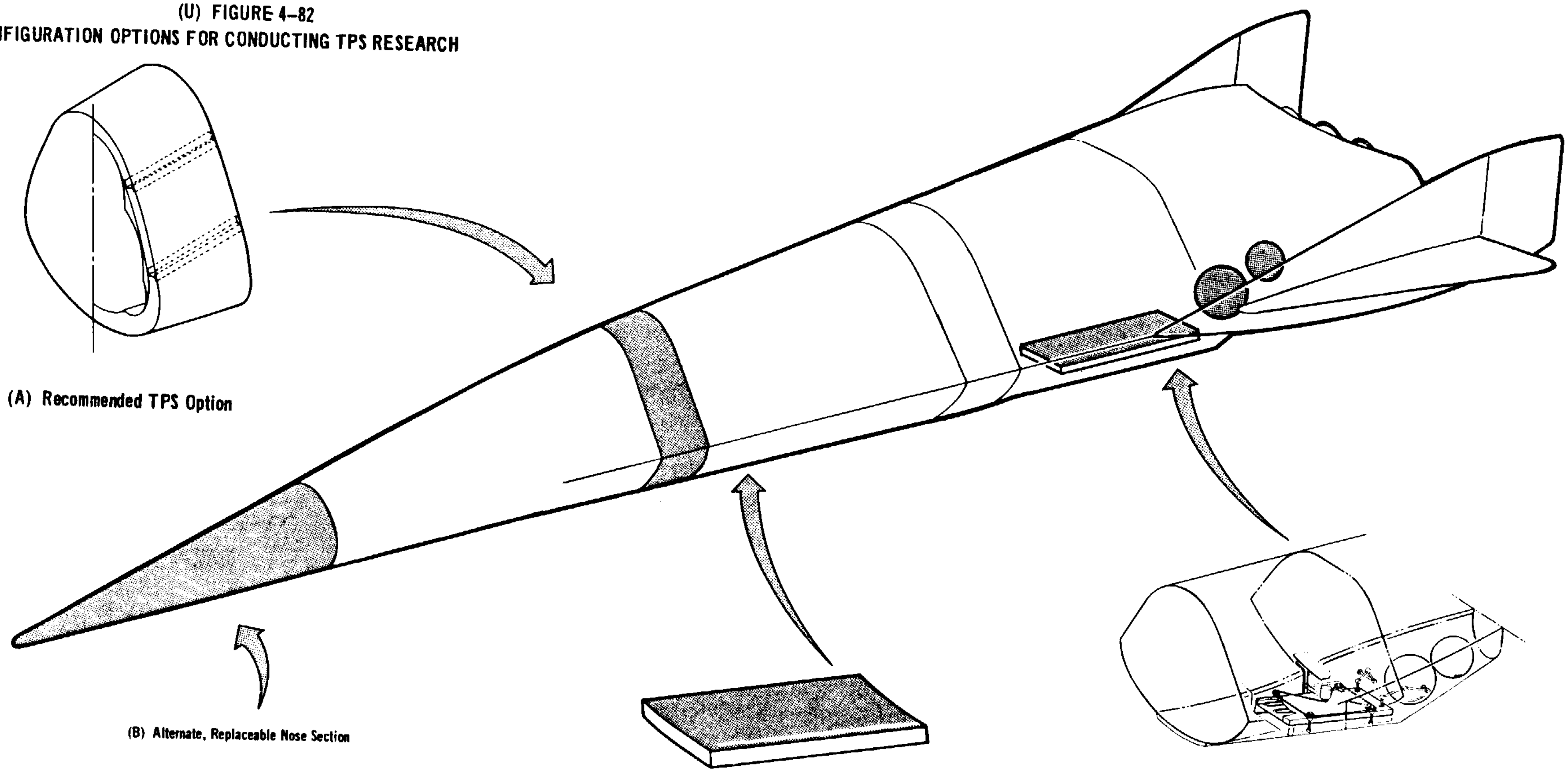
- o Replaceable nose cone - approach (B).
- o Lower surface test panel - approach (C)
- o Add-on TPS test package - approach (D)

Each of these alternate approaches may offer an advantage for conducting specific TPS tests. For example, approach (B) may be preferred for conducting film and transpiration cooling tests, whereas, approaches (C) and (D) which have a flat or nearly flat 4 x 8 ft (1.22 x 2.44m) test section may be preferred for screening tests. Approach (D) would of course change vehicle aerodynamic characteristics and preclude performing TPS and propulsive research on the same flight vehicle.

(U) In addition to its primary objective of conducting thermal protection research, the TPS option provides a suitable test bay which could be used for conducting boundary layer studies. For example, simultaneous measurements of heat transfer and skin friction at controlled wall temperatures, boundary layer trip experiments, and boundary layer surveys using retractable cooled temperature and pressure probes.

(U) Staging (STG) Options - A typical flight profile (from C-5 drop through recovery of the basic vehicle) for performing staging investigations is presented in Figure 4-83. As shown in this figure, heating conditions for the simulated space

(U) FIGURE 4-82
CONFIGURATION OPTIONS FOR CONDUCTING TPS RESEARCH



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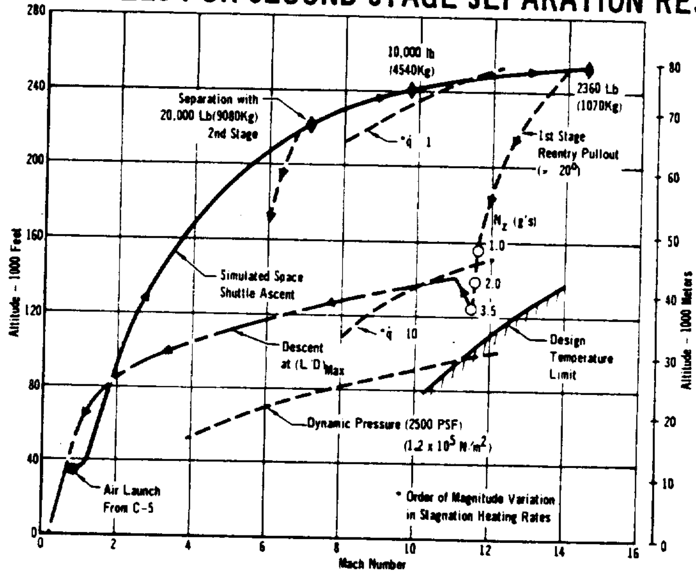
shuttle ascent are considerably lower than those experienced during the boost phase for a nominal Mach 12 test flight. Because of these low heating conditions and the relatively thick skins of the unmanned STG vehicle, skin temperatures will be less than 200°F (367°K) at separation. Since these vehicles are designed as expendable boilerplates, no thermal protection of the STG vehicle is required and conventional construction methods using lightweight materials such as aluminum or fiberglass can be employed.

(U) FIGURE 4-83
FLIGHT PROFILES FOR SECOND STAGE SEPARATION RESEARCH



(U) Although actual aerodynamic heating rates are low during the simulated shuttle ascent, total air temperatures of the order of 7500°F (4425°K) will be experienced in regions where the flow is stagnated. As a precautionary measure to insure against any overheating of the basic vehicle, an ablation fairing has been added to prevent the possibility of locally high heating due to near total temperature air flow in the gap between vehicles. As indicated in Figure 4-83 no additional thermal protection of the basic vehicle is required in that it can recover without exceeding the design temperature limit.

4.7 STRUCTURES

(U) A primary goal of the Phase III refinement effort was to identify structural concepts and materials for the research aircraft with potential for satisfying all operational and functional requirements with maximum reliability and minimum development requirements. The structural concepts that meet these requirements incorporate maximum utilization of currently available materials which have a high strength to weight ratio, a high toughness rating, and are readily fabricated. Additional dominant considerations are weight, cost, methods of inspection, and ease of field repair. Strength analyses conducted during this study and the results of recently completed studies of similar vehicles were used as the basis for selection of the research aircraft structural concepts and materials. The selected concepts and materials do not necessarily represent the lowest structural weight, rather, they are considered to provide low cost, reliable, conventional structure with high potential for providing near minimum weight when optimized.

(U) A basic Mach 12 aircraft and several configuration options were considered during the refinement studies. Basic configuration structural concepts and materials are described in this section. Aircraft components that received major consideration were the primary fuselage structure, fuel tanks, thermal protection system structural components, and control surfaces. Significant features of this vehicle that influence structure selection are the all body shape, rocket engines, LH_2 fuel, airdrop launch mode, and the cruise test speed of Mach 12.

4.7.1 (U) ENVIRONMENT - The aircraft structural design temperatures are based on the flight trajectory shown in Figure 4-75. The maximum surface temperatures occur during the 3.5g maneuver at Mach 12. Maximum design surface temperatures are shown in Figure 4-76.

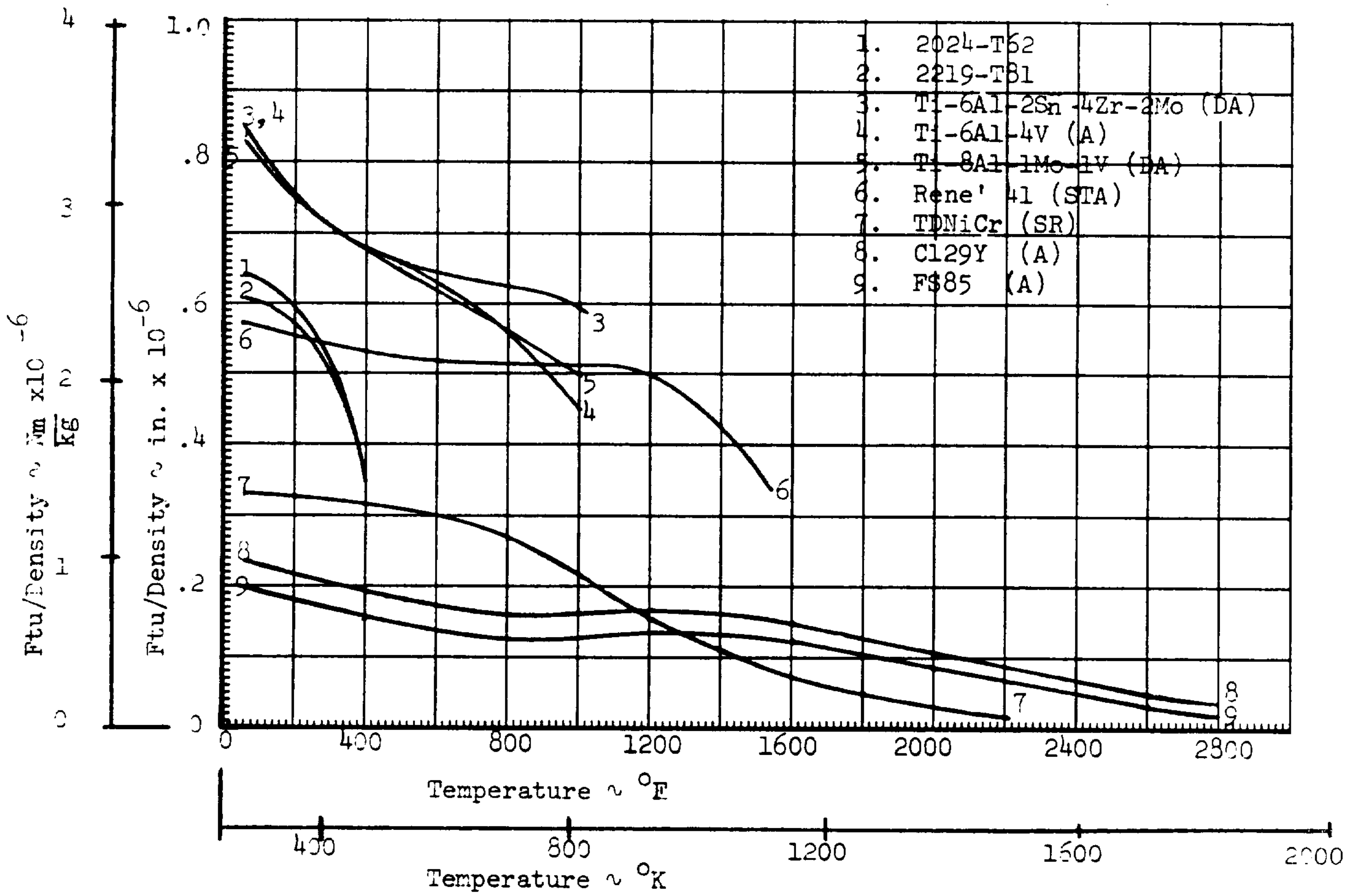
4.7.2 (U) LOADING CONDITIONS - Loads used for concept selection criteria are based on the flight profile (Figure 4-75) and the loading conditions described in MIL-A-8861 (ASG) and MIL-A-8862(ASG). These were used as a guideline in determining the design loads used in Phase III evaluations. The loading conditions of greatest significance are the ground loads during taxi (2 g), rocket engine gimballing during takeoff, landing at the design sink speed of 20 fps (6.08 m/sec), and the flight loads during rocket boost (4 g longitudinal) and maneuvering (3.5 g at Mach 12 and 5 g at low speed).

4.7.3 (U) MATERIALS - The materials used in the Phase III refinement were selected on the basis of the following criteria: the material must (1) be capable of performing its design function, (2) withstand the expected structural loads, temperatures, and acoustic environment for the vehicle's lifetime, (3) exhibit fabricability, and (4) be presently available. Generally, when more then one material will satisfy these requirements the selection criteria becomes minimum "system" cost which usually means minimum weight of the structural element. Material and fabrication costs are the significant factors in material selection when the choice is between alternates of nearly equal weight. However, system cost analysis has shown that the reduced vehicle cost resulting from lower structural weight is almost always more significant than the cost difference between material options.

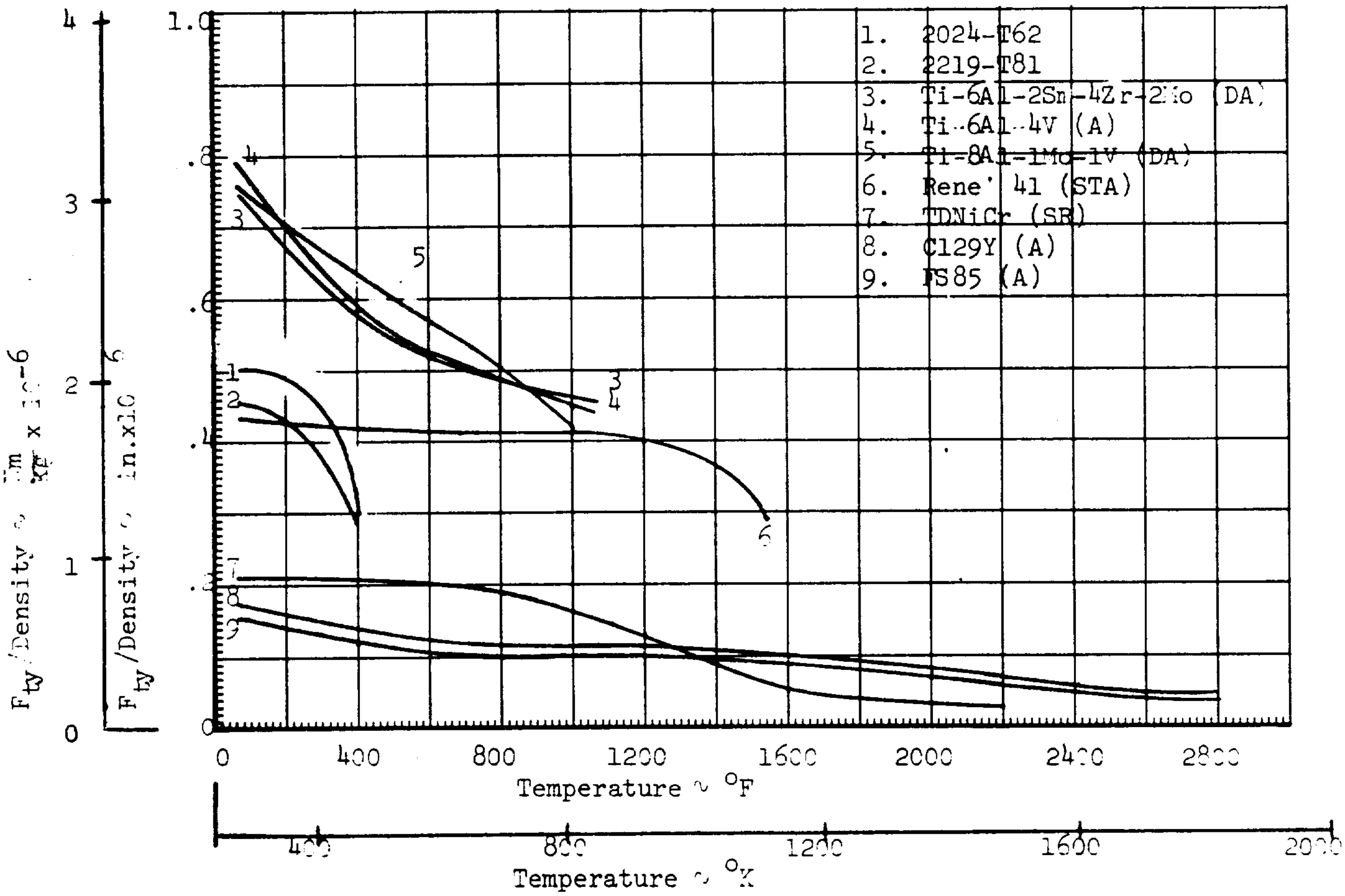
(U) The results of a materials survey are summarily presented in the following paragraphs which discuss the materials in relation to particular applications. Relative tensile and yield strength efficiencies of the competitive materials are shown in Figures 4-84 and 4-85. These are indicative of the properties used in the Phase III analyses. Comparison of these properties gives an indication of the relative weight of equivalent structure. Additional material characteristics considered in the Phase III concept selection are shown in Figure 4-86.

(U) Selection of the materials used for radiation shingles and the high temperature tails and control surfaces are influenced by the material "temperature limits." For this study, "temperature limit" is defined as the maximum temperature at which the material can perform its function in its specific application. These limits have been estimated for material selection purposes as shown in Figure 4-86.

(U) FIGURE 4-84
ULTIMATE TENSILE STRENGTH EFFICIENCY



(U) FIGURE 4-85
TENSION YIELD STRENGTH EFFICIENCY



4.7.3.1 (U) Fuselage Materials - A variety of metal alloys are available for the specific applications in the fuselage structure where the principal selection criteria are strength, efficiency, and fabricability. Since long fatigue life and high temperature capability are not major requirements, a broad choice of aluminum alloys is available with an adequate variety of mill forms, tempers, and protective surface treatments. Aluminum is preferred because of its cost and weight advantage in low temperature applications involving compression or shear loading. Two particularly attractive alloys from the standpoint of yielding efficient lightweight structure are 2024 and 7075. Titanium alloys such as Ti-6Al-4V and Ti-6Al-6V-2Sn exhibit high strength-to-weight ratios up to about 900°F (755°K), have a sophisticated level of fabrication technology and a history of successful hardware experience. Ti-6Al-4V will be used where significant weight savings are achieved, compared to aluminum, and where its high temperature capabilities are needed. Such applications include highly loaded fittings, TPS support details, parts designed by tension loading, and certain critical areas where the existance of heat shorts is possible and would cause permanent damage to a lower temperature material. Very little titanium is used on the Mach 12 vehicle.

(U) FIGURE 4-86
PROPERTIES OF METALS AT ROOM TEMPERATURE

Alloy Designation	Condition (1)	Room Temperature Properties (Guaranteed Minimum)								Temperature Limitation (2)			
		Ultimate Tensile Strength		Yield Tensile Strength		Elastic Modulus		Density		Primary Structure		Secondary Structure	
		KSI	$N/m^2 x 10^{-6}$	KSI	$N/m^2 x 10^{-6}$	$psi x 10^{-6}$	$N/m^2 x 10^{-8}$	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
2014-T6	STA	62	427	54	372	10.7	738	0.101	2794	325	436	375	464
2024-T62	STA	64	442	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-6Al-2Sn-4Zr-2Mo	DA	135	932	125	864	16.5	1138	0.164	4540	1000	810	1100	866
Ti-6Al-2Sn-4Zr-6Mo	STA	160	1102	150	1034	16.5	1138	0.169	4680	750	670	1000	810
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
T D NiCr	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
FS-85 (Columbium)	A	75	517	60	414	20.0	1379	0.383	10060	2700	1756	2800	1810
C129Y	A	75	517	60	414	16.0	1103	0.343	9510	2700	1756	2800	1810
Cb-752	A	75	517	60	414	15.0	1034	0.326	9020	2400	1590	2800	1810
T-222 (Tantalum)	R	110	758	105	725	29.0	2000	0.605	16740	3000	1920	3400	2140
ASTAR 811C	R	105	725	85	586	29.0	2000	0.605	16740	3000	1920	3400	2140

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

(2) Maximum allowable service temperature limits are based primarily upon considerations of metallurgical stability and coating capability and are distinguished by application type; secondary structure is non-strength critical.

4.7.3.2 (U) Cryogenic Tankage Materials - Cryogenic (LH_2) tank material selection is based upon weldability, corrosion resistance, strength, and toughness at both ambient and cryogenic temperatures. The highest strength, weldable, commercially available aluminum alloys are 2014 and 2219. The 2219 alloy has better weldability and greater resistance to stress corrosion cracking than 2014. The 2219 alloy would be used in the T81 temper with alclad on one side, alodined, and treated with an organic coating for maximum corrosion resistance. Titanium alloys Ti-5Al-2.5Sn ELI and Ti-6Al-4V ELI exhibit higher strength-to-density ratios than other candidate alloys for cryogenic service and are weldable and corrosion resistant. However, neither was selected because of the higher cost and the lack of conclusive experimental data concerning the possibility of embrittlement by the environmental hydrogen. Inconel 718 has a strength efficiency comparable to that of 2219 alloy and is weldable and corrosion resistant. However, the structure of Inconel 718 weighs more than that of aluminum because the stability failure mode precludes the use of thin gage design. Therefore, Inconel 718 was not selected. The 300 series stainless steels do not compete sufficiently with 2219 alloy in terms of strength efficiency, although they are weldable and corrosion resistant. Thus, the 2219-T81 alloy was selected on the basis of low cost, adequate mechanical and physical properties, and good fabricability.

4.7.3.3 (U) Heat Shield Materials - Sheet materials were evaluated for heat shield applications to temperatures approaching 3000°F (1920°K). Considerations for selection included yield strength and stiffness efficiencies, temperature limitations imposed by possible surface (oxidation), metallurgical and creep instabilities, and fabrication characteristics. Ti-6Al-2Sn-4Zr-2Mo is the most attractive of the currently available weldable titanium alloys, which include Ti-6Al-4V and Ti-8Al-1Mo-1V, for reasons which include good strength retention, stability, and short time oxidation resistance to about 1100°F (867°K). Rene' 41 has the highest strength-to-density ratio of the superalloys for service to about 1550°F (115°K) to about 2400°F (1590°K). Although its strength efficiency is low at these temperatures, TD NiCr can be used uncoated to about 2000°F (1377°K), whereas other superalloys generally are limited to service below 2000°F (1367°K) even with oxidation resistant coatings. TD NiCr cannot be fusion welded, but resistance welding and brazing have been demonstrated. Columbium alloys such as C129Y and FS85 show the best combination of material characteristics from 2000°F (1367°K) to about 2800°F (1810°K). They also exhibit good creep strength and are considered highly fabricable. C129Y has a higher strength-to-density ratio than FS85, but costs more. Cb-752 also is competitive, but has low elevated temperature creep strength. Tantalum alloys such as T-222 and ASTAR 811C do not compete with columbium alloys in terms of strength efficiency, but do provide a limited service life capability above 2800°F (1810°K) to about 2400°F (2140°K). Refractory metal alloys can be electron beam and resistance spot or seam welded. Both columbium and tantalum alloys require a protective coating and have shown good response to the Sylvania R512 series of fused slurry silicide coatings for oxidation resistance.

4.7.3.4 (U) Control Surface Materials - Primary applications for superalloys and refractory metal alloys are on the vertical tail. In addition to the material candidates discussed for heat shields in Section 4.7.3.3, Hastelloy X and Inconel 625 have application for components which are not as strength critical. Fabrication is much less difficult with these materials than with Rene' 41. The movable surfaces, rudders and tip controls, require columbium alloys because of the high temperature. C129Y and FS85, as on the heat shields, provide the most efficient structure.

4.7.3.5 (U) Non-Metallics - Non-metallic materials contribute to a relatively small amount of the total weight of the aircraft, but have applications of major importance. Some of the physical properties which describe the non-metallics are presented in Figure 4-87. Current optical materials, such as 96% silica glass, which is available in flat panels for service to 2000°F (1367°K), can provide a more than adequate capability for the multi-glaze retractable windshield. The zirconia nose cap technology developed for the ASSET vehicle would provide a limited service life capability to about 4100°F (2530°K). High temperature insulations are required to protect the structure from aerodynamic heating. Fibrous, powder, foam, and multi-layer foil insulations were considered and Flexible Min-K was selected for service to 1800°F (1250°K). Dynaflex insulation raadily conforms to complex shapes and because of its low density is used where weight is a more important consideration than insulation thickness. Flexible Min-K is less conformable. However, it is used most efficiently in applications with a restricted size envelope. Dynaquartz has a lower thermodynamic efficiency, but has service temperature capability to 2700°F (1755°K). Sealed polyurethane foam insulation was selected for internal insulation in the liquid hydrogen tank in conjunction with a laminated metallic foil/polyimide film (Kapton-H) vapor barrier.

(U) FIGURE 4-87
PROPERTIES OF NON-METALLIC MATERIALS

Application	Material	Maximum Temperature Limit °F (°K)	Density lb/ft³ (kg/m³)	Thermal Conductivity $\frac{\text{Btu in}}{\text{hr-ft-°F}}$ $\left(\frac{\text{joule}}{\text{m-sec-°K}}\right)$	Specific Heat Btu/lb°F $\left(\frac{\text{joule}}{\text{kg-°K}}\right)$
Nose Cap	Zirconia	4100 (2535)	270 (4320)	7.5 (1.08) 1	0.158 (656) 1
Windshield	96% silica glass	2000 (1367)	136 (2180)	13.7 (1.97) 1	0.306 (1288) 1
High Temperature Insulations	Flexible Min-K (High Temp.)	1800 (1256)	16 (256)	0.5 (0.072) 1	0.27 (1130) 1
	Flexible Min-K (Light Weight)	1200 (922)	8 (128)	0.46 (0.066) 2	0.26 (1090) 2
	Dynaflex	2700 (1755)	6 (96)	2.31 (0.332) 3	0.27 (1130) 3
Cryogenic Insulation	Polyurethane foam	250 (394)	2 (32)	0.12 (0.017) 4	0.30 (1255) 4

1 Mean Temperature = 1600°F (1143°K), Ambient Pressure
2 Mean Temperature = 1000°F (811°K), Ambient Pressure
3 Mean Temperature = 2000°F (1367°K), Ambient Pressure
4 Mean Temperature = -200°F (144°K), Ambient Pressure

4.7.4 (U) CONCEPT COMPARISON - The following paragraphs discuss a comparison of the alternates available for primary structure, including tankage and the heat shield element of the thermal protection system.

4.7.4.1 (U) Fuselage Structure - Selection of the structure for the fuselage is highly dependent on the choice of the thermal protection system used. If no external TPS were used, the structure would reach temperatures of 2500°F (1640°K) to 2800°F (1810°K) on the lower surface and require a coated columbium alloy. Protecting the structure with a passive insuulation can reduce the structural temperature to any level desired down to about 200-250°F (367-394°K). Trade studies on a variety of aircraft configurations and missions show that when passive insulation is used, the performance is most improved by reducing the structural temperature to less than 800°K (700°K), which allows use of efficient titanium structure. When the dssign mission requirements are considered, that is the relatively short (5-minute) cruise at Mach 12, the least vehicle weight is realized by reducing the structural temperature to 250-300°F (394-422°K). At this temperature both aluminum and titanium are candidates. Application of an active TPS such as water wick will reduce the structural temperatures to less than 200°F (367°K) thereby permitting either titanium or aluminum structure.

(U) In addition to temperature, the aircraft fuselage structure selection is most influenced by the loading level (inplane and normal) and geometric considerations such as curvature, cutouts, and load path direction changes. Also influencing the selection is the tankage/structure/thermal protection system combination. Concepts considered were:

- Ring stiffened skin with longitudinal stringers - aluminum
- Ring stiffened skin with longitudinal stringers - titanium
- Single faced corrugated sandwich - aluminum
- Single faced corrugated sandwich - titanium
- Double faced corrugated sandwich - titanium
- Honeycomb - titanium face, aluminum core.

(U) Parametric analysis based on room temperature properties shows that stiffened skin concept is limited to stress levels of about 75% of that of the sandwich structure concept because of the local instability mode of failure. The aircraft will, however, employ both the stiffened skins and the sandwich concepts. The sandwich structure will be used in the areas where the structure consists of larger panels uninterrupted by cutouts, whereas the stiffened skin concept is more efficiently used in areas where there are cutouts, load path interruptions, and load redistributions. A comparison of the single faced sandwich with other forms of sandwich structure has shown that double faced corrugation, honeycomb, and single faced corrugation sandwich all result in nearly the same weight for this application (Figure 4-88). Single faced corrugation sandwich, however, is slightly lighter than either honeycomb sandwich or double faced corrugation sandwich. Material comparison on the single faced corrugated sandwich showed titanium to be slightly lighter than aluminum.

(U) FIGURE 4-88
FUSELAGE STRUCTURE COMPARISON

Concept Geometry	Concept Description	Basic Panel Weight			Edging Weight	Radial Stiffening Weight	Combined Weight + 10% 2			Min. Gage	Weight Minimum Gage Design
		1000 lb/in. 3 (1750 N/cm)	2000 lb/in. 3 (3500 N/cm)	3000 lb/in. 3 (5250 N/cm)			1000 lb/in. 3 (1750 N/cm)	2000 lb/in. 3 (3500 N/cm)	3000 lb/in. 3 (5250 N/cm)		
		lb/ft² (kg/m²)									
	Double Faced Corrugation E.B. Welded Titanium	0.55 (2.68)	0.90 (4.39)	1.20 (5.85)	0.10 (0.488)	0.142 (0.693)	0.87 (4.24)	1.26 (6.15)	1.59 (7.76)	.008 in. (.0203 cm)	1.042 (5.08)
	Single Faced Corrugation E.B. Welded Titanium	0.48 (2.34)	0.77 (3.76)	1.01 (4.93)	0.11 (0.537)	0.306 (1.49)	0.99 (4.84)	1.31 (6.39)	1.57 (7.66)	.010 in. (.0254 cm)	1.086 (5.3)
	Single Faced Corrugation E.B. Welded Aluminum	0.52 (2.54)	0.82 (4.0)	1.08 (5.27)	0.11 (0.537)	0.41 (2.0)	1.14 (5.56)	1.47 (7.17)	1.76 (8.6)	.016 in. (.0406 cm)	1.34 (6.55)
	Honeycomb Core – Al Face – Ti	1 0.80 (3.90)	1 1.03 (5.03)	1 1.23 (6.00)	0.20 (0.976)	0.126 (0.615)	1.24 (6.05)	1.50 (7.31)	1.71 (8.35)	.008 in. (.0203 cm)	1.43 (7.0)

1 Includes 10% for attachment inefficiency

2 Non-optimum = 10%

3 Compression load.

(U) General and parametric analyses show no specific trend on which to base the selection of one of these options. Detailed analysis on each specific structural element is required to select the best combination of material and construction concept. Such an analysis is beyond the scope of this study. However, history of cost data shows titanium structure is about twice as expensive as aluminum. Since both materials satisfy the functional requirements and are nearly equal in weight, aluminum has been selected as the fuselage structure for costing and weight estimation.

4.7.4.2 (U) Tankage - With the insulated structure concept, the option is provided to use either integral or non-integral tanks. Integral fuel tanks, in which the tank structure is also used to carry primary airframe loads, have a definite advantage when the tank pressures are low, since the airframe, when designed for bending and torsion loads, has an inherent capability for pressure loading. When the tank pressures are high, a cylindrical shape yields the lightest weight. However, this usually doesn't match the aerodynamic shape of the fuselage. In this case, a non-integral tank would have a weight advantage even though two independent structures are required. The inherent pressure capability of the primary structure is dependent on the size and shape of the fuselage and the design load factor requirements. The basic aircraft has a large non-circular fuselage cross section which would, under normal design, yield low pressure capability. For this reason the multi-bubble integral tankage is used, since it offers the low weight advantage of cylindrical pressure vessels and retains the desired fuselage shape. The tank wall and the webs at the bubble intersection are constructed of single faced corrugated aluminum.

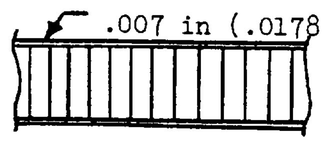
4.7.4.3 (U) Heat Shield - The thermal protection system considered for the basic aircraft, as a result of the Phase II trade study, employs a radiation shield. Of all the elements of the TPS (i.e., heat shield, insulation, coolant, vapor barrier, coolant distribution system), the heat shield presents the greatest structural challenge and highest weight element. There are over 2000 ft^2 (18.58 m^2) of surface area on the aircraft that will be protected by some type of heat shield (Figure 4-76). Thus, a minor increase in weight of the shield would result in a significant change in aircraft weight.

(U) The primary requirements of the shield are to radiate heat energy away from the aircraft, protect the insulation system from airflow, and act as an aerodynamic surface. To satisfy these, the shield must be made of a material with a high temperature capability and high emissivity and have structural capability to transmit airloads at the elevated temperature.

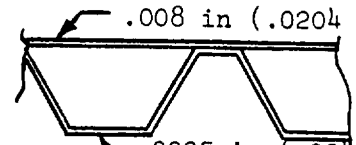
(U) The study made for this application showed that a honeycomb sandwich panel supported on four posts and a single faced corrugated sandwich panel supported on two longitudinal edges and restrained in the center are very competitive, Figure 4-89. For normal loads at the elevated temperature, the honeycomb shield yields the lowest weight and the smallest direct heat path to the substructure. When used on a rapidly accelerating aircraft (e.g., rocket acceleration on a minimum energy path), the thermal gradient through the honeycomb is about twice that of the single faced corrugated sandwich. Figure 4-90 shows the thermal gradient for each concept through the design mission. The high gradient of the honeycomb could result in a thermal deflection as high as twice that of the single faced corrugation. The single faced corrugation, because of its lower gradient and geometric difference, can be designed to have a lower thermal stress level. This has made the single faced sandwich a more practical design than the honeycomb and, therefore, was selected as the basic concept.

(U) FIGURE 4-89
HEAT SHIELD COMPARISON

HONEYCOMB



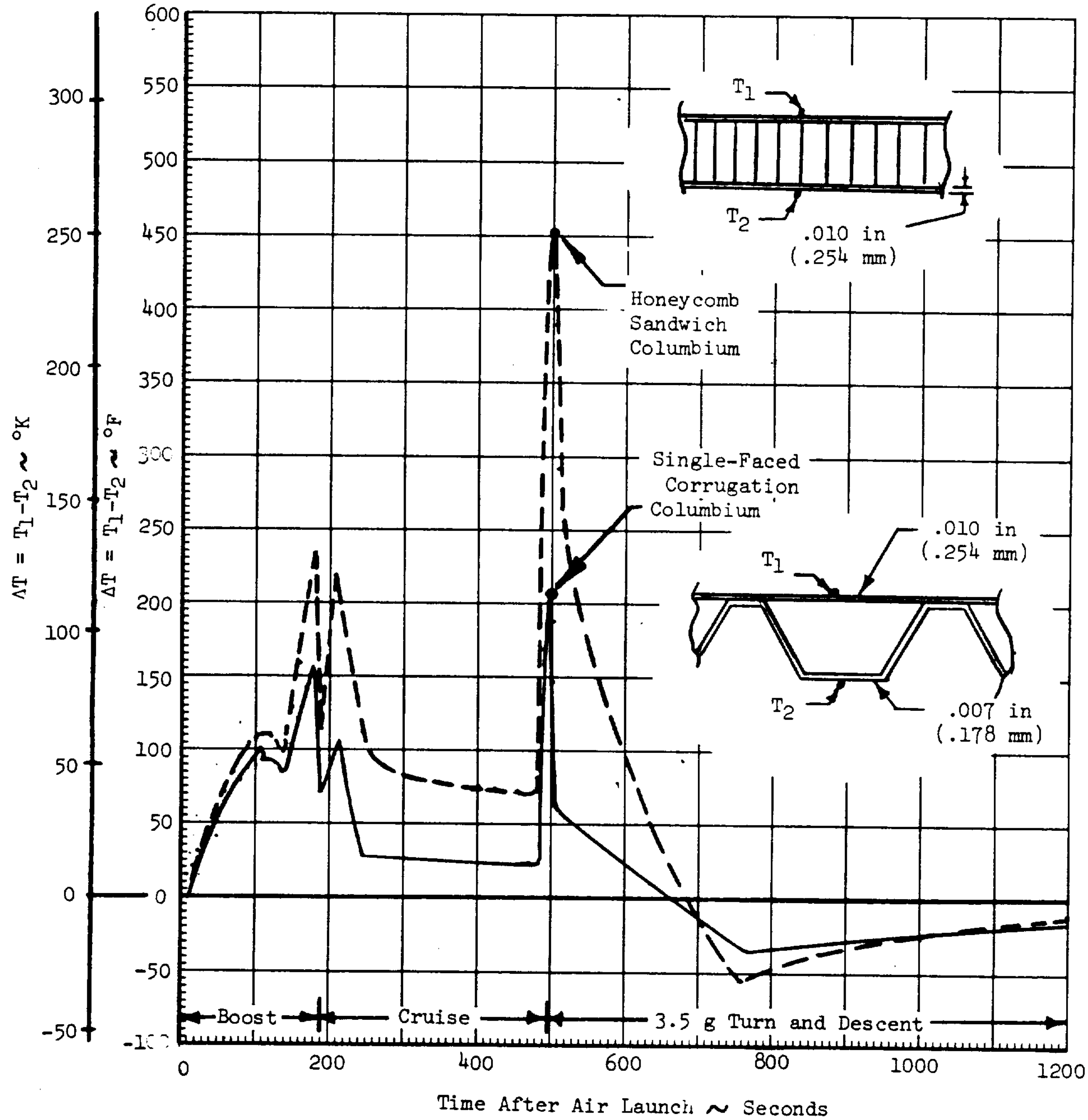
SINGLE FACED CORRUGATION



Material - T.D. NiCr
Density - .306 lb/in^3 (8.48 g/cm^3)
Design Temp. - 2200^oF (1477^oK)
Design Pressure - 3.0psi (2.06 N/cm^2) limit
@ Elevated Temperature

Honeycomb		Single Faced Corrugation
Brazed ---	Construction	--- E.B. Welded
Four Post ---	Support	--- Continuous End
	Weight-lb/ft^2 (g/cm^2)	
.62 (2.95)	Face Skin	1.00 (4.76)
.34 (1.62)	Core	--
.12 (.57)	Edging	.09 (.43)
.09 (.43)	Attachments	.29 (1.38)
.15 (.71)	Braze Matl.	
.13 (.62)	Non-Optimum (10%)	.14 (.67)
1.45 (6.90)	TOTAL	1.52 (7.24)

(U) FIGURE 4-90
HEAT SHIELD TEMPERATURE GRADIENTS
Nominal Mach 12 Research Vehicle Trajectory

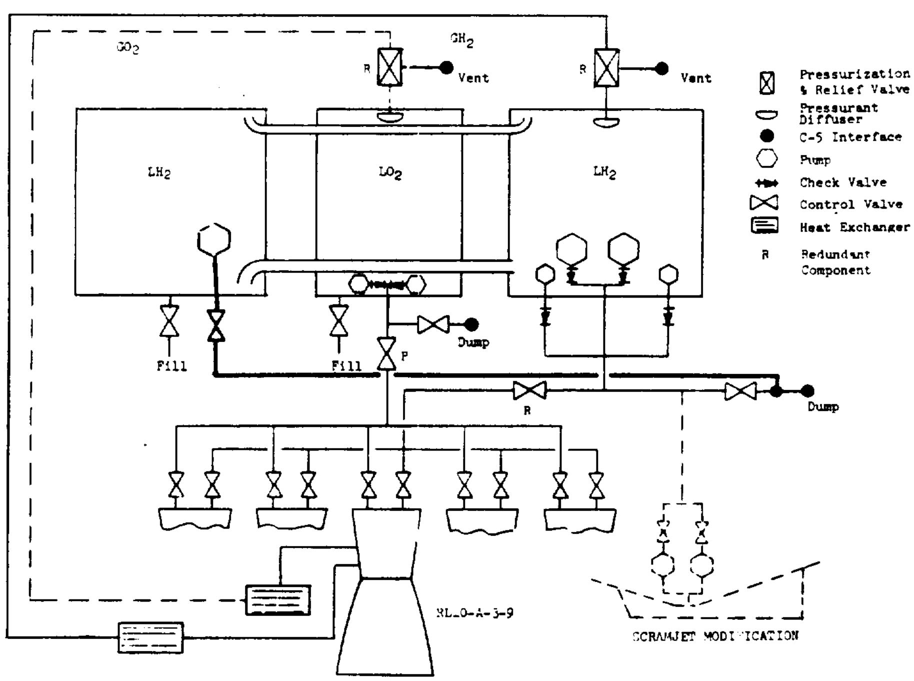


4.8 PROPELLANT SYSTEMS

(U) The propellant system is designed to make maximum use of existing cryogenic system technology, adapted to horizontal flight operation with redundancy in critical elements. The result of this approach is a high confidence system design. Redundancy is provided for critical flow system elements (such as boost pumps, pressure regulators, and main flow distribution valves) to insure flow system integrity in the event of any single component failure. These redundant areas are noted in Figure 4-91 which illustrates major system functional elements and not final system detailed design.

(U) Other safety features include vapor and fire detection/suppression systems in conjunction with provisions for emergency propellant dumping in both the launched and unlaunched condition. Proposed vehicle options do not require any major propellant system changes except for the J2S propulsion option. In order to provide sufficient propellant flow significant increases in component size are necessary, but these enable reductions in manifold complexity.

**(U) FIGURE 4-91
PROPELLANT SYSTEM SCHEMATIC**



(U) A notable safety feature is the lack of need for top-off propellants in the C-5A during the air-launched flight operations. Because subcooled LH_2 is used, the boil off encountered with normal boiling point propellants during both ground and airborne holds is eliminated. Instead the heat leak is absorbed by a bulk fluid temperature rise. Previous air launched research aircraft have experienced difficulty during propellant top-off operations while attached to the launch aircraft.

(U) In Phase II the use of subcooled hydrogen was determined to be both technically feasible and economically desirable. Significant performance gains are possible without an attendant penalty in either airborne or ground systems. Other related advantages of subcooled operation identified in Phase II include reduced tank pressure, decreased ullage, and improved loading and ground hold procedures.

(U) The propellant system is comprised of the following major subsystem/ operational areas:

- Tankage
  - Inter-tank Distribution
  - Pressurization
- Feed System
  - Boost Pumps
  - Engine Feed
  - Dump
- Safety Provisions
- Ground Support and C-5A Interface
- Flight Vehicle Options

Locations of the major system components and launch vehicle interface are shown in Figure 4-1 for the basic flight vehicle.

4.8.1 (U) TANKAGE - The fore and aft internally insulated LH_2 tanks are connected by two vacuum insulated lines to accommodate gravity transfer from the forward tank to the aft tank. The aft LH_2 tank is the primary feed tank. LO_2 is contained in the center tank located over the aircraft's cg to minimize cg travel.

(U) Pressurization systems for both the LH_2 and the LO_2 tankage are shown schematically in Figure 4-91. Liquid hydrogen, tapped off the high pressure side of the rocket engine pumps, is heated to approximately 600°R (333°K) and fed to the tankage at a regulated pressure of 10 psig (6.89 N/cm^2). During periods such as ground hold, rocket start up, and descent, wherein autogenous gaseous hydrogen (GH_2) is not available, gaseous helium (GHe) is used to pressurize the LH_2 tankage to a positive 2 psig (1.38 N/cm^2). The helium is stored at 3000 psia (2065 N/cm^2) and at LH_2 temperatures by immersing the storage bottles in the aft LH_2 tank. Autogenous gaseous oxygen (GO_2) is used to pressurize the LO_2 tank during rocket engine operation. An alternate method for pressurization is the use of a full time GHe system. For the LH_2 tankage, use of GHe results in a small weight increase due to its higher gas density, as compared to GH_2. For LO_2 tank pressurization, a slight reduction in weight can be realized, less than 100 lb (45.4 kg), but the increased cost of pressurant and helium storage requirements will more than offset this weight penalty.

(U) Other tankage system features not shown in Figures 4-1 and 4-91 include baffles to control sloshing, continuous monitoring and point level propellant quantity probes, and other pressure, temperature, and flow control/sensing instrumentation.

4.8.2 (U) FEED SYSTEM - The propellant feed system, provides the necessary propellant flow and inlet pressure schedule to the rocket engines. Design NPSH for the RL10-A-3-3 turbopump are 17.1 psia (11.8 N/cm^2) and 27.9 psia (19.2 N/cm^2) for the LH_2 and LO_2 pumps respectively. These conditions were also assumed to apply to the RL10-A-3-9 engines. This NPSH must be supplied by either increased tank pressure or a boost pump. Incorporation of boost pumps provides the required pressure rise at minimum weight penalties to the aircraft. Locations of these pumps are dictated by the effective tank low points during boost and cruise. For the LH_2 system it is necessary to provide a different set of pumps for each flight phase. During boost the large pumps, located on the aft bulkhead parallel to the thrust vector, provide the necessary flow. During cruise or rocket engine idle, the smaller pumps, positioned at the tankage low points for horizontal flight conditions, are utilized. LH_2 boost pumps are not required in the forward tank. Acceleration forces are sufficient for transfer to the aft tank during boost and normal gravity forces are sufficient during cruise. For emergency dump conditions a single boost pump is provided in the forward tank as gravity forces are not sufficient for high dump rates. For the LO_2 feed system only a single set of boost pumps is required. These are mounted on the lower portion of the aft LO_2 tank bulkhead. Dual boost pumps have been incorporated in both systems to provide the degree of reliability necessary to insure the capability of maintaining air worthiness in the event of a single failure, each being capable of total flow demand. Submerged LH_2 cooled electric motors drive the LH_2 boost pumps while GHe purges the electric motors that drive the LO_2 boost pumps. Turbine driven boost pumps were investigated as alternate approaches but penalties were incurred in terms of increased complexity and weight due to the necessity of routing hot gas lines to the turbine drives and an increased number of tank wall penetrations.

(U) A fuel distribution manifold connects these pumps to the rocket engine high pressure pumps via pre-valves and flexible metal bellows to accommodate gimballing.

(U) In the event of premature flight termination, the remaining propellant can be dumped overboard through lines routed out to the trailing edge of the wing. The boost pumps provide necessary energy with the LO_2 and LH_2 being dumped sequentially; LO_2 first - to reduce weight as rapidly as practical while minimizing hazards. This is discussed in more detail in Section 7.1.

4.8.3 (U) SAFETY PROVISIONS - A major area requiring extensive design and developmental effort is the detection of any LH_2 and LO_2 leaks in conjunction with rapid response fire extinguishment systems. The requirement for investigation in this area was recognized early in the study. This point is also brought out in the Safety Studies, Section 7.1. Potential leak detection methods include thermocouple networks located in the space between the tank wall and external moldline in conjunction with sniffers in equipment bays.

4.8.4 (U) GROUND SUPPORT AND C-5A INTERFACE - Provisions for ground servicing include LH_2, LO_2, and GHe fill ports located on the lower surface of the aircraft. Vent lines, connecting the research vehicle and C-5A launch vehicle permit remote disposal of hazardous vapors during fill operations. During the captive portion of the research vehicle/C-5A flight profile, the research vehicle is dependent on the C-5A for helium pressurant and emergency dumping. Should it prove necessary to abort the research vehicle flight prior to launch, the propellants can be dumped utilizing pressurized (helium) transfer through the lines connecting the research vehicle and C-5A. The propellants would be routed out the extreme aft portion of the C-5A or possibly through the wing tip to minimize fire hazards.

4.8.5 (U) CONFIGURATION OPTIONS - The design approach employed for integration of the propellant system includes provisions for the various research configuration options wherever practicable. This approach has resulted in configuration refinements which minimize changes to the propellant system that affect structural systems.

(U) The HTO and VTO options do not require any airborne propellant systems changes. Ground servicing units can be adapted to utilize existing research vehicle/C-5A interface connections.

(U) CSJ and SJ options will require minor rerouting of LH_2 distribution lines to provide necessary fuel flow to the airbreathing engines. The small boost pumps in the basic flight vehicle are utilized, as shown in Figure 4-91. Complex flow control and regenerative cooling distribution systems will be incorporated into the CSJ and SJ engine modules prior to installation into the basic aircraft.

(U) The TPS option does not affect the propellant distribution system.

(U) A minor rerouting of research vehicle/C-5A interface connections is required for the ARM and STG options.

(U) The TJ option requires the addition of a JP storage tank. This is accomplished by installing a fuel bladder in the area of the basic vehicle's LO_2 tank along with associated JP distribution lines and boost pumps. Pressurization of all tanks, JP and unused LH_2 tankage, is accomplished by engine bleed air. Pressurization of the unused LH_2 tankage is required to negate the effects of local atmospheric pressure changes which may potentially cause structural damage to the aircraft structure.

(U) Design flow rates for the J2S alternate engine option are approximately twice that required for the basic vehicle. Therefore, that particular option will require significant rework of the propellant distribution system. Larger boost pumps, removal of the distribution manifolds, and increased pressurant system flow capacity are the major areas influenced. The resulting system, although with a significantly larger flow capacity, is considerably less complex than the basic vehicle system and should present no operational difficulty.

4.9 SUBSYSTEMS

(U) The vehicle subsystems consisting of avionics and miscellaneous subsystems are described in the following paragraphs.

4.9.1 (U) AVIONICS SYSTEMS - Both the Mach 12 and Mach 6 basic aircraft, described in this report have essentially the same avionic systems. The missions of these two aircraft are similar, as viewed from requirements placed on avionics. The avionic system functions designed to meet these requirements are described in the following sections. The details on the avionic equipment were established in Phase II and were presented in Volume III.

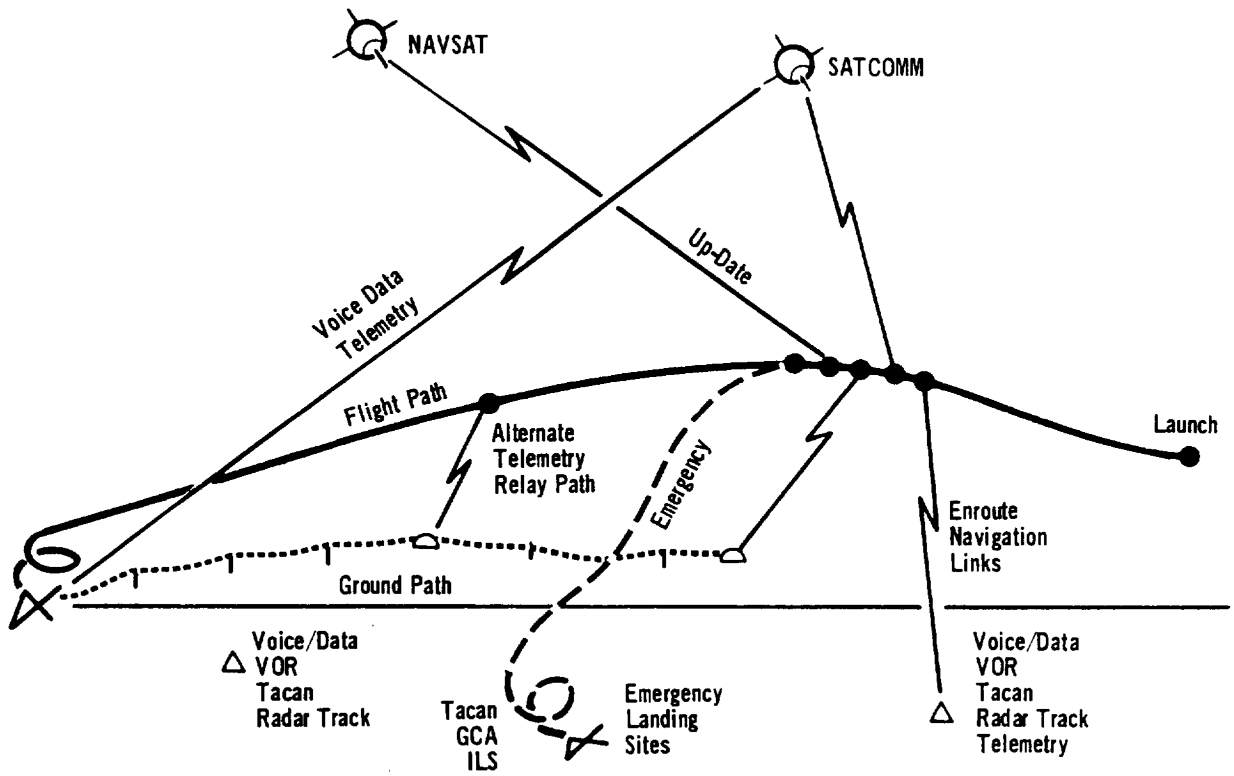
(U) It is believed that current technology is applicable due to the similarity of the missions postulated in the HYFAC study to the X-15 aircraft program. No new avionic functions have to be developed to fly the HYFAC missions. It is planned that, except for the automatic flight control systems (AFCS) and unique interface the aircraft avionics will consist of off-the-shelf equipment current in that time period. Also, the avionics will be compatible with the command and control environment operating at the time of the HYFAC flights. The environment may or may not include satellites for communication/data relay and navigation up-date.

4.9.1.1 (U) Mission Description and Operational Sequence - The analysis of the flight operational requirements are presented in Section 6. A brief summary of the flight planning is presented here to provide a basis of understanding of the avionics system requirements. The basic flight plan for the Mach 12 vehicle calls for the test vehicle to be carried to launch altitude, headed toward the landing site, and released. Boost, acceleration to cruise velocity and climb to equilibrium altitude, are followed by the 5 minute Mach 12 cruise. Having completed the cruise phase, the vehicle begins an unpowered descent to the landing site. The major portions of the flights are flown with the cockpit in the stream line position, which denies the pilot visual contact with the earth.

(U) For the basic 5 minute cruise at Mach 12, the vehicle is launched near Cecil NAS in Florida and flies a distance of approximately 1900 nm (3520 km) to Edwards AFB in California. The navigation, guidance and control of the vehicle are programmed to be completely automatic during the major part of the flight. This flight requires about 30 minutes. During a typical flight, as illustrated in Figure 4-92, the navigation system is in contact with navigation aids along the entire route and may also receive position update information from navigation satellites. The test vehicle has communication opportunities with Edwards through communication satellites, an aircraft relay network or a ground link. For the horizontal takeoff (HTO) and vertical takeoff (VTO) missions, the test vehicle is launched and landed at the same base.

(U) For the missile launch tests (ARM option), the vehicle is launched within the Edwards AFB tracking range, accelerates to Mach 12, the desired altitude, and establishes a heading such that the missile will follow a ballistic path and impact within the Pacific Missile Range. The test vehicle then executes a 3.5 "g" turn and returns to Edwards AFB.

(U) FIGURE 4-92
GENERAL NAVIGATION COMMUNICATION PLAN



(U) A typical trajectory for accomplishing tests involving the STG module includes air-launch at a point up-range of Holloman AFB, acceleration and climb to a maximum altitude of about 250,000 ft (76,200 m) where the STG module is released. This trajectory (flight path) is designed so that the STG module can land at Holloman AFB and the Mach 12 test vehicle has the capability to proceed to Edwards.

4.9.1.2 (U) Functional Requirements Definition - The functional requirements imposed on the avionics subsystems are within the current "state-of-the-art", or have reached a state of devolcpment to indicate that "off-the-shelf" systems will be available during the 1975 to 1980 time period.

4.9.1.2.1 (U) Navigation - The navigation system must include the capability to provide accurate automatic guidance from all launching areas to a point within the range of the localized navigation and landing aids at Edwards AFB or to the alternate emergency landing sites. The pilot will be flying IFR until the airspeed is below Mach 6. The navigation system must provide automatic guidance to these sites with an adequate steering display to allow the pilot to fly the IFR mission as a back-up to the automatic system. The distance flown on some of the proposed missions is longer than those accomplished on the X-15. Therefore,to enhance the safety of the research vehicle in emergency situations, the navigation system will maintain a file of emergency landing sites, continuously compute the flight paths to them and, based on the energy management program, select the optimum site for immediate utilization in an emergency situation. The navigation system must therefore, provide programmed control of the on-board avionic navigation aids (TACAN, NAV, and ILS) to assure continuous use of the enroute ground stations. The accumulated cross range and down range errors must be sufficiently small at TACAN acquisition so that turns greater than 30° are not required to accomplish the landing.

4.9.1.2.2 (U) Communications - Communications are required with the carrier aircraft prior to launch and with the control center at Edwards during the remainder of the flight. The communication requirements include two way voice and data, plus the requirement to telemeter parametric and housekeeping data for use at the control center. The communication route to the control center at Edwards is beyond the horizon but can be established through

a. relay aircraft deployed along the range.

b. a combination of ground microwave links,

c. hard lines, or

d. through a communication satellite.

Constant contact is required with the preselected emergency landing sites to assure communication for verbal landing instruction or automatic landing commands.

4.9.1.2.3 (U) Flight Sensors and Control - The research aircraft requires the standard attitude and heading reference information for manual and automatic control, an air data system adequate to provide measurements of dynamic pressure, angle of attack and side slip angle for flight control at the high altitudes and velocities encountered during the mission. A method for determining altitude, an automatic landing system for blind mode operation, and adequate information for the pilot to accomplish a normal landing are required.

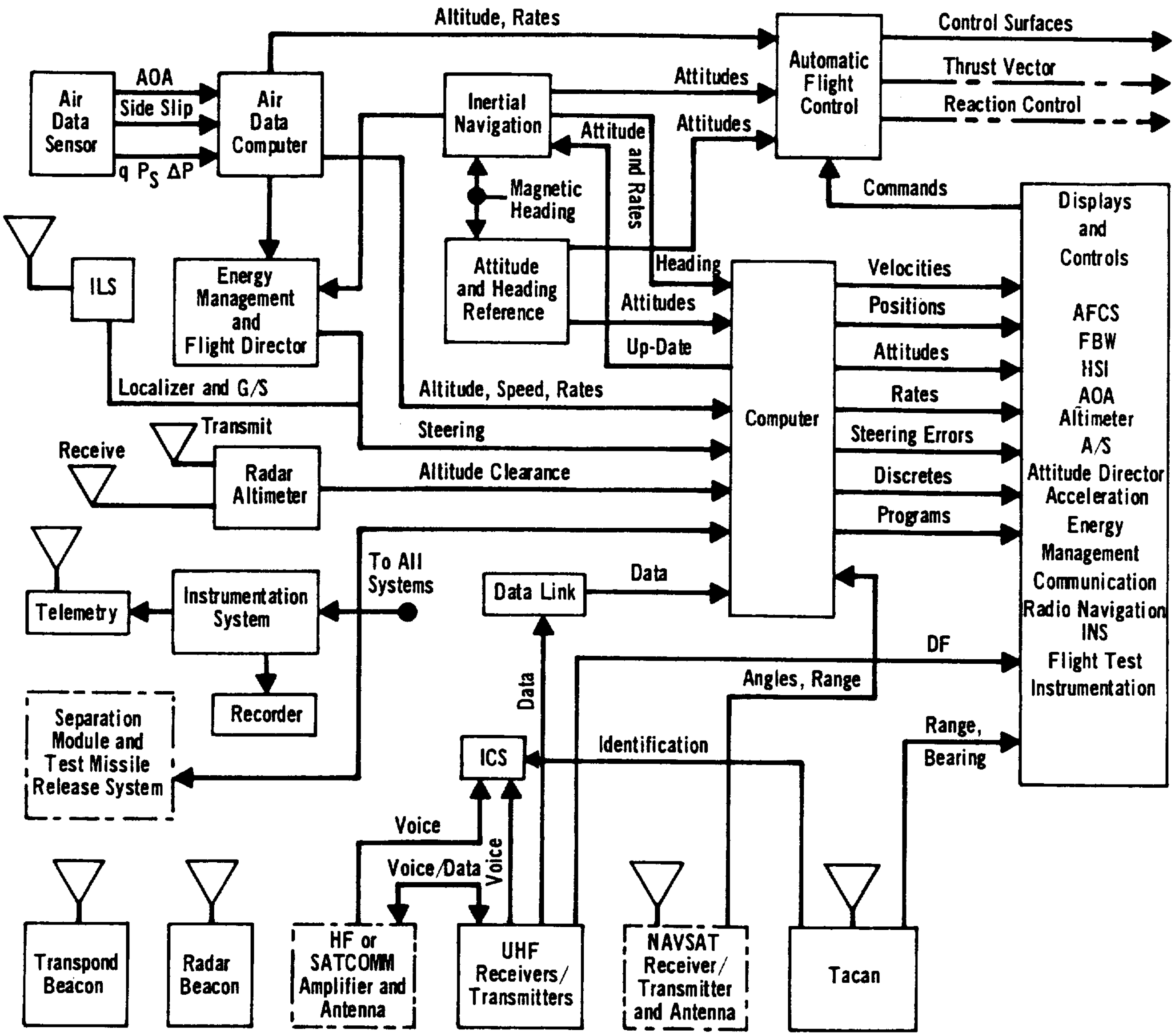
4.9.1.2.4 (U) Controls and Displays - Control panels are required to provide all necessary pilot interface with the aircraft systems, with the separation module and with the test missile systems.

(U) The displays must provide the pilot with the capability to assess his current flight conditions and to determine the action required for emergencies occurring at any phase of the flight. This requirement applies to the blind flying situation at high speed, the head-up or head-down landing situation, the command guidance mode and the manual mode. Display of inertially derived parameters and of the footprint achievable with the energy management system in the glide phase of flight and in emergency situations is required.

4.9.1.2.5 (U) Special Equipment - The research vehicle must have the capability for initialization, monitor and control of the STG module and armament system launches. The STG module, which is designed specifically to investigate the separation characteristics of high speed vehicles, requires the means for transferring alignment data, monitoring the status of the module itself and initiating the separation. For armament system tests only the missile separation characteristics are of interest, thus, a fire control system is not required and the missile test launches require minimum avionics.

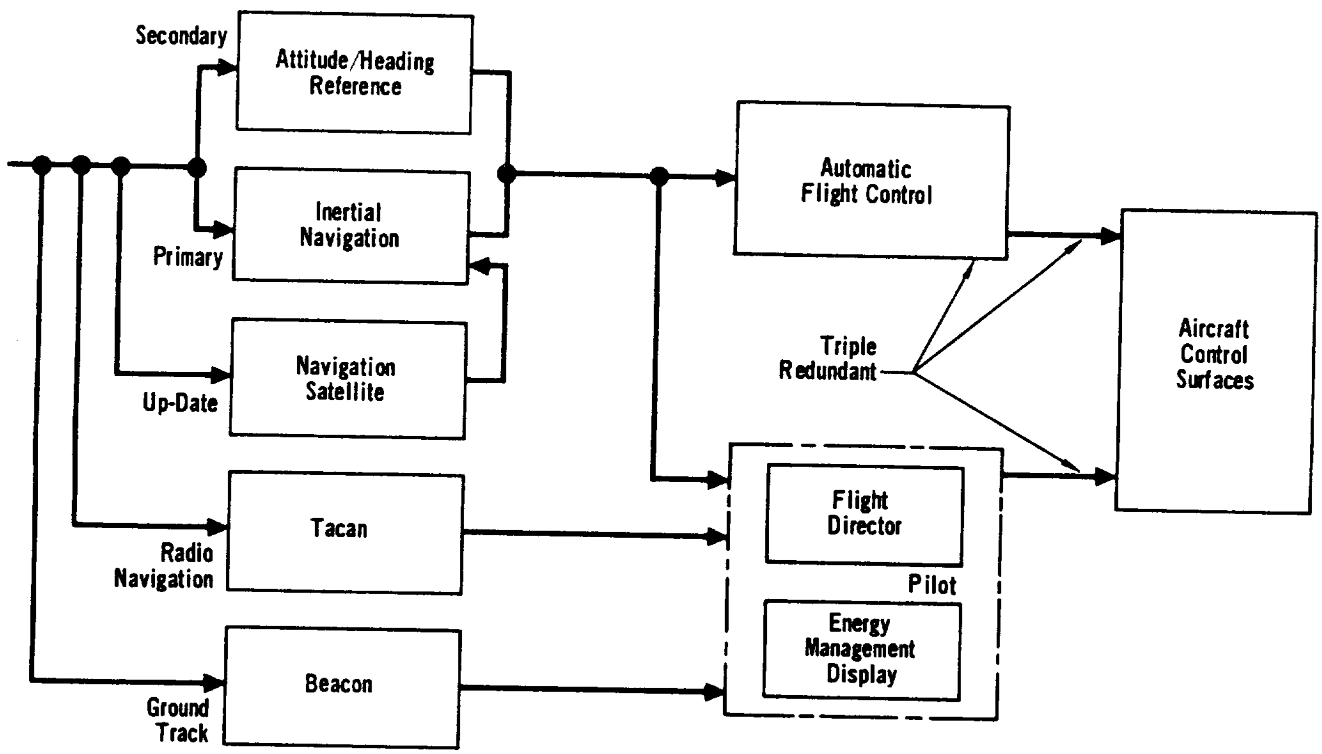
4.9.1.3 (U) Avionics Subsystems Functional Description - This section describes the avionics functional subsystems included in the research vehicle. Figure 4-93 is a functional block diagram which shows the interface between the individual subsystems. These equipment were previously described in more detail in Volume III.

(U) FIGURE 4-93
AVIONIC FUNCTIONAL BLOCK DIAGRAM



4.9.1.3.1 (U) Navigation System - This system is made up of the inertial navigation system (INS), automatic flight control system (AFCS), the energy management and flight director computing system, and TACAN. The INS with NAV SAT update is the primary navigation mode designed and programmed to operate automatically from launch until final approach. Radio navigation aids provide routine traffic control plus a back-up to the automatic INS as shown in Figure 4-94. For flight safety, the navigation system includes the primary INS with a back up provided by the radio navigation aids, the ground track radar and a data link. In addition, the pilot is provided with sufficient displays so that manual override in the IFR situation is an emergency mode. An added feature of the basic navigation system shown in Figure 4-94 is its compatibility with navigation satellites. The equipment included in this system is basically "off-the-shelf" for the 1975 to 1980 time period and with the deployment of a network of navigation satellites, a cost and weight saving may be possible by complete utilization of their capabilities.

(U) FIGURE 4-94
GENERAL NAVIGATION/CONTROL SYSTEM



4.9.1.3.2 (U) Communication System - This system provides the direct link to the launch aircraft, the UHF and HF voice and data systems, the instrumentation/telemetry system and the communication satellite interface system.

(U) The instrumentation/telemetry system is included to handle housekeeping data as well as the parametric measurements indicating the performance of the research vehicle throughout flight.

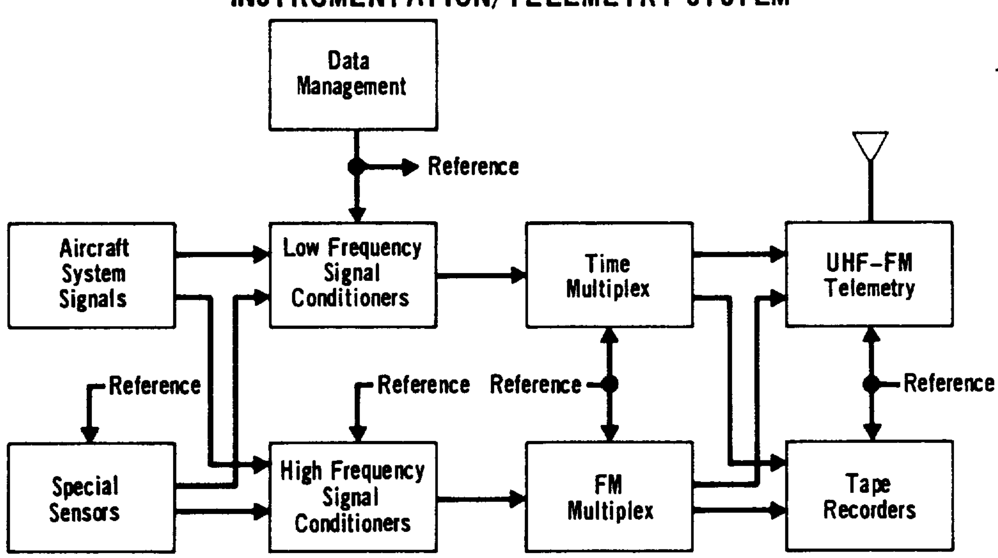
4.9.1.3.3 (U) Flight Sensors and Control - The flight sensor and control system includes the automatic flight control system which accomplishes the actual control of the vehicle on the basis of inputs of the air data computer, the attitude heading reference system, and the energy management and flight director computing system. The overall flight control system provides altitude, speed, attitude, heading and angle of attack outer control loops and is mechanized to include control augmentation inner loops. The flight control concept utilized is a fly-by-wire system with three-axis control augmentation. It is designed to operate if desired on a programmed trajectory with altitude and speed hold loops in operation during cruise. The AFCS has three redundant channels to provide normal operation following a first failure and a fail-operational action after the second failure. This design provides the vehicle control system with redundant sensor data, as shown in Figure 4-93, to provide a reliable and fail-operational system for use in blind and visual flight.

4.9.1.3.4 (U) Controls and Displays - The display system presents, in addition to the more standard flight parameters and aircraft conditions, unique displays for energy management and horizontal situation, to permit landing at automatically selected emergency sites. In addition, flight path angle, angle of attack, and a digital data display of commands, responses, and projected performance are provided.

4.9.1.3.5 (U) Special Equipment - The avionics to support the separation module and missile launch tests includes the interface equipment to monitor the auxiliary vehicles and to transfer alignment and other required initialization data.

4.9.1.4 (U) Test Instrumentation - The test instrumentation will provide a history of the flight and the means for continously monitoring the critical parameters of the vehicle during flight. A generalized block diagram of this system is shown in Figure 4-95. The data management block is the clock timing, logic circuitry for programing and signal switching, and commands for interval recording or telemetry.

(U) FIGURE 4-95
INSTRUMENTATION/TELEMETRY SYSTEM



(U) Data from the aircraft systems is status data such as, aircraft position, velocities, attitude, rates, acceleration, fuel flow, engine parameters, etc. These status signals are generated normally within the avionics or other aircraft subsystems. Signals are tapped off and conditioned for recording or transmission using standard instrumentation techniques.

(U) Data such as aircraft skin temperature, structural stress and vibration, etc.,are also recorded. The sensors for these signals are installed specifically for instrumentation and include thermocouples, strain gauges, pressure transducers, accelerometers, etc. These sensors are currently available off-the-shelf over the range of interest for the proposed vehicles.

(U) Signals to be monitored are time shared on a single channel, i.e., sampled data, or monitored continuously via frequency modulation on a carrier on a subchannel. The expected rate of change of the monitored signals determines the type of modulation used. Selected signals are transmitted via telemetry. Satellite or ground stations relay the telemetry data to the landing site.

(U) The components of the proposed telemetry system are in use today; a suitable antenna and location on the aircraft would be selected during the vehicle design. The recorders proposed are 14 channel magnetic tape recorders, currently available off-the-shelf.

4.9.1.5 (U) Equipment Description - The size, weight, and power of the avionics equipment are summarized in Figure 4-96. The weight estimates of the separation module and test missile release system, and also the instrumentation/telemetry system are included as part of the payload system. Although the equipment to interface with navigation and communications satellites are shown in Figure 4-93, estimates of their size, weight, and power are not given. A trade-off between the quality and weight of the primary INS with the flexibility, accuracy, and weight of the satellite aided system is required to establish the weight of the navigation system which includes satellite compatibility.

(U) FIGURE 4-96 AVIONIC EQUIPMENT SUMMARY

Subsystem	Weight (lbs)	Weight (kg)	Volume (ft^3)	Volume (m^3)	Power (watts)
Inertial Nav.	76	34.4	2.19	.062	675
Attitude and Heading Reference	35	15.9	.72	.0204	550
Energy Management/Flight Director	61	27.6	1.15	.0326	236
Air Data	36	16.3	.70	.0198	200
TACAN	25	11.3	.38	.0108	120
UHF Comm.	21	9.5	.34	.0096	250
HF Comm.	58	26.3	1.08	.0306	800
Radar Altimeter	13	5.9	.28	.0079	85
Data Link	47	21.3	.79	.0224	330
Beacons	9	4.1	.11	.0031	105
Antennas	27	12.2	.54	.0153	-
ILS	8	3.6	.12	.0034	51
Autopilot	54	24.5	1.25	.0354	175
Controls	36	16.3	.69	.0195	48
Displays	60	27.2	.87	.0246	326
Totals	566	257.	11.21	.318	3951

4.9.2 (U) MISCELLANEOUS SUBSYSTEMS - The electrical, hydraulic, environmental control and reaction control systems used in the research aircraft are described in the following paragraphs.

4.9.2.1 (U) Electrical and Hydraulic Systems - Electrical and hydraulic power are supplied by two (redundant) chemically fueled auxiliary power units (APU). Each APU turbine shaft drives a gear box with two gear driven power take-off shafts. One power take-off shaft of the gear box drives a 20 KVA Integrated Drive Generator (IDG) which supplies electrical power for the avionics and test equipment, and the second power take-off shaft of the gear box drives a 60 gpm (3800 cc/sec), 3000 psi (2068 N/cm^2) pump for the hydraulic system. The APU's use LO_2 as the oxidizer and LH_2 as fuel and are carried in separate tanks located adjacent to the power units. The APU's are started just prior to launching and run continuously to landing. Peak power requirements, both electrical and hydraulic, are estimated to be 155 hp (115.6 Kw) with average requirements of 50 hp (37.3 Kw) throughout the flight. Before launch, all power required for ECS, systems checkout, etc., is supplied by the C-5A launcher.

4.9.2.2 (U) Environmental Control System (ECS) - The ECS selected for the test vehicle is a direct loop heat sink system which rejects heat through intermediate fluid transport loops to the APU cryogenic fuel as a heat sink. Cockpit temperature control and avionic and test equipment cooling are provided by the system. The ECS, electrical and hydraulic heat loads are transferred to the APU cryogenic fuel via heat exchangers located in the discharge lines of the fuel tanks. Cockpit pressurization is provided by separate supplies of LO_2 and LN_2, which are mixed in a gaseous form, and are supplied to the cockpit under pressure to provide an acceptable environment during the entire flight.

4.9.2.3 (U) Reaction Control System (RCS) - The staging modification of the basic aircraft requires the use of a reaction control system to provide control in a high altitude, low dynamic pressure environment of the staging mission. Control forces are generated by H_2O_2 fueled rocket thrusters producing from 40 to 110 pounds (178 to 489.5N) of thrust each. The system provides roll, pitch and yaw control. Each control unit consists of a group of thruster nozzles and a pressurization/fuel tank with control valves. Control units are installed in three locations, the aircraft nose and at the outboard ends of the rocket propulsion system compartment. Vehicle control is obtained by firing the thrusters in the proper combination to produce the desired corrective forces. Thrust control is accomplished by an electro-mechanical system added to the basic aircraft control system.

4.10 WEIGHTS

(U) During Phase III the results of the parametric studies conducted in Phase II were incorporated into the basic Mach 12 vehicle design and the configuration was then further refined. When sized to meet the design mission (Section 2.3) the vehicle requires a planform area (S_p) of 813 ft^2 (75.5 m^2) and a TOGW of 79,650 lbm(36,129 kg). The key factors that affected the final refined vehicle weights are discussed below and the associated weight increments are summarized in Figure 4-97.

(U) FIGURE 4-97
WEIGHT SUMMARY – PHASE II TO PHASE III

Item	Operating Weight Empty		Fuel Weight		Takeoff Weight	
	lbm	kg	lbm	kg	lbm	kg
Baseline Phase II Aircraft	24600	11159	53350	24200	77950	35358
Configuration Changes - Total	-1260	-572	+2960	+1343	+1700	+ 771
o Rocket Motor (5) RL10-A-3-9 in Lieu of (1) LR-129	+4000	+1815	+19900	+9027	+23900	+10841
o Payload - Increase Payload to 1500 lbm (681 kg) from 1000 lbm (454 kg)	+ 900	+ 408	+ 1990	+ 903	+ 2890	+ 1311
o Thermal Protection System - Active in Lieu of Passive System	-3600	-1633	- 7050	-3198	-10650	-4831
o Fatness Ratio Increase S_π/S_p to 0.125	-1060	- 481	- 4500	-2041	- 5560	-2522
o Mixture Ratio Increase to 6:1 From 5:1	-1300	- 590	- 3200	-1452	- 4500	-2041
o Miscellaneous Design Changes	+ 600	+ 272	+ 700	+ 318	+ 1370	+ 621
o Resizing-Reduce Vehicle Size & Weight Based on Final Performance Calculations	- 800	- 363	- 4880	-2214	- 5750	-2608
Mach 12 Research Vehicle Weight	23340	10587	56310	25543	79650	36129

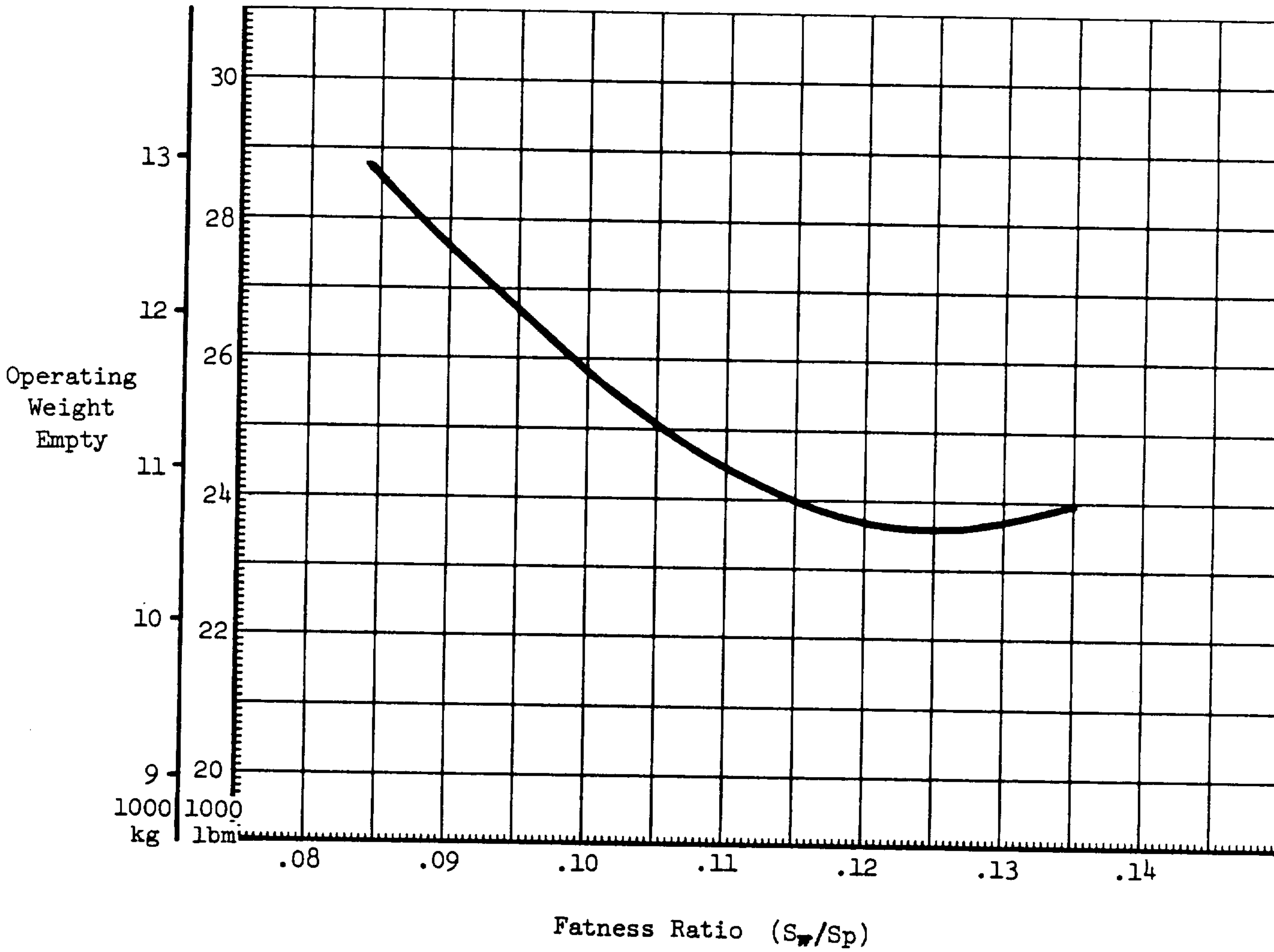
(U) RL10-A-3-9 Rocket Motor - The Phase II vehicles were configured with an LR-129 rubberized engine for preliminary studies, with the thrust requirements tailored to the specific mission requirements. A trade study was made during Phase II to determine the impact on performance and cost of using near term rockets such as the RL10-A-3-9 and J2S and it was determined that a significant cost reduction was available with either of these engines. Of the two available options the RL10 appeared more attractive than the J2S engine. The installation weight for the (5) RL10 engines is only 450 lbm (204 kg) more than for a single LR-129 motor. However, the RL10's lower specific impulse, compared to that of the LR-129, requires adding 19900 lbm (9030 kg) of propellant in order to complete the design mission. The vehicle TOGW is therefore increased a total of 23900 lbm (10840 kg) to meet the design mission (constant performance) with the use of the near term RL10 motors. However, in spite of the larger and more expensive airframe, there is still a significant program cost reduction when using RL10 engines, resulting from the large decrease in development cost.

(U) Payload - Originally the payload weight was selected as 1000 lbm (454 kg). Trade studies showed that the program cost increased only 3% for a 50% increase in data acquisition. Further study of the vehicle showed that there was sufficient volume available to contain 1500 lbm (681 kg) of payload. Therefore, the payload weight was increased to 1500 lbm (681 kg). This 500 lbm (227 kg) increment causes the takeoff weight to grow by 2890 lbm (1312 kg). This takeoff weight change includes the additional mounts, racks, and wiring necessary to support the payload.

(U) Thermal Protection System - The orginial study configuration used a passive thermal protection system as a base for comparison. However, tradeoff studies showed that the weight/cost reduction, as well as the growth potential, of the active system was very attractive. Replacing the passive system with an active one reduced the takeoff weight 10650 lbm (4830 kg). It also decreased the maximum cross-sectional area and planform area by approximately 9.5%. The features of the active system which make it attractive weightwise are its ability to maintain a uniform maximum internal temperature of 100°F (311°K) and minimize heat-shorts. Both of these items are prime candidates for weight growth. For example, a non uniform structural temperature from the windward to the leeward surfaces, which is inherent with a passive system, will produce undesirable thermal stresses. If these stresses are high enough, additional structural weight will be required to react them. Heat shorts would require additional insulation, structural depth and standoff supports to isolate surrounding components from locally high temperatures. In addition to those factors the controls, systems and equipment bays operate in a cool environment. This eliminates any further weight penalties for system operation at high temperatures.

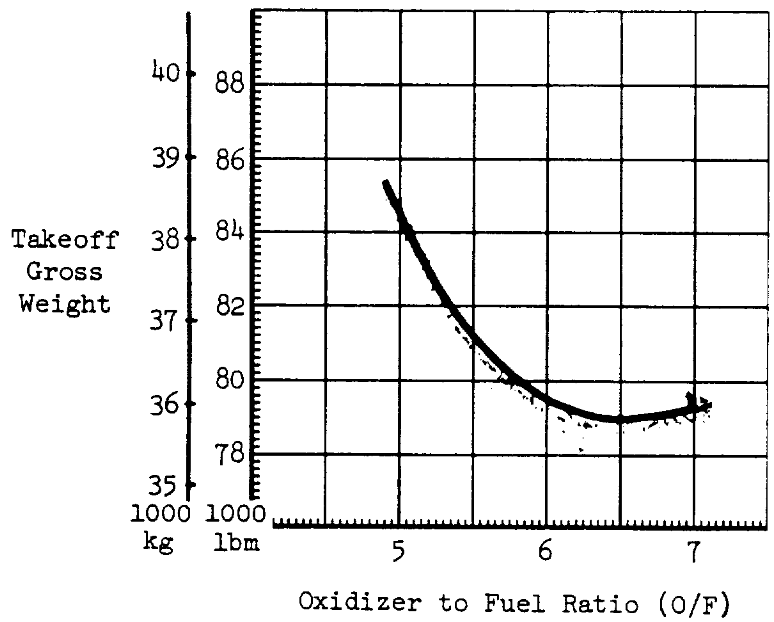
(U) Fatness Ratio - In Phase III the fatness ratio (S_π/S_p) was investigated to determine the best value of maximum cross-sectional area to planform area ratio. Figure 4-98 illustrates the rapid weight reduction from $S_\pi/S_p = 0.085$ to about 0.11 at which point the slope lessens with the minimum weight occuring at $S_\pi/S_p = 0.125$. The physical significance of this study may be interpreted as follows: As the fatness ratio increases, the structural weight decreases because there is less wetted area for a given volume.

(U) FIGURE 4-98
FATNESS RATIO OPTIMIZATION
MACH 12 AIRCRAFT DESIGN



(U) Mixture Ratio Optimization - The RL10 Rocket engine was orginally selected to operate at a mixture ratio of 5:1. Because of the relatively low bulk density of the liquid oxygen/liquid hydrogen propellant system it is desirable to operate the engine at as high a mixture ratio as practical to minimize the airframe size and weight. Increasing the mixture ratio increases the propellant density which then allows the aircraft size and weight to be reduced. However, the specific impulse decreases with increasing mixture ratio. A trade study was made between the structural weight and fuel required to determine the optimum mixture ratio. Figure 4-99 illustrates operating weight and fuel weight variation with mixture ratio. Minimum takeoff weight occurs at 6.5:1.0 but the upper operating limit of the RL10-A-3-9 is 6:1. Therefore, 6:1 was selected as providing minimum weight within the constraints of the system.

(U) FIGURE 4-99
MIXTURE RATIO OPTIMIZATION



(U) Miscellaneous Design Changes - As Phase III progressed, more detailed design definition was available. As a result the OWE increased 600 lbm (272 kg) to account for later design data. Most of the weight increments associated with the design changes were minor, amounting to 100 lbm (45 kg) or less. The largest single weight increase was 195 lbm (89 kg) and occurred in the landing gear group. This resulted from a closer analysis of the landing conditions which revealed loads higher than those previously assumed.

(U) Resizing the Vehicle - Final Performance Calculations - After all of the previously discussed design refinements were incorporated into the Phase II vehicle it was re-performed. Propellant weight was calculated to be 56310 lbm (25500kg) which is 2960 lbm (1342 kg) more than the Phase II baseline vehicle. OWE decreased 1260 lbm (572 kg) in spite of the increased fuel weight due to incorporation of the design refinements which resulted in a smaller sized vehicle. The refined Mach 12 research vehicle takeoff weight increased 1700 lbm (771 kg) to 79650 lbm (36129 kg). Figure 4-97 is a tabular weight summary showing the key changes from the Phase II vehicle to the refined configuration.

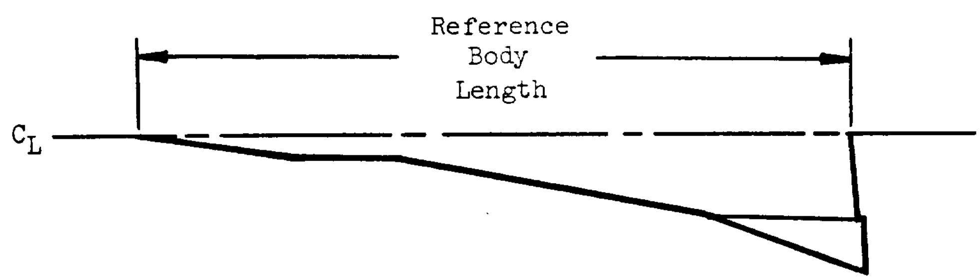
4.10.1 (U) GROUP WEIGHT SUMMARY - Figure 4-100 is the weight summary by functional groups for the basic Mach 12 vehicle. These weights do not include the effects of the research options which are presented in Section 4.10.5. Heat protection weight includes the radiation shingles, insulation, water blanket and nose cap. The active coolant, water, is an expendable item and is therefore carried as useful load. Wing and body weight are entered as a single item since there is no clear dichotomy between the two for this type of configuration. Rotating tip weight includes the moveable surfaces and bearings but not the actuators. All actuators, hydraulic lines from the reservoir to the actuators, valves and controls are coded to the surface controls group. An aircraft development allowance of 422 lbm (192 kg) or about 2% of the weight empty is provided as a contingency.

(U) FIGURE 4-100
GROUP WEIGHT SUMMARY – BASIC MACH 12 VEHICLE

Group	lbm	kg
Wing/Body	(9978)	(4526)
Structure	6297	2856
Heat Protection	3681	1670
Rotating Tips	887	402
Vertical Tails	1868	847
Landing Gear	1225	556
Surface Controls	393	178
Rocket Motors (5) RL10-A-3-9	1605	728
Fuel System	1055	479
Auxiliary Power	430	195
Motor Controls	50	23
Instruments	175	79
Hydraulics	373	169
Electrical	300	136
Electronics	715	324
Furnishings	400	181
Environmental Control	250	113
Contingency	422	191
Weight Empty	20126	9127
Crew and Equipment	240	109
Payload	1500	681
Ullage	241	109
Pressurant	123	56
Auxiliary Power Propellant	150	68
Coolant (Water)	960	435
Operating Weight Empty	23340	10585
Propellant		
Boost - Hydrogen/Oxygen	7431/44615	3371/20237
Cruise - Hydrogen/Oxygen	609/3655	276/1658
Takeoff Gross Weight	79650	36129

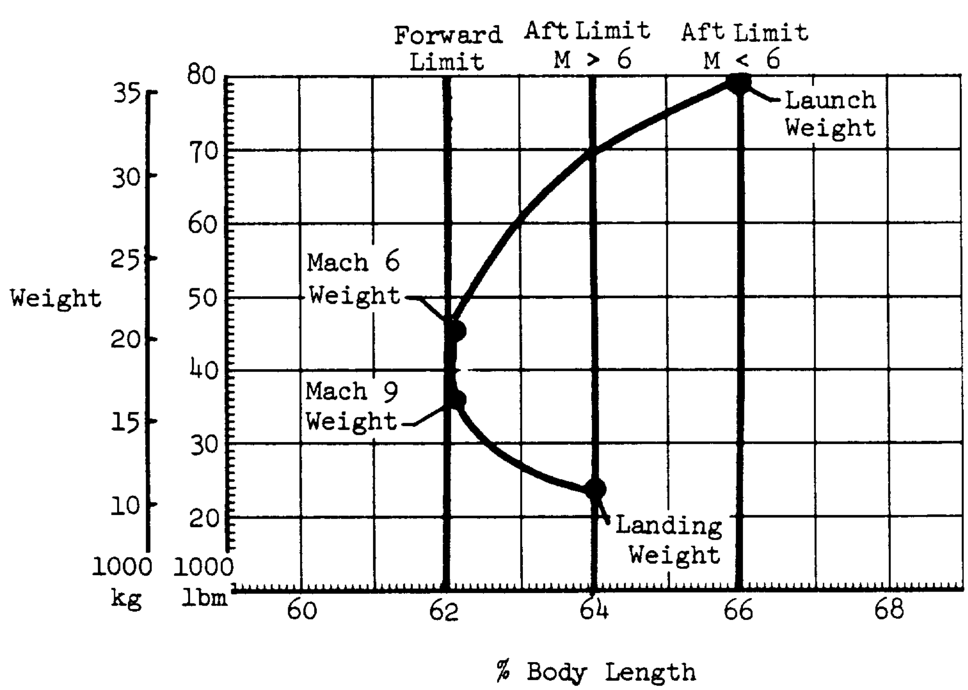
4.10.2 (U) CENTER OF GRAVITY - The aft center of gravity limit for takeoff and landing was set at 66% of the body length. For high speed flight (M > 6) the aft limit moves forward to 64% of the body length. Body length is defined as the distance from the front of the nose cap to the intersection of wing trailing edge and vehicle center line, as shown in Figure 4-101. The 2% forward movement in the aft limit assures a positive static margin during all flight conditions. See Figure 4-37. A forward center of gravity limit of 62% of body length was established to insure adequate control power for takeoff rotation and inflight maneuvering.

(U) FIGURE 4-101
BODY LENGTH DEFINITION



(U) Both the propellant tank location and propellant sequencing were selected to insure that the vehicle c.g. was within the aerodynamic limits at all times. In addition, the c.g. was positioned to provide the best compromise between stability and control throughout the flight profile. For example, Figure 4-102 shows that the c.g. is as far aft as possible at launch to keep control forces at a minimum when the vehicle was heaviest. As fuel is used, the c.g. goes forward to meet the high speed (M > 6) aft limit as well as provide increased stability during cruise at Mach 12. The c.g. then moves aft directly between the limits to provide good stability as well as good handling characteristics at landing.

(U) FIGURE 4-102
CENTER OF GRAVITY TRAVEL
MACH 12 AIRCRAFT



(U) Because liquid oxygen is 86% of the total propellant weight, the LOX tank was situated near the vehicle's center of gravity to minimize its effect on the overall balance. Liquid hydrogen tanks were then placed fore and aft of the liquid oxygen tank. Both the forward liquid hydrogen tank and the liquid oxygen tank were further divided into two separate cells so that the c.g. travel could easily be tailored to the flight profile. A brief description of the fuel sequence is shown below.

Phase of Flight	LOX	LH_2	Total
1. Launch & Accelerate to Mach 6: Empty Aft LOX Cell and Part of the Aft LH_2 Tank	28,760 lbm (13,045 kg)	4,800 lbm (2,177 kg)	33,560 lbm (15,223 kg)
2. Accelerate to Mach 9: Use Part of Fwd LOX cell, Empty Aft Cell of Fwd LH_2 Tank	8,657 lbm (3,927 kg)	1,443 lbm (655 kg)	10,100 lbm (4,581 kg)
3. Accelerate to Mach 12, Cruise 5 Minutes, Descend and Land: Empty Fwd Cell of Fwd LH_2 Tank, Use Remaining LH_2 in Aft Tank, Use Remaining LOX in Fwd Cell	10,840 lbm (4,917 kg)	1,810 lbm (821 kg)	12,650 lbm (5,738 kg)
Total	48,257 lbm (21,889 kg)	8,053 lbm (3,653 kg)	56,310 lbm (25,542 kg)

4.10.3 (U) MOMENTS OF INERTIA - Moments of inertia for takeoff and operating weight conditions were calculated for the basic vehicle and are shown in Figure 4-103. The low value of the product of inertia when compared to conventional vehicles is a result of the low aspect ratio wing/body. This design concept inherently gives low products of inertia and a very shallow principal axis inclination. The roll moment of inertia is also uncommonly low for the weight. Again, this is characteristic of the shape.

(U) FIGURE 4-103
BASIC MACH 12 VEHICLE
Moments of Inertia

Item	Operating Weight Empty		Takeoff Gross Weight	
	Slug-Ft^2	Kg-M^2	Slug-Ft^2	Kg-M^2
I_{XX} Roll	20310	27540	54244	73555
I_{YY} Pitch	373153	505995	459629	623257
I_{ZZ} Yaw	384985	522040	478871	649349
I_{XZ} Product	1042	1413	2518	3414
ϕ Principle Axis	0° 10'		0° 20'	

4.10.4 (U) BASIC MACH 12 VEHICLE MATERIAL SUMMARY - A material breakdown by weight was prepared to aid in costing the airframe. Material types and amounts were determined by temperature and location on the vehicle. The majority of the airframe structural weight is designed in aluminum. Materials were tailored to the temperature distribution to save weight and cost. Figure 4-104 presents the material distribution by functional group. "Other" includes insulation, transparencies, rubber, etc.

4.10.5 (U) WEIGHT INCREMENTS FOR MACH 12 DESIGN OPTIONS - Several design options to the basic vehicle were investigated to enhance its research value. These options were designed as "add-on" kits. However, some modification to the basic vehicle is necessary to easily accommodate installation of these options. For example, strengthening of major bulkheads, hardpoints for attaching research kits, intercostals for distributing loads into the primary structure, etc., should be built into the basic structure. Down time and cost for incorporating the design options are then minimized.

(U) Weight increments for the design options were estimated in two categories, installation provisions and the research package. Installation provision weight accounts for local strengthening of structural members, cutouts, hardpoints, load introduction and redistribution members, fuel lines, valves, wiring, etc. Research package weight is the total weight of the particular design option. It includes the weight of the test article such as a scramjet, as well as the installation provision. Both weight increments are shown so that the total weight change for the

(U) FIGURE 4-104
MACH 12 PHASE III AIRCRAFT MATERIAL DISTRIBUTION – lbm

	Structure					Fixed Equipment			Thermal Protection System								
	Aluminum Alloys	Titanium Alloys	Nickel Alloys	Columbium Alloys	Other	Running Gear	Hydraulics	Equipment	Titanium	Rene '41	T.D. NiCr	Columbium	Tantalum	Nose Cap	Insulation	Expendables	Total Weight
Wing/Body	4341	574	388	474	520				50	588	1517	605	80	150	691		9,978
Rotating Tips			798	89													887
Vertical Tails			1681	187													1,868
Nose Gear			168			60	57										285
Main Gear			837				103										940
Controls							393										393
Engines – (5)RL10A-3-9								1605									1,605
Fuel System								1055									1,055
Auxiliary Power System								430									430
Engine Controls								50									50
Instruments								175									175
Hydraulics							373										373
Electrical								300									300
Electronics								715									715
Furnishings	20							330									400
Environmental Control								250							50		250
Contingency	212	126	84														422
Weight Empty																	20,126
Crew and Equipment								240									240
Payload								1500									1,500
Ullage and Vent Fuel																241	241
Pressurant																123	123
APU Propellant																150	150
Water																960	960
Operating Weight																	23,340
Propellant – LH_2																8,040	8,040
Propellant – LO_2																48,270	48,270
Takeoff Gross Weight	4573	700	3956	750	520	60	926	6650	50	588	1517	605	80	150	741	57,784	79,650

MACH 12 PHASE III AIRCRAFT MATERIAL DISTRIBUTION – Kg
(Presented in International Standard Units)

	Structure					Fixed Equipment			Thermal Protection System								
	Aluminum Alloys	Titanium Alloys	Nickel Alloys	Columbium Alloys	Other	Running Gear	Hydraulics	Equipment	Titanium	Rene '41	T.D. NiCr	Columbium	Tantalum	Nose Cap	Insulation	Expendables	Total Weight
Wing/Body	1969	260	177	215	236				23	267	688	274	36	68	313		4,526
Rotating Tips			362	40													402
Vertical Tails			762	85													847
Nose Gear			76			27	26										129
Main Gear			380				47										427
Controls							178										178
Engines – (5)RL10A-3-9								728									728
Fuel System								479									479
Auxiliary Power System								195									195
Engine Controls								23									23
Instruments								79									79
Hydraulics							169										169
Electrical								136									136
Electronics								324									324
Furnishings	9							150									181
Environmental Control								113							23		113
Contingency	96	57	38														191
Weight Empty																	9,127
Crew and Equipment								109									109
Payload								681									681
Ullage and Vent Fuel																109	109
Pressurant																56	56
APU Propellant																68	68
Water																435	435
Operating Weight																	10,585
Propellant – LH_2																3,648	3,648
Propellant – LO_2																21,896	21,896
Takeoff Gross Weight	2074	317	1794	340	236	27	420	3017	23	267	688	274	36	68	336	26,212	36,129

option, as well as the modification required to the basic vehicle, can be identified. Any changes in fuel or in the fuel to oxidizer relationship is not reflected in the basic package weight. Figure 4-105 is a summary of the design option weights for the Mach 12 vehicle.

(U) FIGURE 4-105
DESIGN OPTION WEIGHT SUMMARY
Mach 12 Vehicle

Design Option	Installation Provisions-Basic Vehicle Only		Research Package - Total Weight		Takeoff Weight	
	lbm	kg	lbm	kg	lbm	kg
Horizontal Takeoff	339	154	831	38	79733	36167
Vertical Takeoff	598	271	342*	155	79992	36284
Convertible Scramjet	300	136	7458	3383	52300	23723
Scramjet	275	125	5969	2708	69200	31389
Thermal Protection System	5	2	273	124	79923	36253
Armament	189	86	4698	2131	84348	38260
Staging	180	82	3558	1614	83208	37743
Subsonic Turbojet	653	296	18965	8603	52305	23726

*Note that installing the research package actually lowers the vehicle weight below that of the vehicle with the installation provisions. This of course is a result of the reduced engine and fairing weights.

(U) HORIZONTAL TAKEOFF (HTO) OPTION - The rocket motor weight was reduced 205 lbm (93 kg) by replacing the existing nozzles, which have an expansion ratio $\varepsilon = 32$, with ones that have an $\varepsilon = 7.4$. This modification was required to reduce back pressure losses for low altitude operation. Because of the shorter nozzles the existing fairing around the rockets was removed and replaced with a shorter one. A horizontal takeoff requires conventional main gear. Replacing the skids with a conventional running gear would require extensive structural rework and relocation of the main gear. This was circumvented by designing a wheeled dolly which is attached by quick release pins to hardpoints provided in the primary structure. It was also necessary to strengthen the bulkheads to react the loads from the dolly. Placing the dolly underneath the vehicle rotated the aircraft to a nose down position. Therefore, the nose gear strut was lengthened 14 inches (35.6 cm) to reduce the nose down static altitude for takeoff. This change added 179 lbm (81 kg) to the aircraft. Figure 4-106 is a summary of the weight changes required for the horizontal takeoff option.

(U) FIGURE 4-106
HORIZONTAL TAKEOFF OPTION WEIGHT SUMMARY

Group	Installation Provisions Only		Total Research Package	
	lbm	kg	lbm	kg
Wing/Body	(339)	(154)	(159)	(72)
o Beef-Ups	(329)	(149)	(329)	(149)
Bulkheads	279	126	279	126
NLG Back-Up	50	23	50	23
o Shorten Fairing	(10)	(5)	(-170)	(-77)
Remove Fairing	-	-	-180	-82
Quick Access	10	5	10	5
Landing Gear-	-	-	(129)	(59)
Lengthen Nose Strut				
Rocket Motors	-	-	(-205)	(-93)
o Delete $\varepsilon = 32$			-1605	-728
o Add $\varepsilon = 7.4$			1400	635
Total	339	154	83	38

(U) VERTICAL TAKEOFF (VTO) OPTION - The weight increments are quite similar to those of the horizontal takeoff option. A main gear dolly is used to transport the vehicle to the gantry where it is rotated into launch position. Beef-ups to attach the dolly and react the taxiing loads amount to 329 lbm (149 kg). The ε = 7.4 rocket nozzles are also used because of the sea level operation. Launch fittings were added to the lower vehicle surface to aid in launching. Figure 4-107 is the vertical takeoff option weight summary.

(U) FIGURE 4-107
VERTICAL TAKEOFF OPTION WEIGHT SUMMARY

Group	Installation Provisions Only		Total Research Package	
	lbm	kg	lbm	kg
Wing/Body	(598)	(271)	(418)	(189)
o Beef-Ups	(364)	(165)	(364)	(165)
Bulkheads	314	142	314	142
NLG Back-Up	50	23	50	23
o Shorten Fairing	(10)	(5)	(-170)	(-77)
Remove Fairing	-	-	-180	-82
Quick Access	10	5	10	5
o Launch Fittings	(224)	(101)	(224)	(101)
Landing Gear-	-	-	(129)	(59)
Lengthen Nose Strut				
Rocket Motors	-	-	(-205)	(-93)
o Delete ε = 32			-1605	-728
o Add ε = 7.4			1400	635
Total	598	271	342	155

(U) CONVERTIBLE SCRAMJET (CSJ) OPTION - 3539 lbm (1604 kg) was added for installing the convertible scramjet option. Figure 4-108 summarizes the weight increments. Primary structural modifications were limited to local attachments and load transfer fittings. Weight was removed for the thermal protection system in the area occupied by the scramjet. 100 lbm (45 kg) was added for fuel lines to the convertible scramjet and regeneratively cooled ramp panels. The propellant mixtures were varied according to the climb profiles and modes of operation shown in Figure 4-65 to achieve the performance shown in Figure 4-66. The propellant weights shown in Figure 4-108 were determined for climb profile (A) of Figure 4.65. Propellant weights for climb profiles (B) and (C) with the resulting net research package weights are shown below.

Item	Climb Profile (B) (Figure 4-65)		Climb Profile (C) (Figure 4-65)	
	lbm	kg	lbm	kg
INSTALLATION SUBTOTAL	7458	3383	7458	3383
Propellant Utilization	(-16419)	(-7448)	(-34818)	(-15793)
o Rocket - LH_2	-2316	-1051	-5594	-2537
- LOX	-13922	-6315	-33593	-15238
o CSJ - LH_2	3474	1576	8024	3640
- LOX	-3655	-1658	-3655	-1658
TOTAL RESEARCH PACKAGE	-8961	-4064	-27360	-12411

(U) SCRAMJET (SJ) OPTION - The weight increment for this option is similar to the convertible scramjet option. The scramjet and movable ramps are shorter and therefore lighter. Figure 4-109 summarizes the weight increments to install the scramjet option. Propellant was adjusted as required by the climb profiles and modes of operation shown in Figure 4-65 to achieve the performance shown in Figure 4-66. Propellant weights shown in Figure 4-109 were determined for climb profile (A) of Figure 4-65. Propellant weights for climb profile (B) and the resulting scramjet research packages are shown below.

Item	Climb Profile (B) (Figure 4-65)	
	lbm	kg
INSTALLATION SUBTOTAL	5969	2708
Propellant Utilization	(-16419)	(-7448)
o Rocket - LH_2	-2316	-1051
- LOX	-13922	-6315
o CSJ - LH_2	3474	1576
- LOX	-3655	-1658
TOTAL RESEARCH PACKAGE	-10450	-4740

(U) FIGURE 4-108
CONVERTIBLE SCRAMJET OPTION WEIGHT SUMMARY

Group	Installation Provisions Only		Total Research Package	
	lbm	kg	lbm	kg
Wing/Body	(200)	(91)	(35)	(16)
o Remove TPS	-	-	-425	-193
o Add Hard Points	200	91	200	91
o Add Hinges/Fittings for Actuation	-	-	260	118
Fuel System	(100)	(45)	(100)	(45)
CSJ Installation	-	-	(7323)	(3321)
o CSJ Primary Structure			3781	1715
o Accelerator Door			135	61
o Forward Ramp			1348	611
o Aft Ramp			982	445
o Injectors and Cooling			187	85
o Actuation			517	235
o Heat Protection-Exit			373	169
INSTALLATION SUBTOTAL			7458	3383
Propellant Utilization			(-3919)	(-1778)
o Boost (Rocket) - LH_2			-83	-38
- LOX			-522	-237
o Cruise (CSJ) - LH_2			341	155
- LOX			-3655	-1658
TOTAL	300	136	3539	1604

NOTE: Propellant weights are given for Climb Profile (A) of Figure 4-65.

(U) FIGURE 4-109
SCRAMJET OPTION WEIGHT SUMMARY

Group	Installation Provisions Only		Total Research Package	
	lbm	kg	lbm	kg
Wing/Body	(175)	(79)	(56)	(26)
o Remove TPS	-	-	-413	-188
o Add Hard Points	175	79	175	79
o Add Hinges/Fittings for Actuation	-	-	298	135
Fuel Systems	(100)	(45)	(100)	(45)
Scramjet Installation	-	-	(5813)	(2637)
o Scramjet	-	-	3040	1379
o Accelerator Door			135	61
o Forward Ramp			986	447
o Aft Ramp			744	337
o Injectors and Cooling			180	82
o Actuation			376	171
o Heat Protection-Exit			352	160
INSTALLATION SUBTOTAL			5969	2708
Propellant Utilization			(-3919)	(-1778)
o Boost (Rocket) - LH_2	-	-	-83	-38
- LOX	-	-	-522	-237
o Cruise (SJ) - LH_2	-	-	341	155
- LOX	-	-	-3655	-1658
TOTAL	275	124	2050	930

NOTE: Propellant weights are given for Climb Profile (A) of Figure 4-65.

(U) THERMAL PROTECTION SYSTEM (TPS) OPTION - 73 lbm (33 kg) was estimated for provisions for all equipment bay and mounting brackets in the TPS option. The internal structure was already recessed so that a fairing was not necessary when the option was installed. An addition 200 lbm (91 kg) was estimated for the thermal protection system being investigated. Figure 4-110 summarizes the TPS option weight.

(U) FIGURE 4-110
TPS OPTION WEIGHT WEIGHT SUMMARY

Group	Installation Provisions Only		Total Research Package	
	lbm	kg	lbm	kg
Wing/Body	(5)	(2)	(73)	(33)
o Equipment Bay	-	-	68	31
o Equipment Bay Provisions	5	2	5	2
TPS Equipment	-	-	(200)	(91)
Total	5	2	273	124

(U) ARMAMENT (ARM) OPTION - Two 20 inch (51 cm) diameter missile launch tubes were mounted on top of the vehicle. The weight added covers the missile tubes, support structure and firing system. The missile tubes are constructed of steel because of the pressure and temperature generated by the ejectors. The tubes are insulated from the support structure to prevent structural damage. External heat protection weight is based on the basic vehicle active TPS concept. Missile tube door actuation uses a hydraulic system. Each missile weight is assumed to be 1650 lbm (748 kg). Figure 4-111 is the weight summary for the armament option.

(U) STAGING (STG) OPTION - The addition of a second stage and a reaction control system to the basic vehicle compose the primary weight elements for this option. Weight for the second stage is 2360 lbm (1070 kg) and is based on conventional aluminum sheet stringer construction which yields the lowest cost design. Internal instrumentation weight is predicated upon a minimum of data pickups and transmitters. Additional instrumentation can be added at the expense of weight. Considerable latitude in second stage weight is possible as is indicated in Figure 4-69. 1198 lbm (543 kg) is added to the basic vehicle to accommodate the second stage. This weight includes 950 lbm (43 kg) for the hydrogen/peroxide reaction control system (RCS). The RCS was added because the second stage launch altitude is anticipated to be beyond the realm of aerodynamic control effectiveness.

(U) A fairing is added between the stages in order to prevent severe local heating problems. Figure 4-112 summarizes the staging option weights.

(U) FIGURE 4-111
ARMAMENT OPTION WEIGHT SUMMARY

Group	Installation Provisions Only		Total Research Package	
	lbm	kg	lbm	kg
Wing/Body	(123)	(56)	(1026)	(465)
o Add Missile Tubes	-	-	276	125
o Support Structure	123	56	123	56
o Missile Bay	-	-	402	182
o Heat Protection	-	-	225	102
Electronics- Fire Control System	(13)	(6)	(63)	(29)
Armament	(53)	(24)	(309)	(140)
o Ejectors/Sabots	15	7	108	49
o Supports	-	-	39	18
o Wiring/Controls	38	17	80	36
o Door Actuation	-	-	30	14
o Insulation	-	-	52	23
Missiles (2)	-	-	(3300)	(1497)
Total	189	86	4698	2131

(U) FIGURE 4-112
STAGING OPTION WEIGHT SUMMARY

Group	Installation Provisions Only		Total Research Package	
	lbm	kg	lbm	kg
Wing/Body	(50)	(23)	(223)	(101)
o Attach Fittings	50	23	50	23
o Fairing	-	-	173	78
Electronics	-	-	(25)	(11)
Reaction Controls	(130)	(59)	(950)	(431)
o Thruster	-	-	170	77
o Installation	130	59	130	59
o Propellant	-	-	650	295
Second Stage	-	-	(2360)	(1070)
o Structure			2201	998
o Instrumentation			85	38
o Ejector			24	11
o Miscellaneous			50	23
Total	180	82	3558	1613

(U) SUBSONIC TURBOJET (TJ) OPTION - Subsonic performance and handling characteristics can be determined by incorporating two P&WA JT4A-11 turbojets into the basic vehicle. The rocket motors were not removed. The landing gear is modified for conventional takeoff and landing operations. This requires 205 lbm (93 kg) additional structural weight to introduce and distribute the loads from the relocated main gear. Ground clearance angles dictate a longer main gear which in turn requires the nose strut be lengthened to reduce the nose down static attitude during takeoff. The largest increment is the 10200 lbm (4627 kg) engine weight. Most of the added weight, as shown in Figure 4-113, is aft of the basic vehicle center of gravity. 7200 lbm (3266 kg) of ballast is required in the payload bay to keep the vehicle within the center of gravity limits.

(U) FIGURE 4-113
SUBSONIC TURBOJET OPTION WEIGHT SUMMARY

Group	Installation Provisions Only		Total Research Package	
	lbm	kg	lbm	kg
Wing/Body	(517)	(235)	(1411)	(640)
o Main Gear Backup	205	93	205	93
o Nose Gear Backup	50	23	50	23
o Nacelle Fittings	262	119	262	119
o Heat Protection	-	-	-82	-37
o Nacelle	-	-	976	442
Landing Gear	-	-	(1089)	(494)
o Main	-	-	960	435
o Nose	-	-	129	59
Propulsion	(136)	(62)	(10565)	(4792)
o (2) PWA JT4A-11 Engines	-	-	10200	4627
o Thrust Deflectors	-	-	164	74
o Fuel System	136	62	201	91
Miscellaneous	-	-	(200)	(91)
Remove Payload	-	-	(-1500)	(-680)
Add Ballast	-	-	(7200)	(3266)
Total	653	297	18965	8603

(U) COMBINATION OF COMPATIBLE MODIFICATIONS - The Phase III Mach 12 aircraft can incorporate all modification packages in one airframe except for the subsonic research kit. The subsonic research package contains too many major modifications to the aircraft to be compatible with the others.

(U) Installation provisions for these packages are shown in Figure 4-114. Any redundancy between combined modifications has been eliminated through analysis of critical loadings and common structural requirements. Incorporation of all these modification provisions in the vehicle would only result in an increase in the vehicle O.W.E. of 1272 lbm (577 kg).

(U) FIGURE 4-114
COMBINED MODIFICATIONS

ITEM	HORIZONTAL TAKEOFF		VERTICAL TAKEOFF		CONVERTIBLE SCRAMJET		SCRAMJET		THERMAL PROTECTION		ARMAMENT		STAGING		TOTAL	
	lbm	kg	lbm	kg	lbm	kg	lbm	kg	lbm	kg	lbm	kg	lbm	kg	lbm	kg
FUSELAGE PROV.			598	271	200	91					123	56	50	23	971	441
FUEL SYSTEM					100	45									100	45
EQUIPMENT BAY PROV.	INCLUDED IN VERTICAL TAKEOFF						INCLUDED IN CONVERTIBLE SCRAMJET		5	2					5	2
ARMAMENT PROV.											53	24			53	24
ELECTRONICS INSTL.											13	6			13	6
REACTION CONTROLS INSTL.													130	59	130	59
TOTAL			598	271	300	136			5	2	189	86	180	82	1272	577

4.11 COSTS

(U) Total system costs were derived for the basic Mach 12 vehicle and several configuration options designed to accomplish research in areas of specific interest. The acquisition cost for the basic Mach 12 vehicle is 263 $\overline{M}$ dollars, while the 5 year operating cost for 200 flights is 88 $\overline{M}$ dollars. When the operating cost is amortized equally over the 200 flight program, the operating cost is .44 $\overline{M}$ dollars. The acquisition costs represent approximately 75% of the total system cost which is 351 $\overline{M}$ dollars. The cost format used in presenting the total system cost is essentially the same as the one employec in Phase II.

(U) In order to provide maximum visibility to the effects of each of the various research vehicle configuration options two distinct incremental cost effects were determined. The effect of initially building into the basic vehicle the structural provisions necessary to accommodate the eventual installation of an optional research package was developed individually and designated "installation provisions". The costs involved with the actual development and installation of the particular option in the basic vehicle was also determined and designated "research package".

(U) The development of the basic vehicle costs with an explanation of the important factors which influence the costs is presented in the following sections and is followed by development of the costs associated with the research options.

4.11.1 (U) BASIC VEHICLE COSTS - The basic Mach 12 vehicle detail cost breakdown is presented in Figure 4-115 along with a bar chart illustrating the distribution of costs. The results reflect the refinements discussed in Section 3.6.

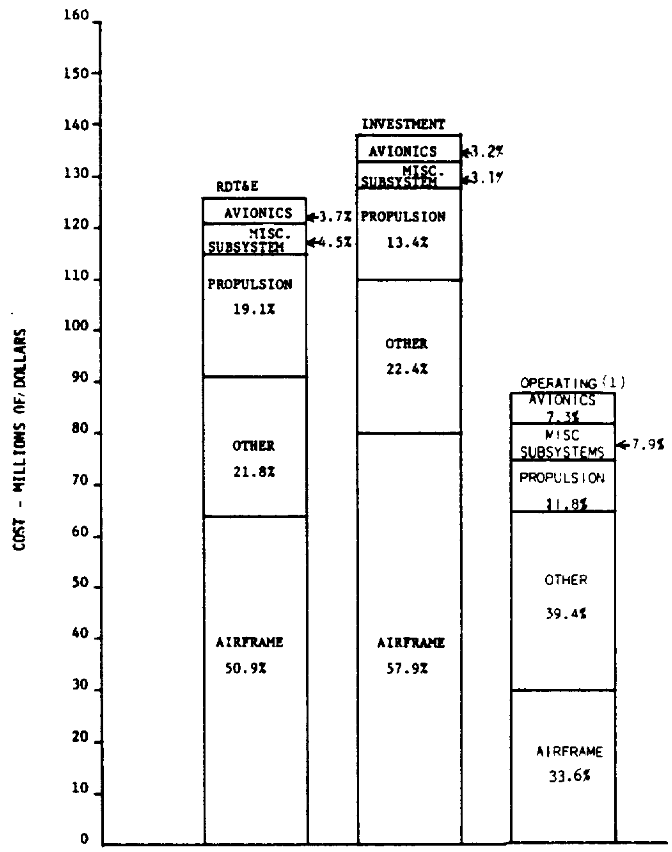
(U) The maintenance/repair cost development was significantly changed from the method employed in Phase II as previously discussed in Section 3.6. In Phase III, the vehicle maintenance/repair cost (Item III-3 of Figure 4-115) per flight for the Mach 12 vehicle is 0.84% of the flight research vehicles investment cost, Item II-1 of Figure 4-115 . It can be readily seen from the bar chart distribution in Figure 4-115 that the airframe costs are predominant in all three major cost categories. The "other" cost segment shown in the three major cost categories includes the following cost elements:

- Tooling - RDT&E
- Support equipment design and system integration - RDT&E
- Support costs - Investment
- Launch platform - Investment
- Range user cost - Operating
- Escort aircraft and logistics - Operating
- Propellant and pressurant cost - Operating
- General purpose maintenance support - Operating

(U) FIGURE 4-115
TOTAL PROGRAM COST SUMMARY AND COMPARISON FOR THE BASIC MACH 12 VEHICLE
(200 Flight Program – 5 Year Duration)
Total Program Cost = 351 $\bar{M}$ Dollars

Cost Categories and Elements	Basic Vehicle Costs (1)
I. RDT&E Costs	
1. Airframe Design and Development	
A. Airframe Design	51.5
B. Miscellaneous Subsystem Design & Development	5.7
C. Development Tests (Including Wind Tunnel)	2.9
D. Test Hardware	5.5
E. Pre-Delivery Flight Test	4.1
Sub-Total	69.6
2. Tooling	9.8
3. Avionics Development	4.7
4. Propulsion Development	24.0
5. Support Equipment Design and System Integration	17.5
6. Ground Test Facilities	-
Total	125.6
II. Investment Costs	
1. Flight Vehicles	
A. Airframe	26.6
B. Miscellaneous Subsystem	1.4
C. Propulsion	2.3
D. Avionics	1.4
Unit Cost (1) Vehicle	31.7
Unit Cost (3) Vehicles	95.2
2. Support Costs	
A. AGE	14.3
B. Training Equipment	0.8
C. Initial Stocks (Engines & AGE Spares)	12.9
D. Initial Training	1.0
E. Initial Transportation	1.4
Sub-Total	30.4
3. Launch Platform Cost	11.9
4. Modification Inst. Cost	-
Total	137.5
III. Operating Cost (200 Flights - 5 Years)	
1. Range, User Cost	5.6
2. Escort Aircraft and Logistics	2.2
3. Vehicle Refurbishment Cost	
A. Airframe	
a. Material	19.7
b. Labor	9.9
B. Miscellaneous Subsystems	
a. Material	6.2
b. Labor	.7
C. Propulsion Systems	
a. Material	8.4
b. Labor	2.0
D. Avionics	
a. Material	5.4
b. Labor	.9
Total Maintenance/Repair Cost	53.2
4. Propellant and Pressurant Cost	1.0
5. AGE Maintenance Cost	2.5
6. General Purpose Maintenance Support	1.0
7. Transportation Cost	0.6
8. Pilot Pay and Support Personnel Pay	19.2
9. Launch Platform Operation Cost	2.5
Total Operating Cost	87.8
Grand Total	350.9

(1) Costs In Millions of 1970 Dollars



(1) Operating Cost Summary

1)	Fixed Cost	24.0
2)	Variable Cost	63.8
	Total	87.8

- Transportation cost - Operating
- Pilot pay and support personnel pay - Operating
- Launch platform operating cost - Operating

4.11.2 (U) PROGRAM COST INFLUENCES - A number of the factors which influence the total program cost are discussed in the following sections.

4.11.2.1 (U) Airframe Cost Influence - Of all the flight research vehicle systems, the airframe exerts the greatest influence on the RDT&E, Investment and Operating costs. Approximately 49% of the total program cost is attributed to the airframe, and it is the dominant factor in the total program cost.

(U) The basic elements that drive the airframe acquisition cost are: (1) Material type such as Aluminum, Titanium, Columbium, etc.; (2) amount of each type of material used (% DCPR weight); (3) airframe size which is represented by DCPR weight; and (4) construction method. These are primarily influenced by the operating temperatures which is a function of the aircraft speed, a factor also contributing to the operating cost.

(U) The influences of the cost elements are reflected in the cost through use of a "weighted complexity factor" which includes the effect of construction type and fabrication difficulty associated with the material, for example, hot forming of titanium, brazing T.D. NiCr, or coating Columbium. The amount of material and the type used is reflected in the DCPR weight and the associated "material type" distribution is given as a % DCPR. These factors are employed in cost estimating for all three previously defined cost categories (RTD&E, Investment, and Operating).

(U) The DCPR weight composition for the Mach 12 basic vehicle is shown in Figure 4-116, while the development of the weighted production complexity factor is shown in Figure 4-117. The weighted complexity factor is used to compare the cost of fabricating the basic Mach 12 airframe relative to the cost of an equal size and weight all-aluminum airframe. For the total basic aircraft airframe which is composed of 49% advanced materials, the relative fabrication cost is 3.9 to 1. The estimation procedure for establishing the cost of the aluminum airframe is reliably established through historical experience and the material cost and fabrication complexity factors are established on MCAIR manufacturing experience as well as data from Battelle Memorial Institute and the Air Force Machinability Center.

(U) Although the weighted complexity factor is used for the airframe as a whole, the elemental fabrication complexity factors can be used to establish relative costs between individual structural elements. In applying the estimation factors to structural elements, we have found, contrary to popular belief, that materials with the higher temperature capability are not always the most expensive. The following paragraphs show an example of this in a comparison between materials for a highly loaded structure, in different environments and between material costs for a lightly loaded structure also, in different temperature environments. Figure 4-118 gives an example of the relative cost of a section of fuselage

(U) FIGURE 4-116
DCPR WEIGHT COMPOSITION BASIC MACH 12 VEHICLE

Material Type	Basic Structure lb	Basic Structure kg	Equipment lb	Equipment kg	Total lb	Total kg
Aluminum	4,573	2,074			4,573	2,074
Titanium	750	340			750	340
Steel	1,477	670			1,477	670
Rene' 41	588	267			588	267
T.D. NiCr	3,996	1,812			3,996	1,812
Columbium	1,355	614			1,355	614
Tantalum	80	36			80	36
Nose Cone	150	68			150	68
Insulation	741	336			741	336
Other	520	236			520	236
Equipment	-		3,207	1,454	3,207	1,454
Total	14,230	6,453	3,207	1,454	17,437	7,907

(U) FIGURE 4-117
AIRFRAME PRODUCTION COMPLEXITY – BASIC MACH 12 VEHICLE

Material	DCPR Weight lb	DCPR Weight kg	% of DCPR Wt.	Fabrication Complexity Factor	Weighted Complexity
Advanced					
1) Tantalum	80	36	.56	4.0	2.24
2) Columbium	1,355	614	9.52	4.0	38.08
3) T.D. NiCr	3,996	1,812	28.08	9.0	252.72
4) Rene' 41	588	267	4.13	7.5	30.98
5) Titanium	750	340	5.27	2.0	0.54
6) Nose Cone	150	68	1.05	4.0	4.20
Sub-total	6,919	3,137	48.61		338.76
Conventional					
1) Insulation	741	336	5.22	1.0	5.22
2) Aluminum	4,573	2,074	32.14	1.0	32.14
3) Other	520	236	3.65	1.0	3.65
4) Steel	1,477	670	10.38	1.0	10.38
Sub-total	7,311	3,316	51.39		51.39
Total	14,230	6,453	100.00		390.15
Total/100					3.9

(U) FIGURE 4-118
RELATIVE FABRICATION COSTS – AIRFRAME
(Material and Labor)

Material Type	Fabricated Material Cost (1) $/lb	Fabricated Material Cost (1) $/kg	Relative Material Cost	Fabrication Labor Cost (2) $/lb	Fabrication Labor Cost (2) $/kg	Relative Labor Cost (Complexity Factor)	Fabrication Cost Labor & Material $/lb	Fabrication Cost Labor & Material $/kg	Relative Weight (3)	Relative Structure Cost (4)
Aluminum	37	82	1.00	240	529	1.0	277	611	1.0	1.00
Titanium	50	110	1.36	420	926	2.0	470	1036	1.23	2.09
Rene' 41	50	110	1.36	1275	2811	7.5	1325	2921	2.08	9.94
T.D. NiCr	75	165	2.04	1035	2282	9.0	1110	2447	6.80	27.20
Columbium	500	1102	13.59	448	988	4.0	948	2090	7.13	24.38

(1) Based on avg. material price with allowances for scrap and offal
(2) Labor rate is based on the following DCPR weights
- a) Aluminum - 30,000 lb (13,605 kg)
- b) Titanium - 36,900 lb (16,734 kg)
- c) Rene' 41 - 62,400 lb (28,298 kg)
- d) T.D. NiCr- 204,000 lb (92,514 kg)
- e) Columbium - 213,900 lb (97,004 kg)

(3) This factor represents relative structural weight for a constant size fuselage designed by loads occurring at the following temperatures:
- Aluminum - 250°F
- Titanium - 600°F
- Rene' 41 - 1500°F
- T.D.NiCr- 2000°F
- Columbium - 2500°F

(4) Relative to an aluminum fuselage structure (fixed size, variable weight)

structure designed to operate in different temperature environments and compared on the basis of equal strength. A different result occurs when comparing structural elements that are influenced by minimum material gage limitations. The radiation shingles, which comprise 14% of the DCPR weight, are examples of minimum gage (i.e., thickness is governed by manufacturing limits rather than strength) structure. The complexity factors can also be used in making comparisons in this type structure as shown in Figure 4-119. It is interesting to note that when highly-loaded structure is compared the total cost increases as the environmental temperature increases, which is due to lower strength allowables requiring greater thickness materials. However, when strength is not as great a factor the environmental temperature has less influence on the material gage and the material with low fabrication complexity cost results in the lowest overall cost. In the example shown, Columbium shingles cost less than superalloy shingles because of fabrication complexity; however, superalloy structures (highly loaded) cost less than Columbium because of their high strengths at elevated temperature. These results are different than expected and valid within the accuracy of the factors used. In the final analysis, the maintenance, repair and replacement costs (directly related to structural life) must be considered and may change the ordering illustrated in Figures 4-118 and 4-119.

(U) FIGURE 4-119
RELATIVE FABRICATION COSTS – HEAT SHIELD (1)

Material Type	Fabrication Material Cost (2) $/lb	Fabrication Material Cost (2) $/kg	Relative Material Cost	Relative Labor Cost (Complexity Factor)	Fabrication Labor Cost (3) $/lb	Fabrication Labor Cost (3) $/kg	Fabrication Cost (Labor & Material Cost) $/lb	Fabrication Cost (Labor & Material Cost) $/kg	Weight Per Unit Area lb/ft^2	Weight Per Unit Area kg/m^2	Fabrication Cost Per Unit Area Labor & Material $\$/ft^2$	Fabrication Cost Per Unit Area Labor & Material $\$/m^2$	Relative Heat Shield Cost Labor & Material (4)
Aluminum	37	81	1.00	1.0	240	529	277	610	.70	3.42	194	2086	1.00
Titanium	50	110	1.36	2.0	480	1058	530	1168	.68	3.32	360	3878	1.86
Rene' 41	50	110	1.36	7.5	1800	3969	1850	4079	1.41	6.88	2609	28064	13.47
TD NiCr	75	165	2.04	9.0	2160	4763	2235	4928	1.45	7.08	3241	34890	16.74
Columbium	500	1102	13.59	4.0	960	2117	1460	3219	1.57	7.67	2292	24690	11.84
Tantalum	600	1323	16.31	4.0	960	2117	1560	3440	4.50	21.97	7020	75576	36.25

(1) Based on 2 ft x 2 ft (61 cm x 61 cm) heat shield.
(2) Based on the average price of material.
Includes: pan stock, castings, forgings, extrusions, purchased parts, sheet and bar stock and tubing
The cost also includes allowances for scrap and offal
(3) Labor rate is based on an aluminum airframe of 30,000 lb (13,605 kg) DCPR weight.
(4) Relative to aluminum.

4.11.2.2 (U) Engine Cost Influence - It was found in Phase I, that engine development costs for some vehicle designs required large dollar expenditures. In some cases, more than half of the program development costs were attributed to engine development. This was the case for turboramjets, scramjets, ramjets and convertible scramjets.

(U) The propulsion development cost for the Mach 12 basic vehicle is relatively low because a modified RL10 rocket propulsion system is used which requires only a 24 $\overline{M}$ dollar development expenditure. In Phase II, it was found that appreciable savings could be realized by using RL10 engines instead of rubberized versions of the LR-129 rocket engine. The engine development costs for the Phase II B233 configurations employing rubberized versions of the LR-129 rocket engine ranged from 77 $\overline{M}$ to 195 $\overline{M}$ dollars while the engine development cost of the Phase II B233 configurations employing the RL10 engine was 24 $\overline{M}$ dollars.

(U) The investment cost of the 5 RL10-A-3-9 engines employed in the Mach 12 vehicle is 2.3 $\overline{M}$ dollars or .46 $\overline{M}$ dollars per engine. The total acquisition (RDT&E plus investment) cost attributed to the propulsion system for the Mach 12 basic vehicle is 42.4 $\overline{M}$ dollars which is approximately 16% of the total acquisition cost. Engine investment costs are based on the procurement of 40 engines: 15 installed in 3 aircraft and 25 spare engines.

4.11.2.3 (U) Miscellaneous Subsystems and Avionics System Cost Influence - The acquisition costs associated with the avionics and miscellaneous subsystems are not significant due to the fact that only off-the-shelf hardware was employed. They amounted to 19 $\overline{M}$ dollars, which is approximately 7% of the total acquisition cost.

4.11.2.4 (U) Operating Cost Influences - The Mach 12 basic vehicle maintenance/repair cost is 53 $\overline{M}$ dollars and is the largest of all the operating cost elements, being 60% of the total operating cost. A breakdown of the Mach 12 basic vehicle maintenance/repair cost is shown in Figure 4-120.

The operating cost for the Mach 12 basic vehicle (87.9 $\overline{M}$ dollars) ammortized over the 200 flight research program is .440 $\overline{M}$ dollars per flight.

(U) FIGURE 4-120
MACH 12 VEHICLE MAINTENANCE/REPAIR COST SUMMARY
FOR 200 FLIGHTS

MAJOR SYSTEM	COST - MILLIONS OF DOLLARS		
	Material	Labor	Total
1. Airframe	19.7	9.9	29.6
2. Propulsion Sys.	8.4	2.0	10.4
3. Avionics	5.4	.9	6.3
4. Misc. Subsystems	6.2	.7	6.9
Total	39.7	13.5	53.2
% of Total	75.0	25.0	100.0

4.11.3 (U) RESEARCH VEHICLE OPTIONS - Seven major research vehicle option programs are priced for the Mach 12 basic vehicle and are as follows:

- SJ - Scramjet
- CSJ - Convertible Scramjet
- ARM - Armament
- STG - Staging
- HTO/VTO - Horizontal/Vertical Takeoff
- TPS - Thermal Protection System
- TJ - Subsonic Turbojet

The modification costs for each of the seven major options were separated into the installation provision's cost and the cost of the research package and are presented in Figure 4-121. It was found that installing the vehicle option provisions on the basic vehicle at the contractor's manufacturing facility resulted in lower costs than installing them at the test center. In Figure 4-121, the installation provisions costs for options 1 thru 6 are based on installation at the contractor's facility. When each of the six installation provisions are installed separately at the test facility, an increase of .6 $\overline{M}$ dollars results.

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(U) FIGURE 4-121
DETAILED TOTAL SYSTEM COST BREAKDOWN FOR MACH 12 VEHICLE AND ASSOCIATED OPTIONS

Cost Categories and Elements	Cost – Millions of Dollars											
	Basic Vehicle Cost	Options										
		Option No. 1 SJ		Option No. 2 CSJ		Option No. 3 STG		Option No. 4 Arm.		Option No. 5 HTO/VTO		
		Installation Provisions	Research Packages	Installation Provisions	Research Packages	Installation Provisions	Research Packages	Installation Provisions	Research Packages	Install Provisions HTO	Install Provisions VTO	Research Packages
I. RDT&E Costs (No. of Flights – 200)												
1. Airframe Design and Development												
A. Airframe Design	51.502	0.623	6.127	0.722	7.705	0.179	4.874	0.585	4.185	1.676	2.600	1.847
B. Miscellaneous Subsystem Design & Development	5.673	0.129	0.890	0.129	1.082	0.164	0.588	0.049	0.159	–	–	–
C. Development Tests (Including Wind Tunnel)	2.866	0.031	0.738	0.033	0.901	0.007	0.852	0.020	0.016	0.016	0.032	0.251
D. Test Hardware	5.494	0.055	3.875	0.062	4.928	0.022	0.426	0.036	0.065	0.024	0.089	–
E. Pre-Delivery Flight Test	4.088	–	–	–	–	–	–	–	–	–	–	–
Sub-Total	69.623	0.847	11.630	0.946	14.616	0.372	5.740	0.690	4.425	1.716	2.721	2.098
2. Tooling	9.776	0.110	4.950	0.191	6.188	0.061	0.913	0.118	0.256	0.048	0.190	0.064
3. Avionics Development	4.669	–	–	–	–	–	–	–	–	–	–	–
4. Propulsion Development	24.000	–	150.000	–	175.000	–	–	–	–	–	–	2.400
5. Support Equipment Design & System Integration	17.531	0.094	2.585	0.094	3.196	0.047	1.034	0.047	2.068	0.141	0.235	0.188
6. Ground Test Facilities	–	–	15.000	–	30.000	–	–	–	–	–	–	–
Total	125.599	1.051	184.165	1.231	229.000	0.480	8.687	0.855	6.749	1.905	3.146	4.750
II. Investment Costs												
1. Flight Vehicles												
A. Airframe	26.560	0.248	19.194	0.282	24.422	0.077	0.205	0.172	0.292	0.118	0.447	0.267
B. Miscellaneous Subsystems	1.432	0.037	0.251	0.037	0.304	0.047	0.150	0.014	0.044	–	–	–
C. Propulsion	2.300	–	2.800	–	3.500	–	–	–	–	–	–	2.300
D. Avionics	1.446	–	–	–	–	–	0.043	–	–	–	–	–
Unit Cost (1) Vehicle	31.738	0.285	22.245	0.319	28.226	0.124	0.398	0.186	0.336	0.118	0.447	2.567
Unit Cost (3) Vehicles	95.214	–	–	–	–	–	3.980 ①	–	–	–	–	–
2. Support Costs	–	–	–	–	–	–	–	–	–	–	–	–
A. AGE	14.282	–	1.668	–	2.117	–	0.060	–	0.050	–	–	0.385
B. Training Equipment	0.750	–	–	–	–	–	–	–	–	–	–	–
C. Initial Stocks (Engines & AGE Spares)	12.928	–	0.447	–	0.561	–	0.006	–	0.005	–	–	0.039
D. Initial Training	1.000	–	–	–	–	–	–	–	–	–	–	–
E. Initial Transportation	1.375	–	0.042	–	0.054	–	0.002	–	0.001	–	–	0.008
Sub-Total	30.335	–	2.157	–	2.732	–	0.068	–	0.056	–	–	0.432
3. Launch Platform Cost	11.873	–	0.895	–	1.119	–	0.354	–	0.705	–	–	2.000
4. Modification Installation Cost	–	–	0.083	–	0.083	–	0.042	–	0.055	–	–	0.037
Total	137.422	0.285	25.380	0.319	32.160	0.124	4.444	0.186	1.152	0.118	0.447	5.036
III. Operating Cost												
1. Range User Cost	5.566	–	–	–	–	–	–	–	–	–	–	–
2. Escort Aircraft & Logistics	2.220	–	–	–	–	–	–	–	–	–	–	–
3. Vehicle Maintenance/Repair Cost												
A. Airframe												
a. Material	19.701	–	–	–	–	–	–	–	–	–	–	–
b. Labor	9.855	–	–	–	–	–	–	–	–	–	–	–
B. Miscellaneous Subsystems												
a. Material	6.232	–	–	–	–	–	–	–	–	–	–	–
b. Labor	0.675	–	–	–	–	–	–	–	–	–	–	–
C. Propulsion Systems												
a. Material	8.375	–	0.152	–	0.286	–	–	–	–	–	–	–
b. Labor	2.025	–	–	–	–	–	–	–	–	–	–	–
D. Avionics												
a. Material	5.449	–	–	–	–	–	–	–	–	–	–	–
b. Labor	0.945	–	–	–	–	–	–	–	–	–	–	–
Total Maintenance/Repair Cost	53.257	–	0.152	–	0.286	–	–	–	–	–	–	–
4. Propellant and Pressurant Cost	1.014	–	–	–	–	–	–	–	–	–	–	–
5. AGE Maintenance Cost	2.480	–	0.250	–	0.318	–	0.009	–	0.008	–	–	0.058
6. General Purpose Maintenance Support	1.000	–	–	–	–	–	–	–	–	–	–	–
7. Transportation Cost	0.609	–	–	–	–	–	–	–	–	–	–	–
8. Pilot Pay & Support Per Pay	19.200	–	–	–	–	–	–	–	–	–	–	–
9. Launch Platform Operation Cost	2.540	–	–	–	–	–	–	–	–	–	–	0.100
Total Operating Cost	87.886	–	0.402	–	0.604	–	0.009	–	0.008	–	–	0.158
Grand Total	350.907	1.336	209.947	1.550	261.764	0.604	13.140	1.041	7.909	2.023	3.593	9.944

① Ten Units

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	...tion No. 6 TPS		Option No. 7 TJ		Installation Provisions for Options 1 through 6
...ation ...sions	Research Packages	Installation Provisions	Research Packages	
	1.014	1.821	14.889	4.332
	–	0.176	0.230	0.361
-	1.041	0.032	0.860	0.089
-	0.309	0.043	0.542	0.210
-	–	–	–	–
-	2.364	2.072	16.521	4.992
-	0.653	0.078	0.851	0.616
-	–	–	–	–
-	–	–	–	–
-	0.141	0.240	4.277	0.564
-	–	–	–	–
-	3.158	2.390	21.613	6.172
-	1.545	0.180	2.852	0.981
-	–	0.049	0.042	0.100
-	–	–	0.468	–
-	–	–	–	–
-	1.545	0.229	3.362	1.081
-	–	–	–	3.243
-	–	–	–	– –
-	0.232	–	0.504	–
-	–	–	–	–
-	0.023	–	0.097	–
-	–	–	–	–
-	0.005	–	0.012	–
-	0.260	–	0.613	–
-	0.041	–	2.845	0.191
-	0.019	–	0.028	–
-	1.865	0.229	6.848	3.434
-	–	–	–	–
-	–	–	–	–
-	–	–	–	–
-	–	–	–	–
-	–	–	–	–
-	–	–	–	–
-	–	–	0.037	–
-	–	–	–	–
-	–	–	–	–
-	–	–	–	–
-	–	–	–	–
-	–	–	–	–
-	0.035	–	0.076	–
-	–	–	–	–
-	–	–	–	–
-	–	–	–	–
-	–	–	–	–
-	0.035	–	0.113	–
-	5.058	2.619	28.574	9.606

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(U) The engine option research package costs shown in Figures 4-122 and 4-123 include the engine development and ground test facilities costs in order to show the impact of the engine development cost, which is significant. The incremental costs associated with the six options are shown in bar chart form in Figure 4-123 to show the impact of their respective costs. The development costs associated with the scramjet and convertible scramjet engines are 150 $\overline{M}$ and 175 $\overline{M}$ dollars, respectively. The ground test facility costs are obtained from Volume IV, Part 2, Section 7.7.

(U) FIGURE 4-122 TOTAL SYSTEM COSTS FOR THE BASIC MACH 12 CONFIGURATION AND ASSOCIATED OPTIONS (Millions of 1970 Dollars)

Cost Categories	BASIC VEHICLE	OPTIONS (1) 1. SJ (2)	2. CSJ (3)	3. STG	4. ARM	5. HTO/VTO	6. TPS	7. TJ	BASIC VEHICLE Plus (4) Inst. Prov.
1. RDT&E	125.6	185.2	230.2	9.2	7.6	7.9	3.2	24.0	131.8
2. Investment	137.4	25.7	32.5	4.6	1.3	5.5	1.9	7.1	140.8
3. Operating	87.9	.5	.6	-	-	.2	-	.1	87.9
TOTAL	350.9	211.4	263.3	13.8	8.9	13.6	5.1	31.2	360.5

(1) Includes cost of provisions in the basic vehicle plus the research option package.

(2) Propulsion Dev. Cost - 150 $\overline{M}$ engine, 15 $\overline{M}$ ground facilities.

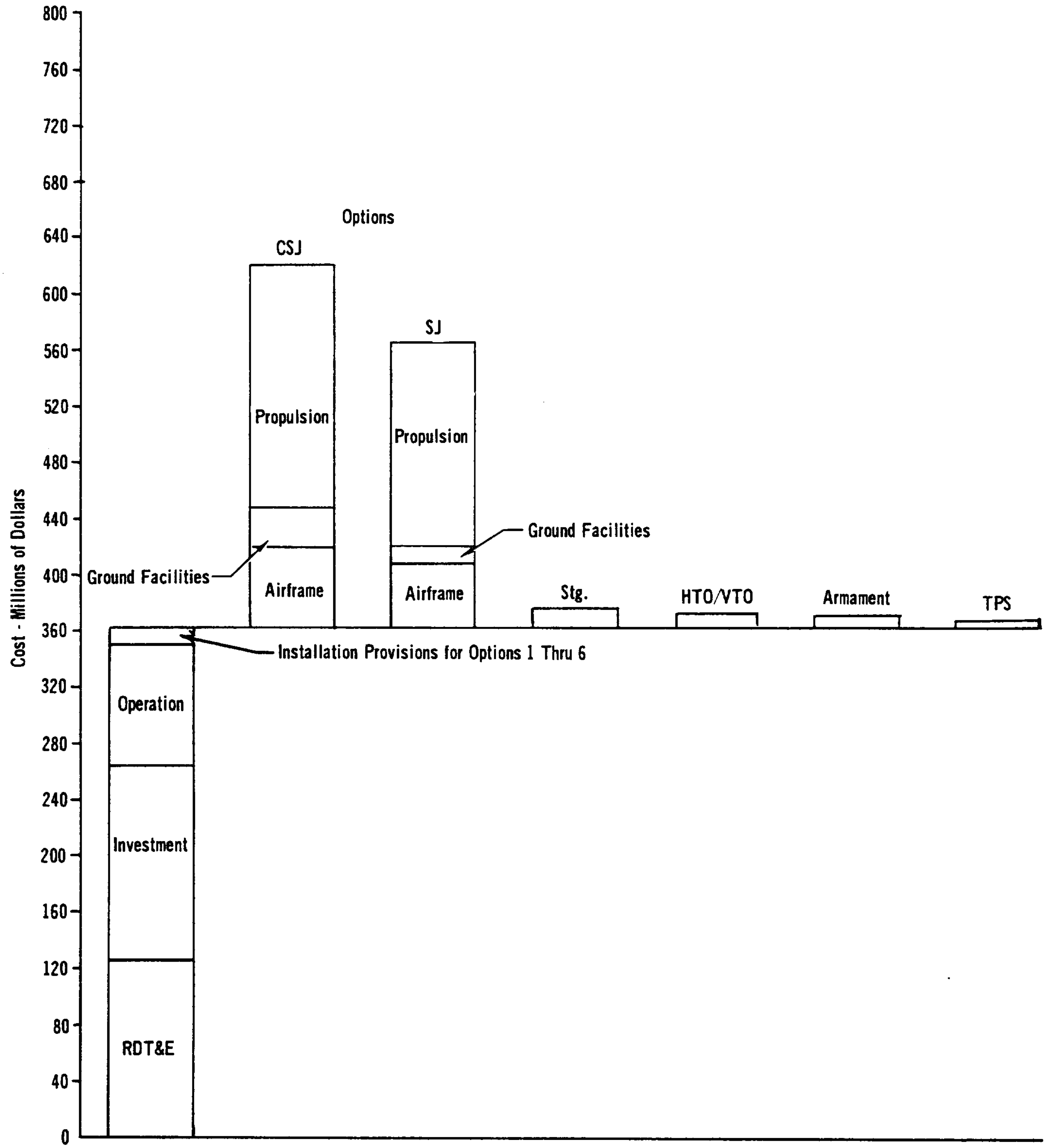
(3) Propulsion Dev. Cost - 175 $\overline{M}$ engine, 30 $\overline{M}$ ground facilities.

(4) Installation provisions for options 1 thru 6.

convertible scramjet engine options. The most expensive options are the engine and staging options, while the least expensive options are the ARM, HTO/VTO, and TPS options. All research option package costs were based on one unit, except the STG option, which was based on ten units. This is the reason why the STG option is one of the higher-priced research vehicle options. Three of the seven options required incremental material costs to allow for the maintenance/repair of the additional engines: namely, the SJ, CSJ, and TJ options. The maintenance/repair material cost increments for the SJ, CSJ, and TJ options are: .152 $\overline{M}$, .206 $\overline{M}$, and .037 $\overline{M}$ dollars, respectively. These incremental material costs were computed on the following basis.

- SJ - \$2,800,000 (Eng. Inv. cost) x .002276 x 24 (No. of flights)
- CSJ - \$3,500,000 (Eng. Inv. cost) x .002276 x 36 (No. of flights)
- TJ - \$468,000 (Eng. Inv. cost) x .006828 x 12 (No. of flights).

(U)FIGURE 4-123
OPTION COST COMPARISON SUMMARY



The factors of .002276 and .006828 used in the above computations represent the ratio of the material cost to the total investment cost of the engine on a per flight basis. The factor of .002276 was obtained from the X-15 data presented in Figure 3-11. This factor was increased by a factor of 3 for the turbojet engines employed in the TJ option to account for the added complexity of a turbojet engine. Labor costs were not increased because it was assumed that the number of engine maintenance personnel located at the refurbishment center allocated for the basic vehicle's propulsion system would be sufficient for the refurbishment of the additional engines.

(U) An estimate of the manhours required and the respective costs for the seven modifications associated with the Mach 12 vehicle are presented in Figure 4-124. These estimates are based on all work being accomplished by NASA personnel at the flight research center at Edwards AFB and pertain to the installation of the research packages. The installation provisions are installed on the basic vehicles at the contractor's manufacturing facility.

(U) FIGURE 4-124
OPTION INSTALLATION COSTS

Options	Time (Mos)	No. of Men	Hr/Rate $/Man Hr (1)	Total Man Hrs (2)	Total Opt. Inst. Cost (3)
SJ	6	12	6.66	12,456	83,000
CSJ	6	12	6.66	12,456	83,000
STG	3	12	6.66	6,228	41,500
ARM	4	12	6.66	8,304	55,300
TPS	2	8	6.66	2,768	18,500
HTO	2	8	6.66	2,768	18,500
VTO	2	8	6.66	2,768	18,500
TJ	2	12	6.66	4,152	27,700

(1) Based on $13,000/yr (1970) NASA shop personnel cost and 1,952 working hrs per yr.
(2) Based on 173 manhours per month.

4.12 DEVELOPMENT SCHEDULE

(U) The development schedule for the Mach 12 flight research vehicle is presented in Figure 4-125 which is a milestone chart depicting the important events and their time of occurrence measured from "go-ahead".

(U) The schedule reflects the cost ground rules stated in Section 3.6.1, that is, (1) minimum cost-to-fly program, (2) soft tooling, (3) limited reliability program, (4) "Zero Defects" program not employed, (5) five year operational test program, (6) limited pre-delivery flight test program and (7) maximum use of existing equipment.

(U) The time allocated for the engineering design of the airframe is 36 months with design freeze occurring 12 months after "go-ahead". Development tests are conducted for a period of 18 months and terminate at the mid-point of the airframe engineering design phase. Three major categories of tests are performed during the 18 month development test period: namely, (1) wind tunnel tests, (2) structural and thermal tests and (3) component and subsystems tests. The assembly of the structural test article begins 16 months after "go-ahead" and is completed 28 months after "go-ahead". Assembly of the three vehicles begins 19 months after "go-ahead" and terminates with the completion of the third vehicle 51 months after "go-ahead".

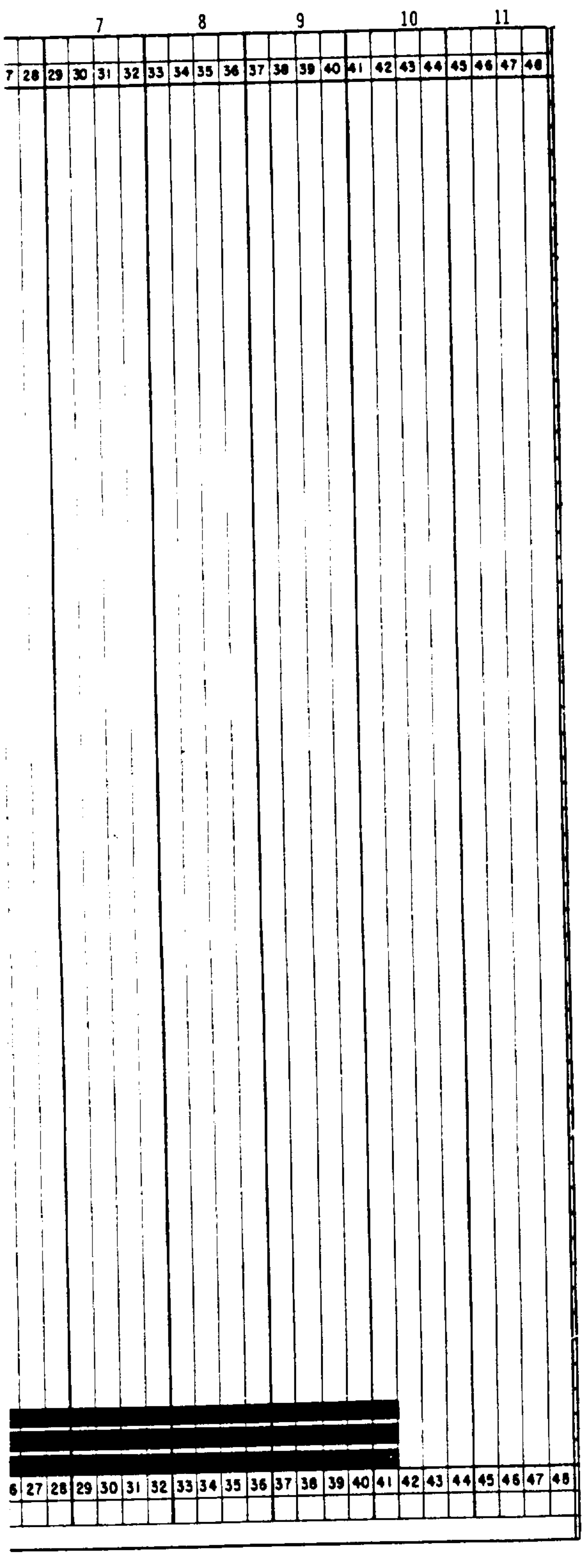
(U) The engine development is concurrent with airframe development; hence, the schedule is not paced by the development of the RL10-A-3-9 rocket engine.

(U) Due to the use of soft tooling, the tool design and fabrication program is only 15 months in duration.

(U) The pre-delivery flight test program is also 15 months in duration. All three aircraft require powered flights while the first two aircraft require captive flights. Only aircraft number 1 requires glide flights. The first captive flights are performed on the number 1 vehicle, 37 months after "go-ahead".

(U) Three months later (40 months after go-ahead) the first glide flight takes place. The first powered flight occurs 48 months after "go-ahead". Upon completion of the first powered flight, the flight research vehicle undergoes four months of flight tests prior to being delivered to NASA. The remaining two vehicles are also delivered at this time.

(U) Twenty five flights are conducted during the pre-delivery flight test program: 3 captive flights, 2 glide flights and 20 powered flights. During the pre-delivery flight test phase, the envelope of the flight vehicle is expanded from sub-sonic to Mach 6 speed. Upon completion of the pre-delivery flight test phase, the envelope expansion phase is initiated. In this phase, 45 powered flights are scheduled to expand the envlope from the Mach 6 to Mach 12 regime. These 45 flights are flown without any test time alloted; that is, the entire mission is dedicated to the expansion of the Mach No. envelope. An additional 30 flights are required for: (1) structural thermo research at sustaining Mach numbers, (2) aerodynamic research, and (3) subsystems research. Test time is allocated for these 30 flights and is expanded with Mach No. Hence, 95 powered flights and 5 unpowered flights are accomplished by the basic vehicle(s) prior to the modification research phase of the program. For the modification research program, 100 powered flights are allocated.



Note (1) Includes Fuel System Integral Tank Design and Development

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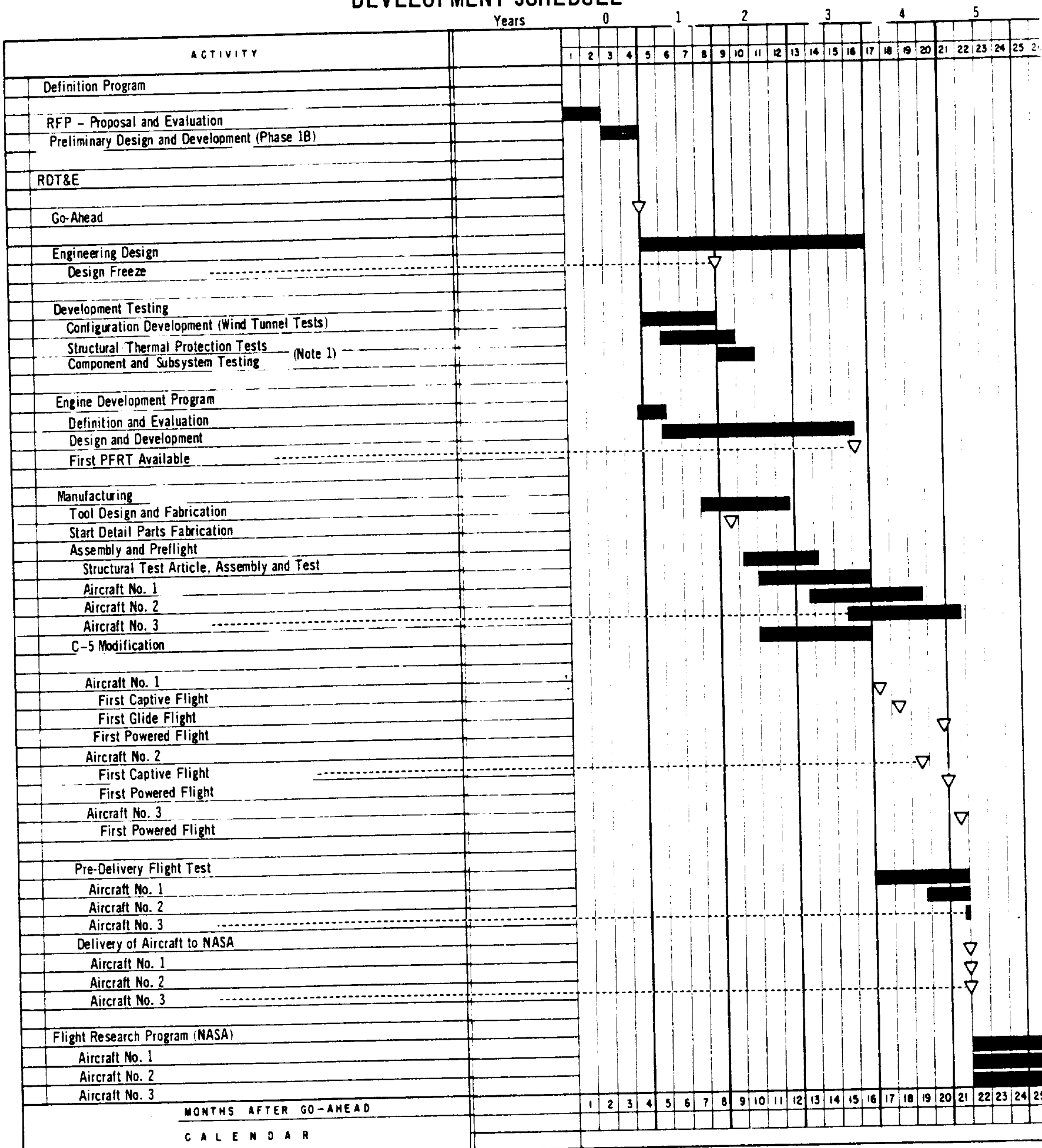
(U) FIGURE 4–125
MACH 12 VEHICLE
DEVELOPMENT SCHEDULE



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5. MACH 6 VEHICLE SYNTHESIS

(U) The basic Mach 6 research aircraft concept selected for Phase III refinement consists of a wing body configuration powered by a near-term turboramjet (STRJ11A-27) propulsion system. It is manned and is designed for conventional aircraft horizontal takeoff and landing. The vehicle, shown in Figure 5-1, was sized to meet the design mission described in Section 2.3. Configuration options were then investigated to provide added research capability, and to determine what effect these design modifications would have on cost, weight, and performance. These data are summarized in Figure 5-2. The substantiating analysis is given in the following sections.

5.1 DESIGN REQUIREMENTS

(U) Design criteria and ground rules selected for the Phase II refinement of the Mach 6 research aircraft are based on the results of the Phase I and Phase II concept selections and trade studies. All of the Phase II ground rules remain applicable in Phase III. However, additional ground rules have been employed as a result of the Phase II conclusions. The design criteria and ground rules for the Phase III Mach 6 aircraft are presented in Figure 5-3.

5.2 CONFIGURATIONS - BASIC AND VARIATIONS

(U) In the following sections, the design, structural arrangement, and materials selection for the basic Mach 6 vehicle and various configuration options are presented. The configurations and concepts shown are considered representative. They are not intended to imply design optimization, nor to infer that the Mach 6 aircraft is capable of incorporating only the research options shown.

5.2.1 (U) VEHICLE DESIGN - The Mach 6 research aircraft concept, though differing significantly from the Mach 12 vehicle, employs the same design philosophy. The aircraft is designed from the standpoint of making provisions, within the basic vehicle structure, to accommodate any of several attractive research configuration options by relatively simple modifications. The modifications considered are in the form of essentially complete research packages or kits.

5.2.1.1 (U) Basic Vehicle - The basic vehicle selected for Phase III study and refinement is a variation of the -210 concept from Phase I. The selection of a near term turboramjet engine in lieu of an advanced technology engine was based primarily on cost considerations. The engine selected is the P&WA STRJ11A-27, which employs an existing, currently in use, JP fueled, J58 turbojet engine modified to include a wraparound hydrogen fueled annular ramjet. The selection of this engine provides high confidence in the propulsion concept and will not require an extensive development program prior to initiating the research program.

(U) Several engine inlet configurations were studied and three designs were investigated, as discussed in Section 2.4. The shoulder mounted design with horizontally oriented inlets and ramps forming a bifurcated duct was selected. It offered more versatility than the lower fuselage inlets by keeping the under surface

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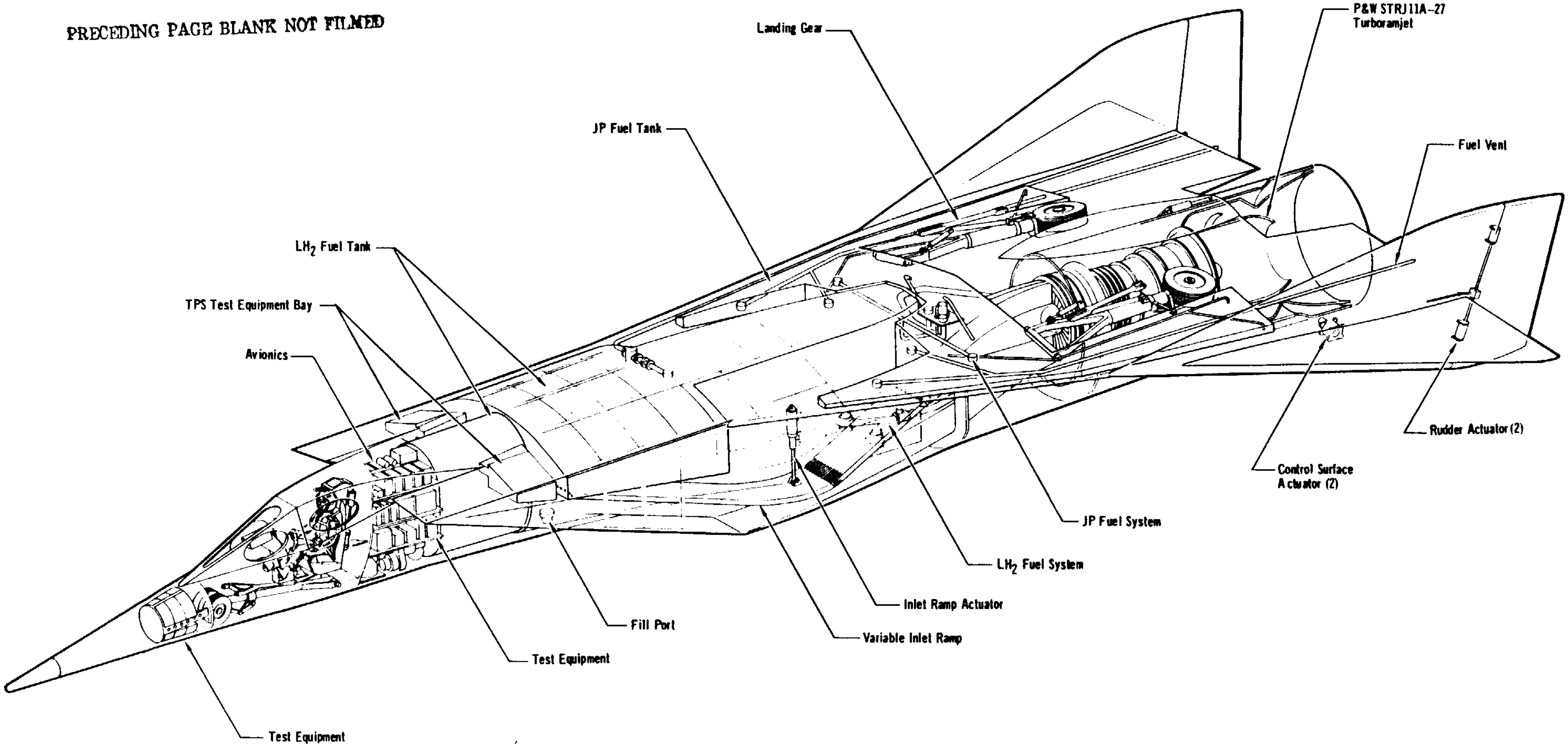
(U) FIGURE 5-1
MACH 6 AIRCRAFT GENERAL ARRANGEMENT



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(U) FIGURE 5-2
CHARACTERISTICS SUMMARY – M = 6, MANNED, HTO, TRJ VEHICLE – FIXED SIZE

CONFIGURATION DESCRIPTION	PERFORMANCE		WEIGHT LB (KG)		ACQUISITION COST MILLION DOLLARS
	MACH	TIME (MIN)	OWE	TOGW	
BASIC VEHICLE	6.0	5.0	48456 (21980)	61426 (27863)	398
ALL LH_2 STRJ11A-27	5.7	0	48456 (21980)	53406 (24225)	-
℄ LH_2 TANK STRJ11A-27	6.0	1.8	48956 (22206)	55506 (25178)	-
ARM OPTION	6.0	3.2	50648 (22974)	63618 (28857)	402
TPS OPTION	6.0	5.0	48792 (22214)	61762 (28015)	405
ADVANCED TURBORAMJET JZ6 (96.5% SCALE)	6.0	0.3	46144 (20931)	51504 (23362)	(1) 443
JZ6 + ℄ LH_2 TANK	6.0	2.2	46644 (21158)	53604 (24315)	-

(1) EXCLUDES $500 MIL. ENGINE AND $376 MIL. GROUND FACILITY DEVELOPMENT COST

NOTE: RAMJET AND SCRAMJET OPTIONS - DESIGN CONCEPTS ONLY WERE DEVELOPED.

open for modification options. The selected design also creates lower inlet air turning angles and reduces possibilities of foreign object damage during ground operations. As a result of this arrangement, the TRJ inlet was located in the free stream rather than within the vehicle pressure field, as in previous phases. For this reason, a new inlet configuration was generated to ensure high performance. Figure 5-3 illustrates the basic Mach 6 vehicle design and Figure 5-4 presents the wetted area and volume plots for the vehicle.

(U) A passive thermal protection system is utilized on the basic vehicle. The high wing, bifurcated inlet duct made a passive system more attractive than an active TPS (initially recommended in Phase II) to reduce structural thermal gradients from the inlet duct to the outside moldline and from the inlet duct to the engine compartment. The inside walls of the bifurcated inlet ducts are regeneratively cooled strucuure. The airframe structure is basically aluminum with titanium used in the inlet area where advantageous.

(U) The design mission of the Mach 6 aircraft is described in Section 2.3. To reduce base drag, the hydrogen fueled ramjet is started at Mach .8 and continues to operate to fuel depletion. The turbojet and ramjet operate together over the speed range of Mach .8 to Mach 3.5. At Mach 3.5 the turbojet is shut down and the

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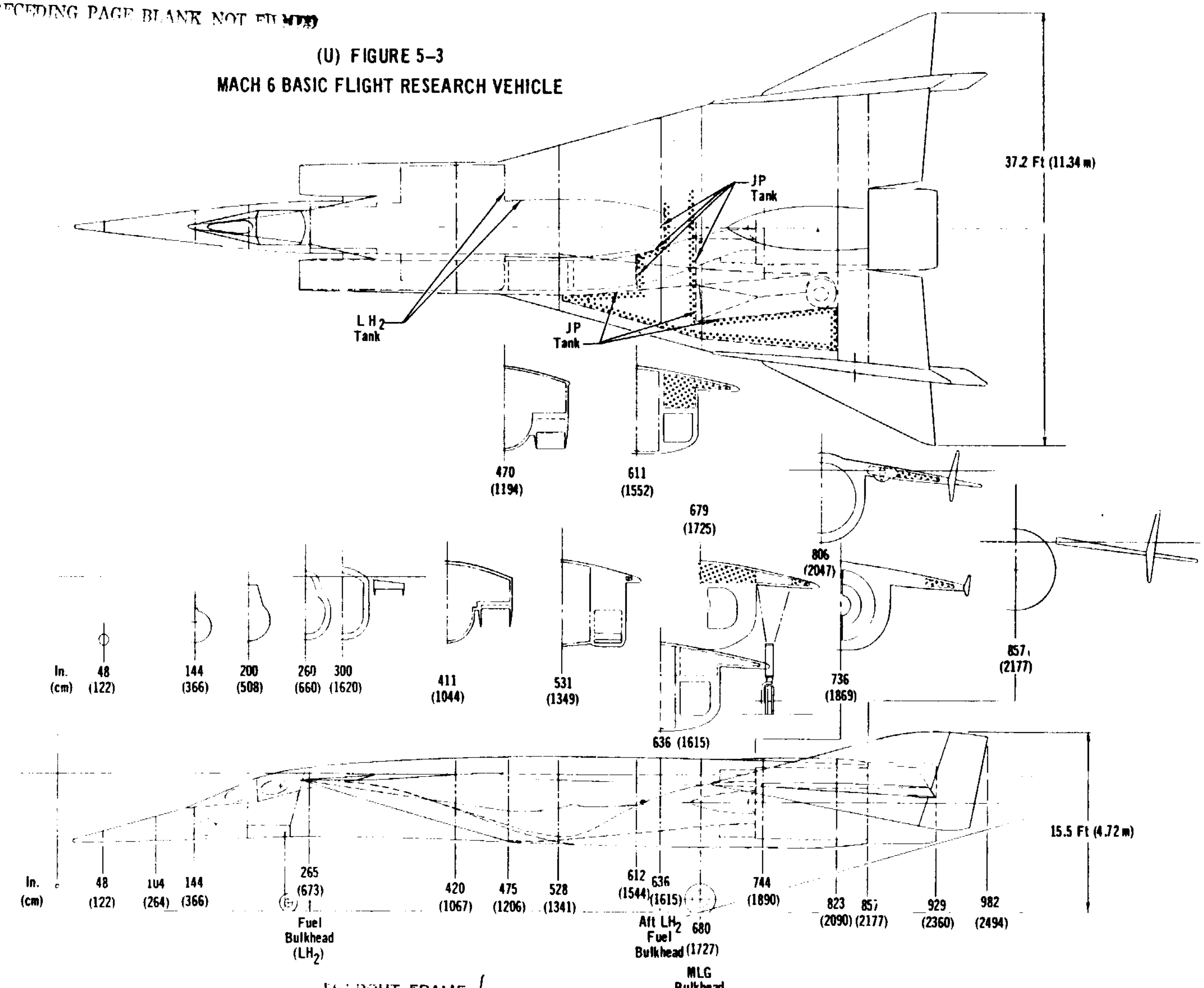
(U) FIGURE 5-3
MACH 6 BASIC FLIGHT RESEARCH VEHICLE



Physical Characteristics			
Wing			
Area (Theor.)		852 ft^2	(79.15 m^2)
C_R		696 in.	(17.68 m)
C_T		181 in.	(4.60 m)
Sweep LE/TE		75^0/5^0	
T/C		3%	
MAC		483 in.	(12.27 m)
Dihedral		–8^0	
Incidence		0^0	
Wing Tips (Movable)			
Area (total)		102 ft^2	(9.48 m^2)
C_R		175 in.	(4.44 m)
C_T		8 in.	(.20 m)
Sweep LE/TE)		65^0/5^0	
MAC		117 in.	(2.97 m)
T/C		3%	
Vertical Tail			
Area (total)		224 ft^2	(20.81 m^2)
C_R		283 in.	(7.19 m)
C_T		87 in.	(2.21 m)
MAC		258 in.	(6.55 m)
T/C		3%	
Sweep LE/TE		75^0/10^0	
Toe-In		5^0	
Cant Out		8^0	
Propulsion			
P&WA STRJ11A-27	Length	193 in.	(4.90 m)
	Dia. Inlet	73 in.	(1.85 m)
	Dia. Nozzle	84 in.	(2.13 m)
Propellant (Usable)			
LH_2 (RJ)		913 ft^3	(25.84 m^3)
JP (TJ)		141 ft^3	(3.99 m^3)
Planform Area		1103 ft^2	(102.47 m^2)

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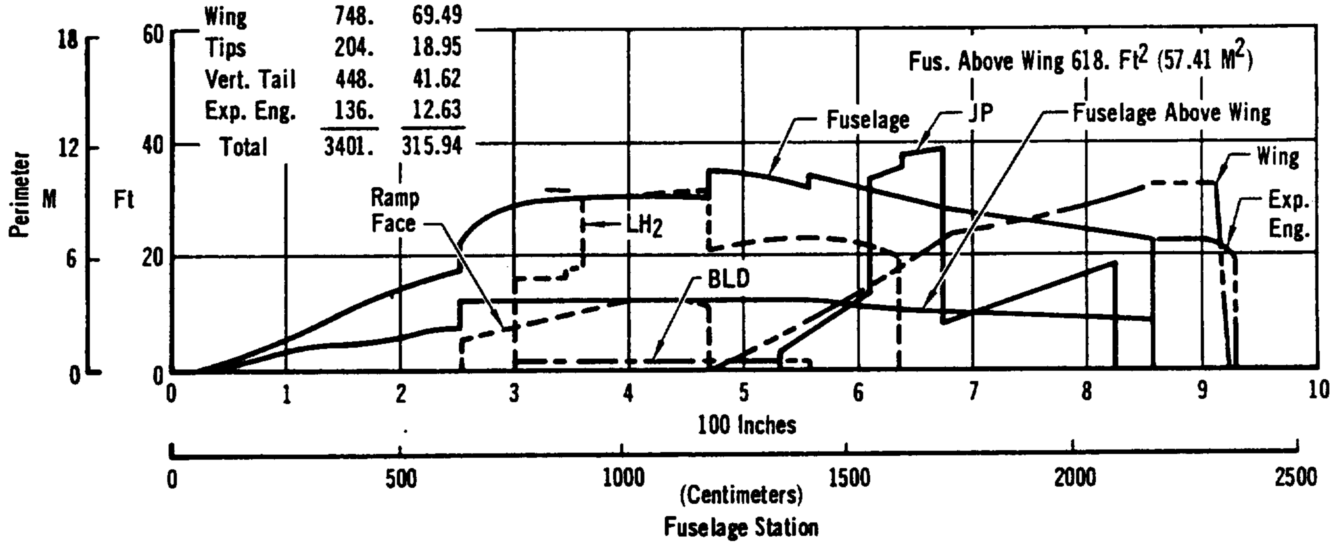
(U) FIGURE 5-4
MACH 6 VEHICLE
Wetted Area and Volume Plot
$S_p = 1103\ Ft^2$

Wetted Area	Ft^2	M^2
Fuselage	1650.	153.28
Ramp Face	180.	16.75
BLD	35.	3.25
Wing	748.	69.49
Tips	204.	18.95
Vert. Tail	448.	41.62
Exp. Eng.	136.	12.63
Total	3401.	315.94

Deductions	Ft^2	M^2
Ramp	180.	16.72
Exp. Eng.	136.	12.63
Total	316.	29.35

Fuel Tank Wetted Area

Propel.	Ft^2	M^2
LH_2	667	61.96
J.P.	410	38.09

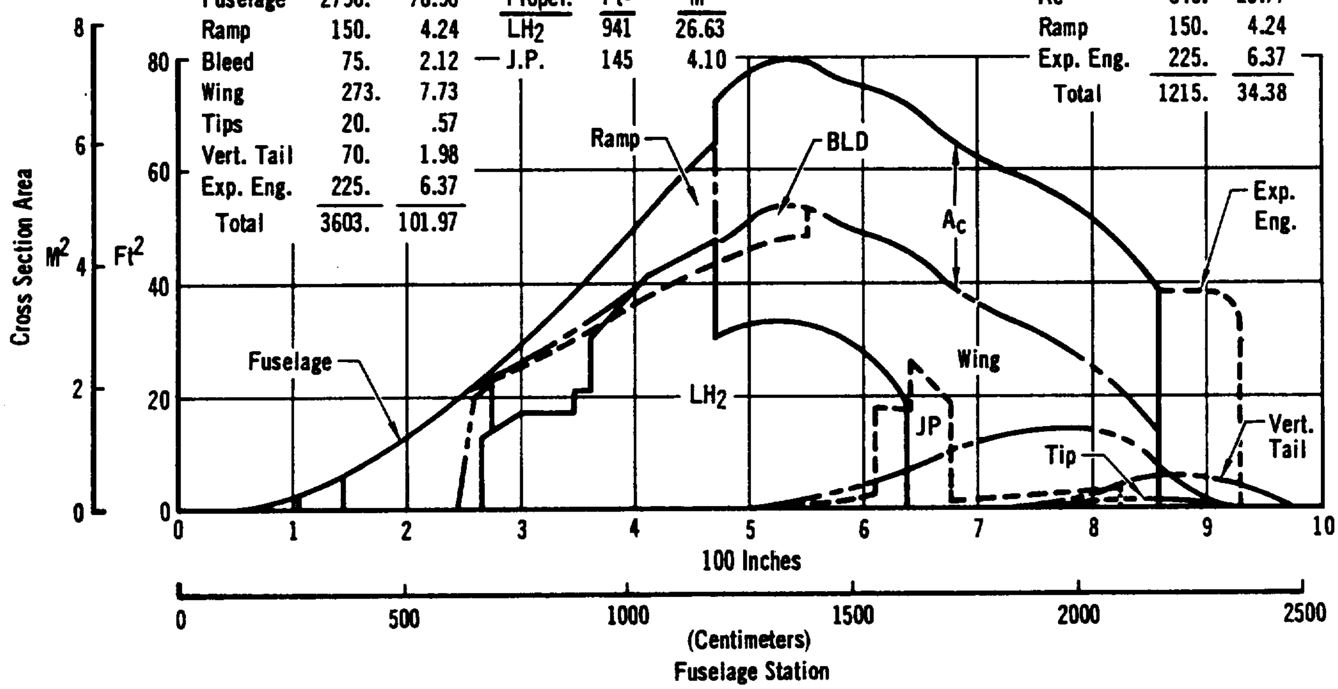


Volumes	Ft^3	M^3
Fuselage	2790.	78.96
Ramp	150.	4.24
Bleed	75.	2.12
Wing	273.	7.73
Tips	20.	.57
Vert. Tail	70.	1.98
Exp. Eng.	225.	6.37
Total	3603.	101.97

Fuel Tank Volume

Propel.	Ft^3	M^3
LH_2	941	26.63
J.P.	145	4.10

Deductions	Ft^3	M^3
Ac	840.	23.77
Ramp	150.	4.24
Exp. Eng.	225.	6.37
Total	1215.	34.38



ramjet accelerates the aircraft to cruise Mach number and is then throttled to maintain constant Mach number until the fuel is exhausted and an unpowered glide and descent is made. The ramjet operates at stoichiometric conditions over the entire acceleration speed range. Energy management to achieve a high key position over the landing base is accomplished by modulating the vehicle L/D as done with the Mach 12 vehicle. A 360 degree overhead approach and landing is made.

(U) Along with the normal complement of avionic equipment, the vehicle carries a payload of 1300 lb (589.7 kg) instrumentation which can be utilized in a manner best suited for the particular research. While a payload of 1500 lb (680 Kg) was selected as a result of the Phase II trade study the point design study of Phase III resulted in a better vehicle integration with a smaller payload while providing adequate research capability. Instrumentation may be altered as required for specific research flights. Each research option will probably require some alteration in the payload to best use the test vehicle capability.

5.2.1.2 (U) Armament (ARM) Option - The armament for possible military applications of a hypersonic vehicle is assumed to be guided missiles or warheads of various sizes. The Mach 6 aircraft utilizes a bomb bay type pod with clam shell doors on the undersurface of the aircraft as shown in Figure 5-5. If desired the pod could be just as easily located on the upper surface of the vehicle. However, previous studies have indicated that for this speed class vehicle the lower surface is probably a better choice for the armament system.

(U) The purpose of this modification is to provide research in the deployment and separation of such weapons, the thermal shock on the weapon, and in the internal heating of a bomb bay type cavity. The use of deflectors and shields to divert airflow and prevent high heat impingement on the inner walls of the open bomb bay are also of interest.

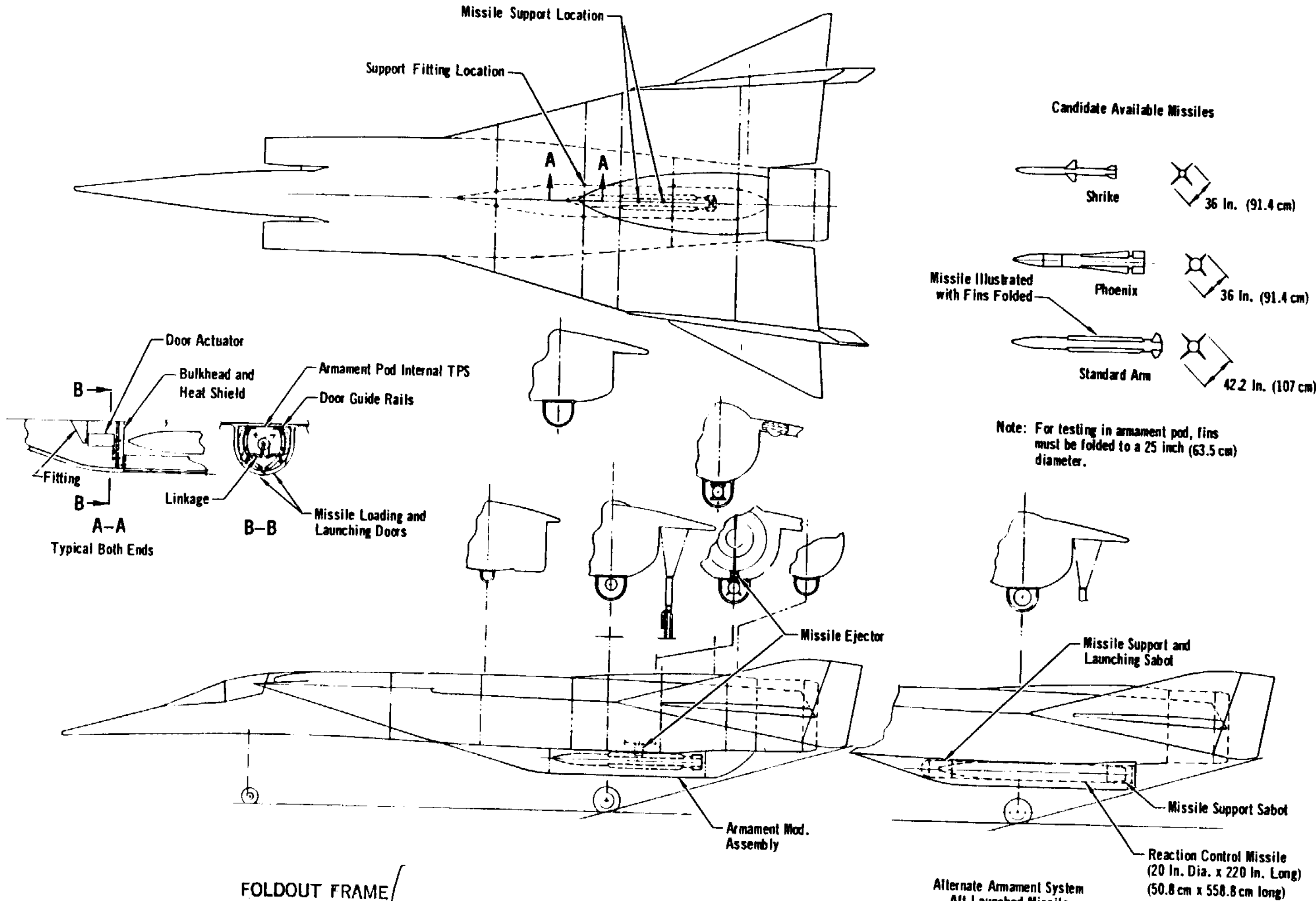
(U) After the deployment methods are defined, missile shapes, weighted appropriately, could be ejected for separation characteristics.

(U) Missile guidance systems will require avionic modifications to the aircraft electronic systems. Initial tests can be made using inertial guidance systems. The aircraft inertial system can be updated by ground tracking systems and the data then fed to the missile guidance system computer for the target acquisition and weapon release. Radar guidance systems will require development of a high temperature radome material for both aircraft and missiles. Such a development program could be incorporated into the Mach 6 test program.

(U) The illustrated missile pod will accommodate a missile or warhead up to 180 in (457 cm) long and 20 in (50.8 cm) in diameter with missile fins folded to fit this envelope. An alternate system, similar to that employed for the Mach 12 aircraft, is shown with two aft firing 15 in (38.1 cm) diameter, 165 in (419 cm) long launch tubes. The centering sabots and gas generating cartridge propulsion system are the same as discussed in Section 4.2.1.7.

5.2.1.3 (U) Thermal Protection System (TPS) Option - The TPS test section is located aft of the forward fuel bulkhead in a cryogenic fuel tank area. The test bay extends around the periphery of the fuselage except in the area behind the inlet ducts. The test bay permits testing thermal protection systems up to 6.0 in (15.2 cm) thick, active or passive systems, regeneratively cooled panels, or any similar type systems required. For thicknesses less than 6.0 in (15.2 cm), special shingle

(U) FIGURE 5-5
MACH 6 ARMAMENT CONFIGURATION OPTION



Baseline Modification for Armament

1. Change Lower ML. Remove baseline vehicle shingles in this area for installation of armament pod.
2. Add structural fittings to bulkheads for armament pod, side and vertical loads (8 required).
3. Add intercostals along lower ML for missile ejector, fore and aft loads, missile support lugs, and door actuator supports.
4. Add electrical and hydraulic disconnect panel.

FOLDOUT FRAME 1

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FOLDOUT FRAME 2

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support stand off intercostals can be used to maintain a flush outside contour. Two equipment bays for instrumentation, cryogenic reservoirs, pumps, etc. are located over the inlet ducts and in the area of the fuselage hidden by the inlet ducts as indicated in Figure 5-6.

(U) This peripheral test bay provides good flexibility in testing systems for long periods of time. Actual environmental conditions are encountered and monitored throughout the flight. All tests will be full scale actual hardware specimens subjected to the temperature, pressure and air flow environment in cryogenic structural applications.

(U) To maintain continuous load paths, the basic structure is tapered from the normal depth of 2.5 in. (6.35 cm) inside the external mold line to a depth of 6.0 in (15.2 cm) inside the mold line. This depth is held constant across the test bay and then tapers the load paths back out to the 2.5 in (6.35 cm) depth inside the mold line of the basic structure. The normal vehicle TPS will be used in this area when no alternate system is being developed or tested.

5.2.1.4 (U) Advanced Turboramjet (JZ6) Option - The GE5/JZ6 hydrogen fueled turboramjet engine was originally considered for the basic aircraft but because of its advanced technology status, cost, and unavailability it was not pursued. However, the feasibility of installing this engine when it becomes available in the Mach 6 research vehicle was investigated. Figure 5-7A illustrates a proposed installation of a JZ6 engine sized to the existing inlet airflow capability. Figure 5-7B provides a comparison of the GE 5/JZ6 and the STRJ11-27A engine installations indicating the inlet duct extension moldline changes and tankage alterations. The hydrogen tankage is expanded, the JP fuel provisions are removed, the regeneratively cooled inlet duct is extended to accommodate the shorter engine and the beef-up of the basic structure makes this an extensive, but feasible, modification. The vehicle, as shown, requires additional liquid hydrogen fuel to obtain a satisfactory test time at test Mach number. It can be provided by adding an external LH_2 fuel tank to the centerline station, as shown in Figure 5-8. The external tank fuel is used to gain altitude and speed to Mach .8 where the empty tank is jettisoned. Internal fuel is then used to accelerate and cruise at the test Mach number for the available test time.

(U) This optional capability to test advanced turboramjets in a proven airframe appears to significantly enhance the long range utility of the basic aircraft concept.

5.2.1.5 (U) Convertible Scramjet (CSJ) Option - The forward inlet ramp angle and the exit nozzle contour establish the additional fuselage depth required to accomodate this modification as shown in Figure 5-9. This propulsion system is effective from Mach 3.5 to Mach 6, which is the design Mach number for the aircraft. The modification extends over the entire length of the vehicle lower surface. The extreme depth of the package creates the need for an extended landing gear but the additional volume is used to house the CSJ fuel tankage and actuators that retract the modular engines during acceleration through the transonic drag rise.

(U) Integration of this option on the vehicle is difficult and not attractive for research because of the limited range of operation. Therefore, performance and weights were not calculated for this modification.

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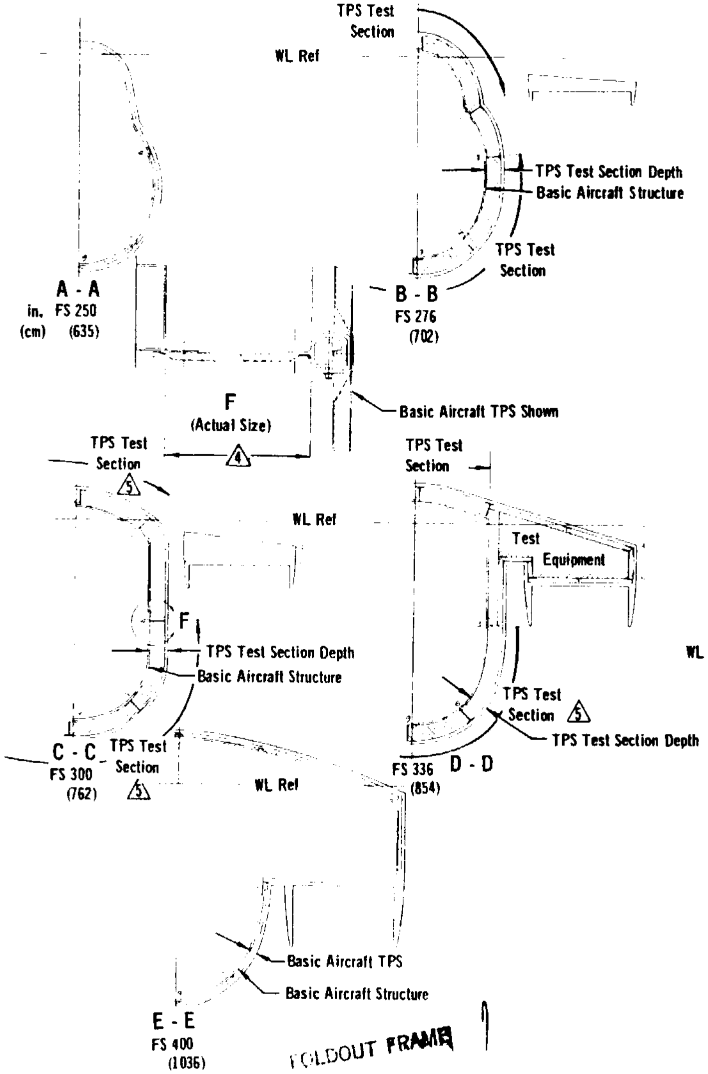
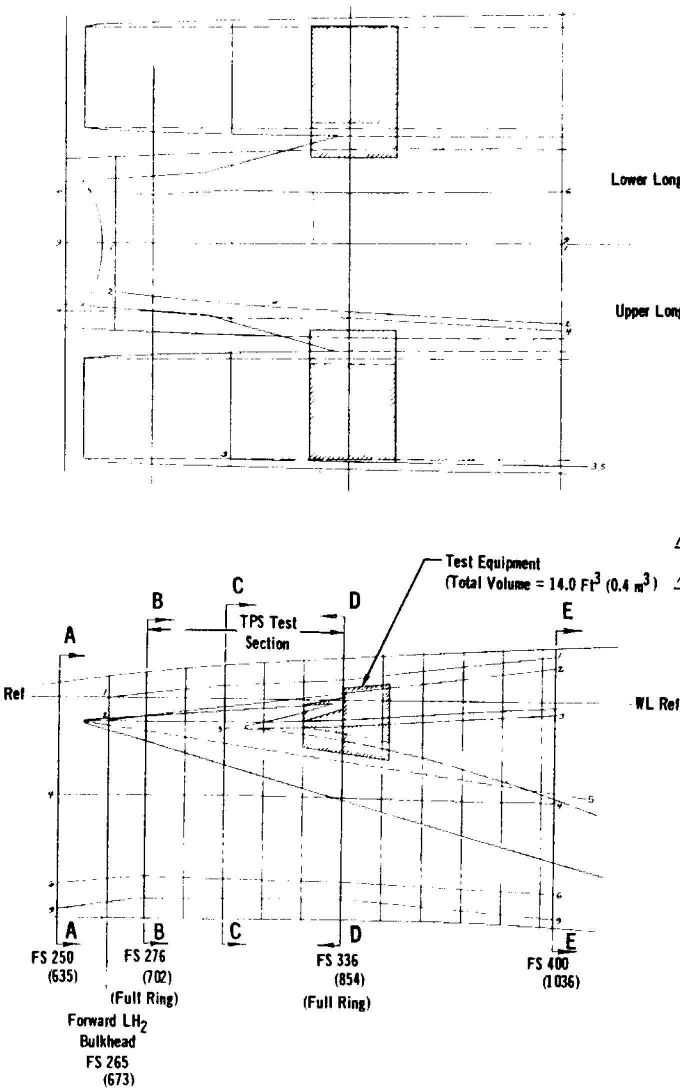
(U) FIGURE 5-6
MACH 6 TPS CONFIGURATION





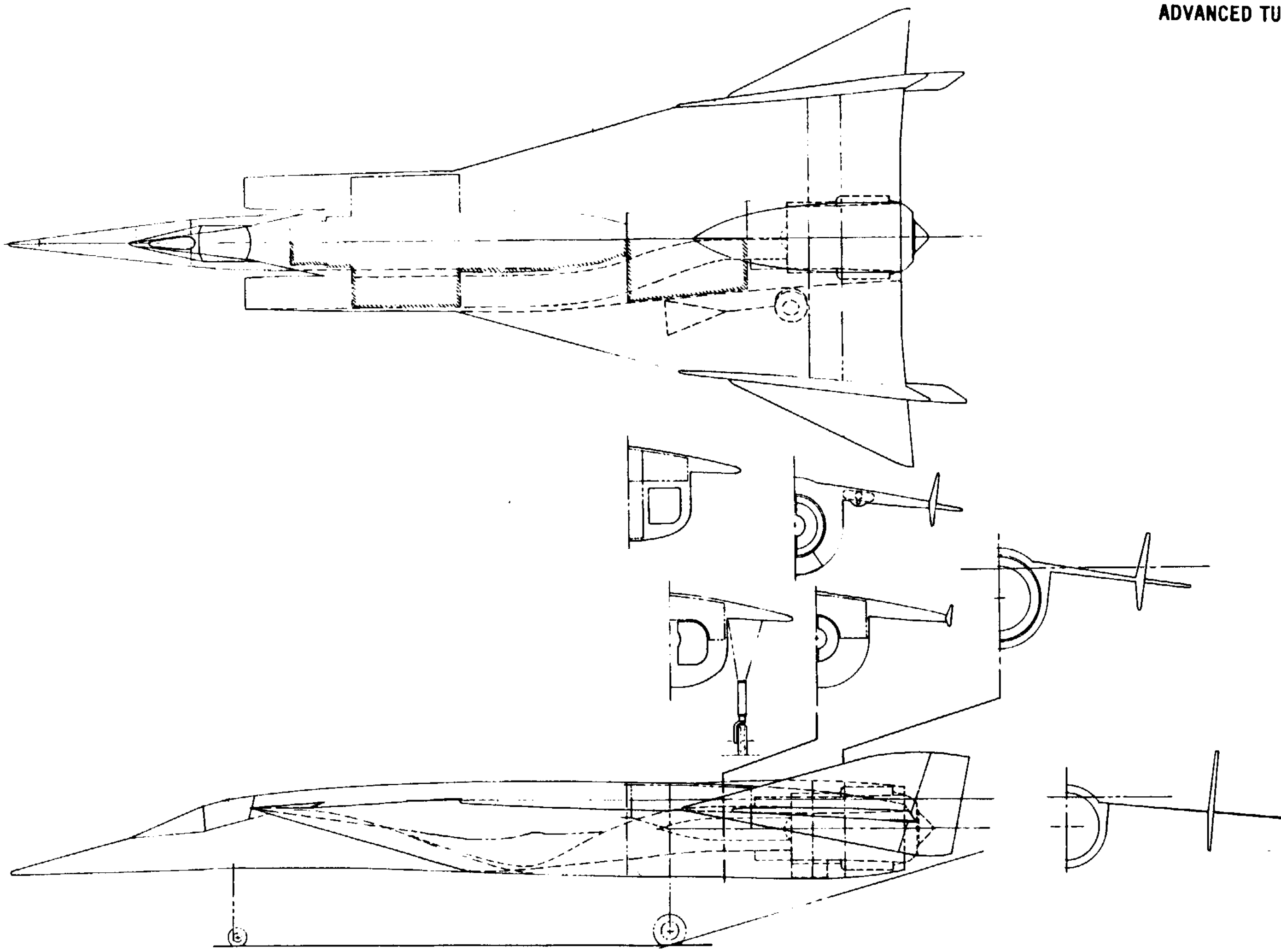
Notes:

1. 6 inches (15.3 cm) minimum maintained between fuel tank ML and fuselage ML for installation of alternate TPS concepts on upper fuselage between longerons No. 2 right and No. 2 left and on lower fuselage between longerons No. 4 right and No. 4 left.
2. Added test equipment bays above each inlet.
3. Tapered structure kick loads redistributed at A-A, B-B, E-E and D-D.
4. TPS support intercostals can be varied in depth to accommodate test TPS and maintain aircraft outside mold line.
5. Basic aircraft TPS removable for test TPS installation in this area.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

(U) FIGURE 5-7A
ADVANCED TURBORAMJET CONFIGURATION OPTION

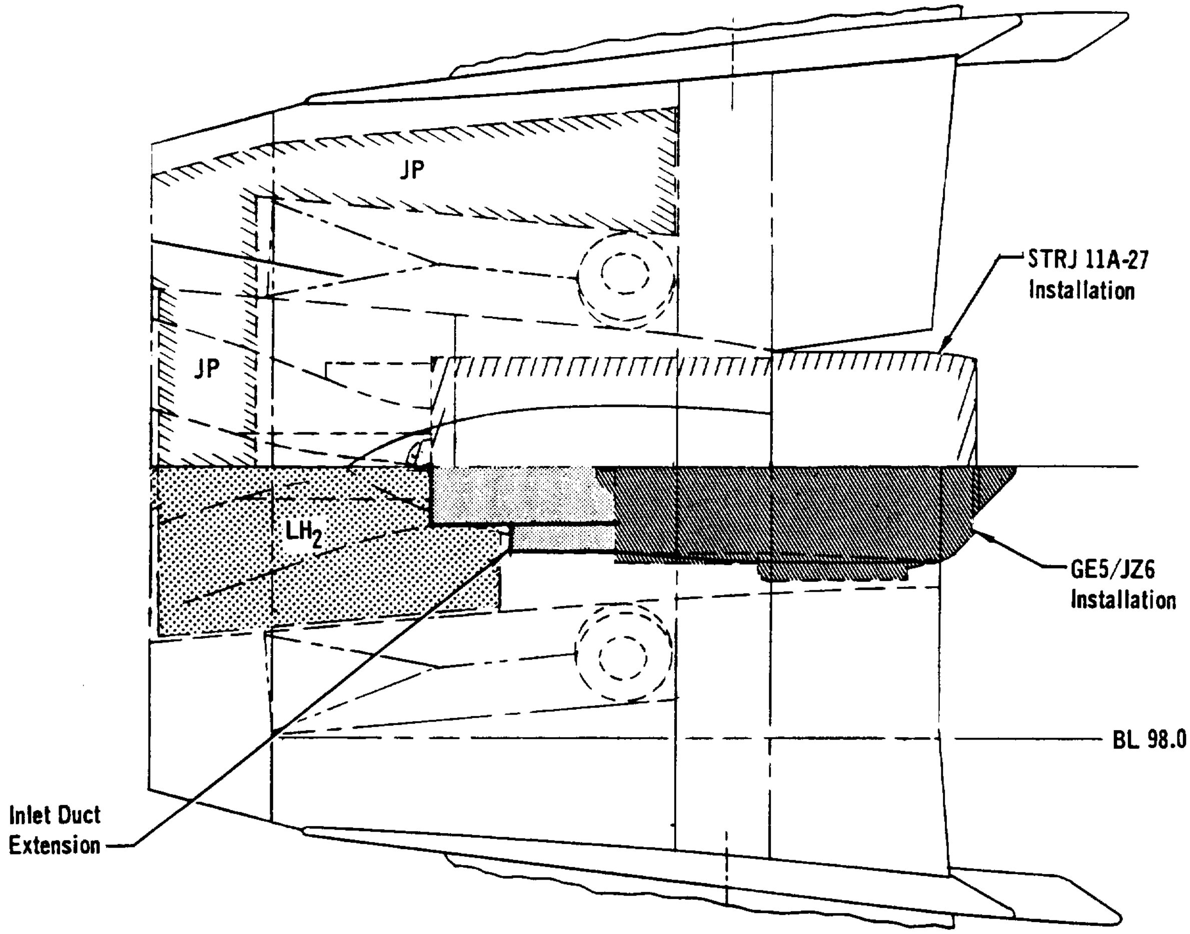


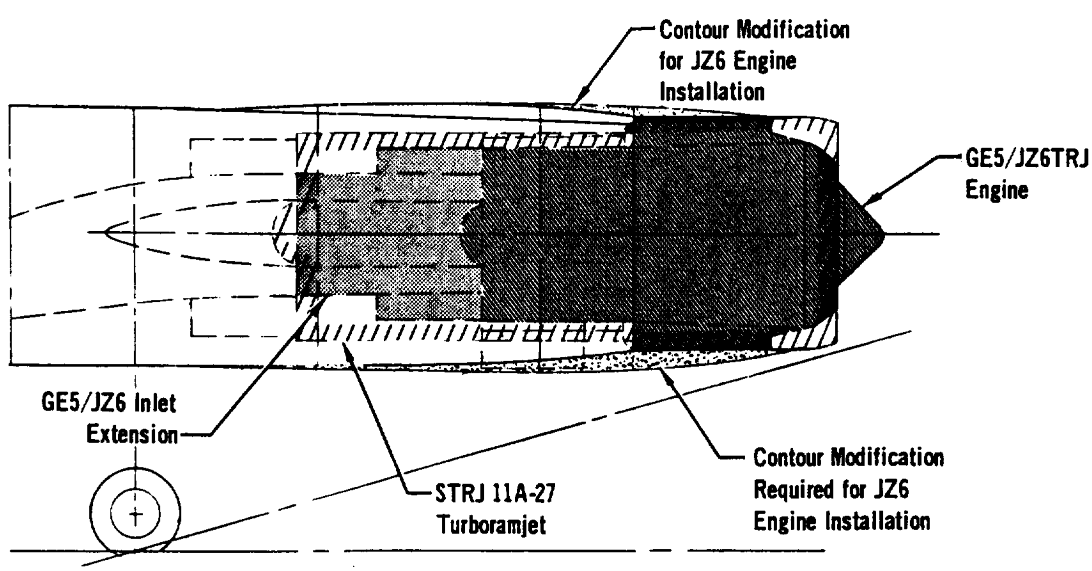
Notes:
1. Remove P&WA STRJ11A-27 engine and its plenum chamber.
2. Remove all JP fuel tanks and associated plumbing.
3. Beef up frame for GE5/JZ6C engine, add a support ring.
4. Change skin ML and extend fuselage.
5. Add constant section engine inlet ducts with divider.
6. Add LH_2 fuel tank. New tank to be joined to existing fuel tank.
7. Fuselage to be refaired to wing trailing edge and between buttock lines 98 in. (249 cm).

FOLDOUT FRAME 2

FOLDOUT FRAME 1

(U) FIGURE 5-7B
ENGINE INSTALLATION COMPARISON
STRJ 11A-27 vs GE5/JZ6





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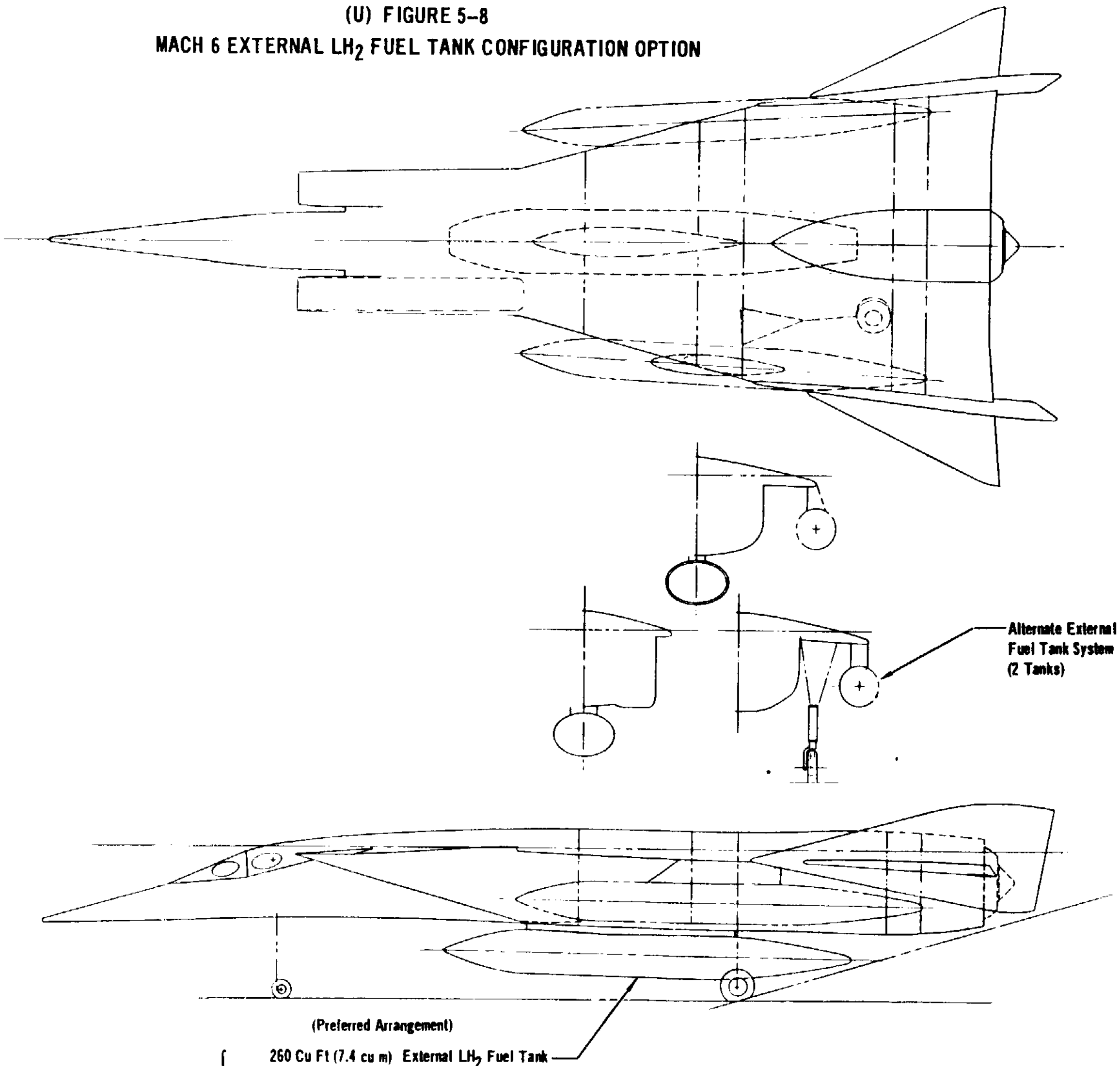
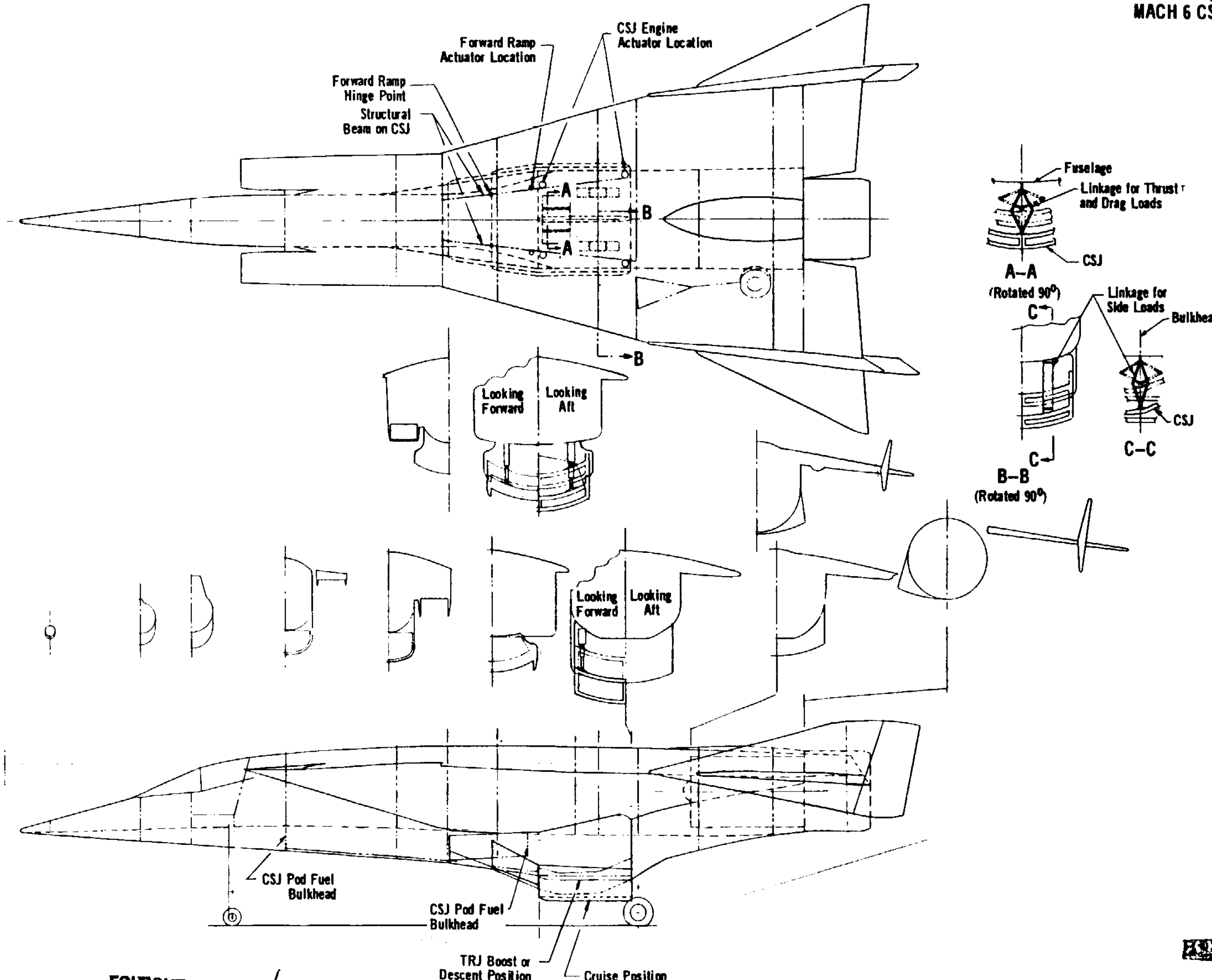
(U) FIGURE 5-8
MACH 6 EXTERNAL LH_2 FUEL TANK CONFIGURATION OPTION



(U) FIGURE 5-9
MACH 6 CSJ CONFIGURATION OPTION



Basic Modification for CSJ

1. Added provisions for thrust and drag load fitting attachment along ℄ lower longeron. Beef up longeron in this area.
2. Added local fittings to bulkhead at B–B for side load reaction.
3. Changed lower ML from the nose to the end of the fuselage by addition of CSJ engine with required ramps, sideplates and nozzle. Remove baseline vehicle shingles in this area.
4. Local beef-up of all bulkheads for ramp, engine and nozzle support. Add structural fuselage attach points for CSJ attach fittings at these bulkheads for vertical and side load reactions.
5. Added intercostal along ℄ lower longeron for thrust and drag load of CSJ.
6. Added 27 inch (68.5 cm) extension to main and nose landing gears. Beef-up nose and main landing gear bulkheads.
7. Beef up shingles on lower surface of wing and fuselage aft of pod for increased sonic fatigue strength.

CSJ Pod

1. Skin and stringer construction with frames to attach to aircraft bulkheads that support the thermal protection system and contour shingles on fixed ramps, engine module, and fixed exit nozzle.
2. Retractable CSJ modules in regeneratively cooled engine package.
3. CSJ fuel tanks, pumps, lines and controls are integral with the pod.
4. Add nose gear door in pod. Remove aircraft nose gear door.

5.2.1.6 (U) Ramjet (RJ) Option - The proposed option concentrates on flight testing small ramjet engines, such as fixed inlet ramjets for missile applications and subscale (20 inch, 50.8 cm inlet diameter) variable geometry engines. However, data and experience gained from these tests would be applicable in advancing the state-of-the-art of designing larger ramjet and turboramjet systems.

(U) This option, Figure 5-10, utilizes a shock wave generator and a boundary layer diverter to produce predictable inlet conditions. JP fuel for the missile application is carried in the dummy missile shape. The variable inlet subscale ramjet could be hydrogen fueled from the basic vehicle fuel supply. Locally high heating due to ramjet shock waves impinging on the basic vehicle would require the use of Columbium shingles or an ablative coating in the interference region surrounding the ramjet option.

5.2.2 (U) STRUCTURAL ARRANGEMENT - The arrangement, material, and fabrication methods of the structural elements have been selected to provide a low weight high reliability structure and to provide the capability to economically modify the aircraft for selected research options.

5.2.2.1 (U) Basic Vehicle - The basic Mach 6 aircraft structure is significantly different from the Mach 12 vehicle primarily because of the airbreathing engine and inlet (Figure 5-11). In the forward part of the aircraft the fuselage structure is integrated with the cryogenic fuel tanks. In the aft part it is integrated with a high temperature inlet and engine.

(U) Inlet and engine compartment temperatures at Mach 6 are in the 2500 - 2600°F (1640-1700°K) temperature range. The inlet is regeneratively cooled and therefore also provides the capability for a dash to higher speed. The heat exchanger on the inner surfaces of the inlet is designed to hold temperatures below 1550°F (1115°K) to allow the use of the most structurally efficient superalloy, Rene' 41. Passive insulation behind the liner protects the primary structure from the 1550°F (1115°K) heat exchanger temperatures. The insulation was sized to maintain the primary structural (titanium frame and skin) temperature at 300°F (422°K).

(U) The cryogenic tank structure is similar in concept to the Mach 12 fuselage tank arrangement. It is of aluminum stiffened skin construction with mechanical fasteners and is integral with the primary fuselage structure. The fuel tank takes the shape of the fuselage moldline rather than a multi-bubble shape, like the Mach 12 aircraft, because vehicle weight is more sensitive to the inlet/engine integration than to tank shape. Fuel tank structure, as in the non-fueled areas, is protected with a passive thermal protection system. Although there is a slight weight advantage in using the water wick system, the passive system was selected to be consistent in concept with the engine/inlet area. Structural temperature is limited by the insulation thickness to 250°F (394°K) or less which is dictated by the temperature limit of the internal tank insulation. The LH_2 tank has a layer of polyurethane insulation and a multi-layer FEP Teflon coated Kapton film vapor barrier inside the tank similar to that employed on the Mach 12 aircraft (Ref. 4.2.2.1).

(U) The wing is an integral extension of the primary fuselage structure. It houses the landing gear and JP fuel tank and supports the vertical tail and all control surfaces. It is of multi-stiffened skin construction with spars corresponding to major fuselage bulkhead locations. Due to the extremely short wing span, ribs

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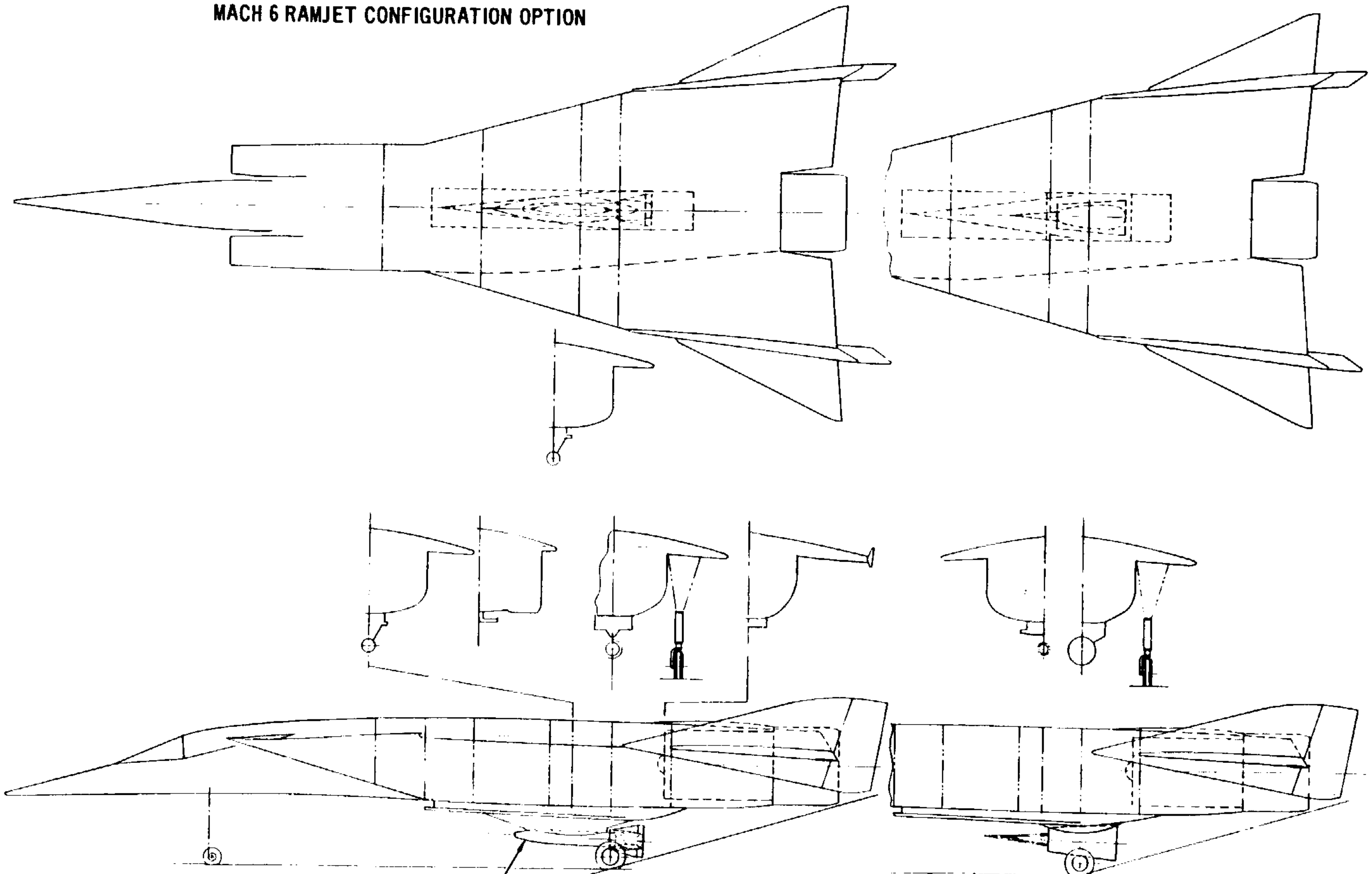
(U) FIGURE 5-10
MACH 6 RAMJET CONFIGURATION OPTION



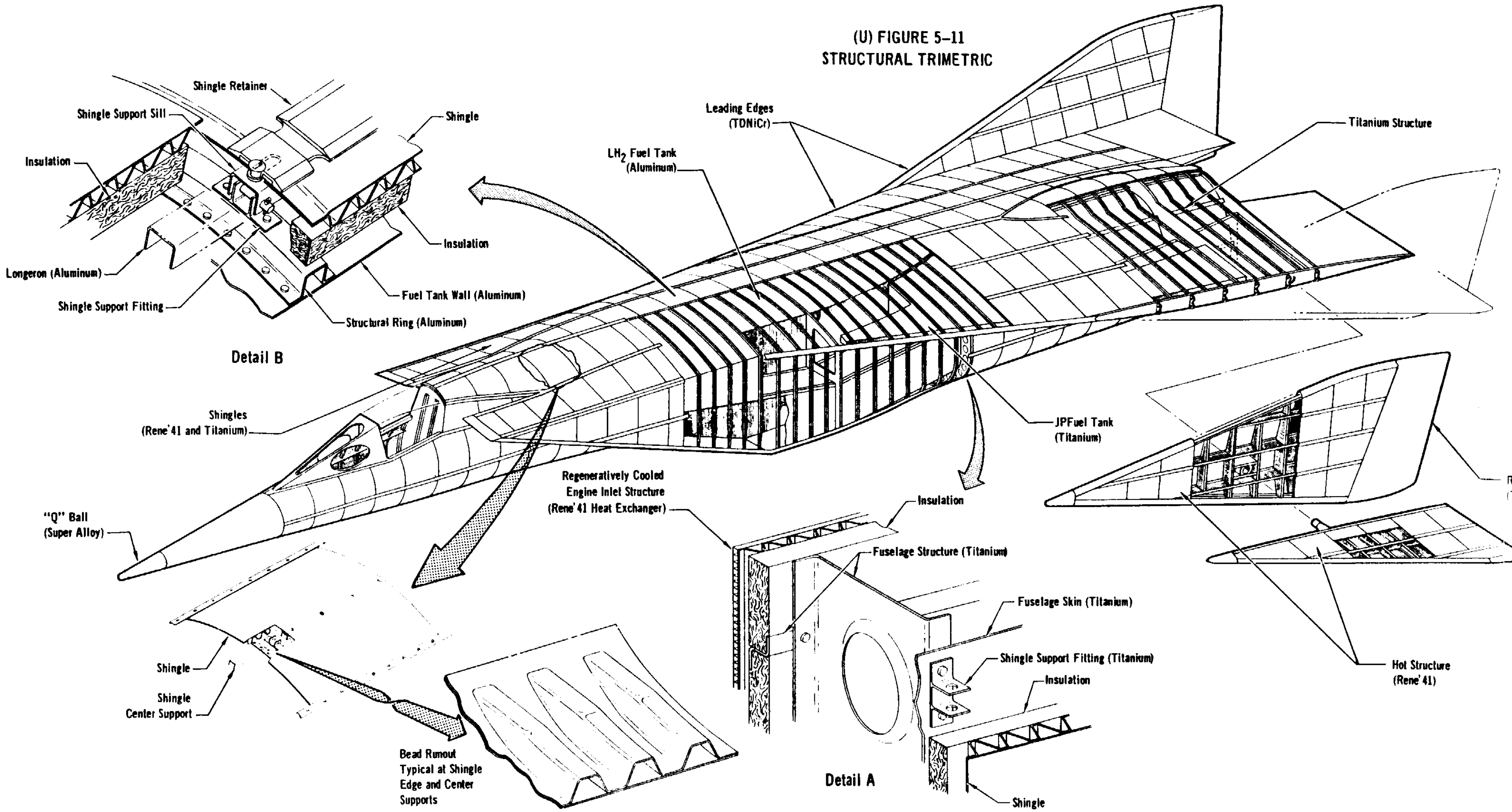
Basic Modification for RJ

1. Local beef-up of bulkheads for ramjet engine, fairing and shock generator support. Add structural fuselage attach points for RJ attach fittings at these bulkheads for side and vertical loads.
2. Add intercostal along lower centerline longeron for RJ thrust and drag loads.
3. Change lower ML by addition of ramjet engine with required fairings and shock generator. Remove baseline vehicle shingles in this area.
4. Beef up shingles along lower surface of wing and aft fuselage for increased sonic fatigue strength.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

(U) FIGURE 5-11
STRUCTURAL TRIMETRIC



FOLDOUT FRAME 2

FOLDOUT FRAME 1

and intercostals are used only where required to close the landing gear well and the JP fuel tank. The fuel tank is integral with the structure. Spanwise stiffening members are simply extensions of the fuselage stiffening rings.

(U) Landing Gear - Landing gear is of conventional design. Struts and supports are high strength steels, brakes are a low weight high heat capacity material such as beryllium and they are retracted into insulated compartments.

(U) Control Surfaces - Like the Mach 12 aircraft the control surfaces are hot structure and are of sheet metal construction as shown in Figure 5-11. The 1300°F (978°K) to 1500°F (1090°K) temperature requires that a superalloy be used. Rene' 41 and TD NiCR have been selected since they represent the most efficient structural materials available in this temperature range.

(U) Leading Edges and Nose Cone - Leading edges are lightly loaded structure and are subject to temperatures near 1500°F (1090°K). The structure is constructed of Rene' 41 formed and stiffened sheet that is segmented to reduce thermal stresses. They are made easily removable since this location is excellent for performing flight evaluations of various high temperature alloys and coatings.

(U) The nose cone is an X-15 type Q-Ball which is used as an attitude sensor as well as a thermal shield. Rene' 41 superalloy is the leading material candidate for the nose cone because of its temperature capability, strength and corrosion resistance.

5.2.2.2 (U) Structural Modifications (Research Options) - The basic aircraft structure provides the capability to adapt vehicle changes for additional flight research. Modifications to the structure for selected research options are described in the following paragraphs.

(U) Armament (ARM) Option - Modifications to the structure of the basic vehicle to accommodate an armament research capability are required on the lower surface. The structure of the armament bay consists of titanium material protected on the external surface by a passive thermal protection system (shingles and insulation). The bomb bay doors are guided and actuated with superalloy tracks and mechanisms. Structure to provide additional strength for the main supports will be required in the bulkheads at the attach points. Longitudinal intercostals must be added to transmit fore and aft loads into the fuselage.

(U) Thermal Protection System (TPS) Option - There is no change required in the primary structure to provide for research on various thermal protection systems. The modification requires minor changes for attachment of the TPS and instrumentation in a manner similar to that employed for the Mach 12 vehicle.

(U) Advanced Engine (JZ6) Option - Incorporating the GE5/JZ6 TRJ into the basic airplane in place of the STRJ11A-27 engine requires a significant modification. All structure aft of the rear LH_2 bulkhead with the exception of the vertical tails and the movable tip controls will have to be revised. This is primarily due to the change in size and shape of the engine and inlet together with the change in magnitude and location of the concentrated loads. It is possible to incorporate this option without modifying the structure forward of the aft LH_2 bulkhead. However, should the airframe load distribution be changed significantly, it is possible that a reduction in the maneuvering load factor capability would be required to avoid overloading the unmodified section of the fuselage structure.

(U) Structural concepts for the GE5/JZ6 would be the same as those for the STRJ11A-27 engine, inlet, and supports. A redesign of the regenerative cooling system may be required because of the increased surface area of the lengthened inlet.

(U) Convertible Scramjet (CSJ) Option - Adding the CSJ to the basic vehicle requires reshaping the lower surface. Changes in the primary structure would be local for attachment only. The added surface consists of support frames and shingles similar to those removed from the original moldline.

(U) The CSJ engine module is similar to that designed for the Mach 12 vehicle and it is supported in the same manner. Local bulkheads will require extensive modification to accommodate the hardpoint loads from the movable ramps and engine module. Intercostals for longitudinal load distribution must also be provided locally. The majority of the thrust loading will be transmitted through these into the fuselage structure.

(U) The critical design pressure condition, 5.0 g maneuver at Mach 6, for the CSJ engine is similar to that of hhe Mach 12 aircraft. Therefore, the design pressures are considered the same as shown for the Mach 12 vehicle in Figure 4-25. Internal temperatures in the scramjet module and on the aft movable ramp is made of titanium structure and superalloy shingles.

(U) The additional fuselage depth requires that both the nose and main landing gear be lengthened. This requires, in addition to longer gear, a strenthened landing gear bulkhead.

(U) Ramjet (RJ) Option - Addition of a ramjet module on the lower surface of the fuselage requires that the primary structure be strengthened to transmit the thrust load into the fuselage and to support the boundary layer diverter and engine inertial loads. A major revision of the lower moldline is required to provide a boundary layer diverter (BLD). Shingled and insulated structural frames are employed. Local areas in the BLD require columbium shingles, since the temperatures will approach stagnation values.

5.3 AERODYNAMICS

(U) The aerodynamic characteristics for the basic Mach 6 vehicle and several configuration options are presented in the following sections. Aerodynamic efforts were directed toward accomplishing the same Phase III objectives for the Mach 6 aircraft, as for the Mach 12 vehicle, namely:

o Substantiation of lift and drag values employed in vehicle performance analysis

o Determination of the basic vehicle static stability and control characteristics

o Analysis of the unaugmented handling qualities of the basic vehicle

(U) The lift and drag characteristics employed in determining the performance data presented in Section 5.5 (Phase II methods) are substantiated by two additional prediction techniques (Gentry and Harris Programs). Static stability and control characteristics for the basic vehicle are presented for both longitudinal and lateral-directional modes throughout the Mach regime. They show the aircraft to be statically stable and aerodynamically controllable in all modes. A redundant three-axis control augmentation system will of course be required to tailor handling qualities to the levels desired. However, the unaugmented handling qualities for the bare airframe are sufficiently good in most instances to assure a safe termination of the mission in spite of an inoperative augmentation system.

5.3.1 (U) LIFT AND DRAG CHARACTERISTICS - The lift and drag coefficients employed in the performance analyses of the Mach 6 turboramjet vehicles are defined as

$$C_L = C_{L\alpha}$$

$$C_D = C_{D_o} + L'C_L^2$$

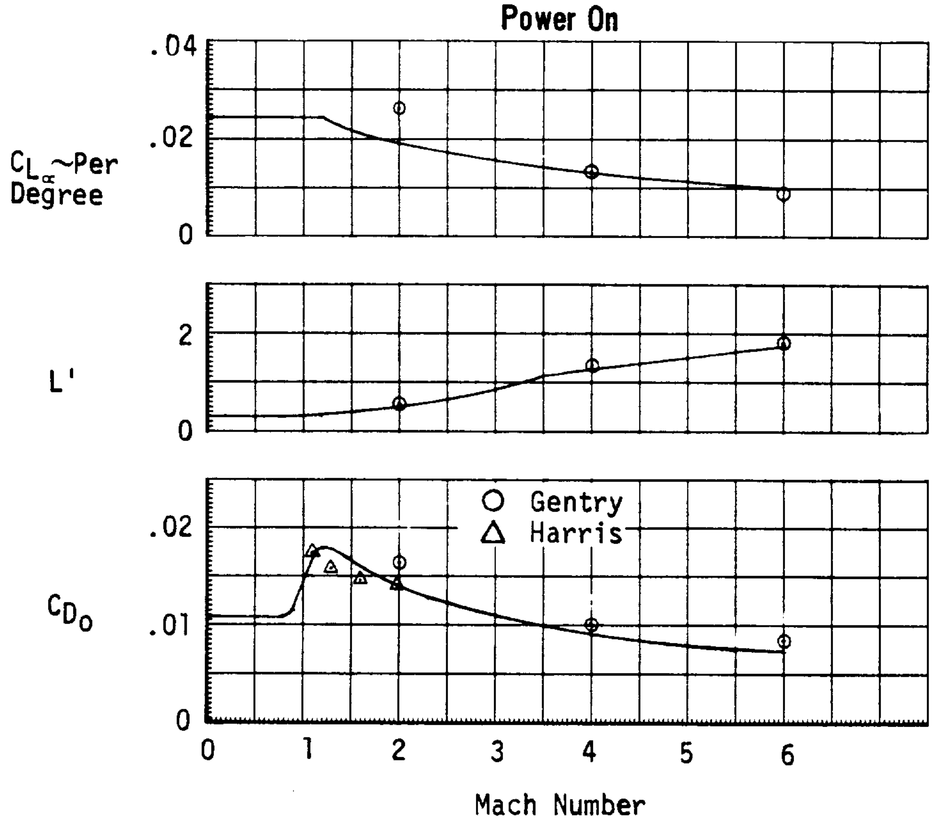
where the lift curve slope (C_{L_α}), the zero-lift drag coefficient (CD_o), and the induced drag factor (L') are determined as described in Section 3.2.I.

5.3.1.1 (U) Basic Vehicle - The values of L' and $C_{L\alpha}$ utilized in the basic vehicle performance analysis (Phase II methods) are shown in Figure 5-12, together with the values obtained from the Gentry Arbitrary Body Program. The values of L' and C_{L_α} employed in determining basic vehicle performance are in good agreement with the Gentry predicted values.

(U) The values of C_{D_o} employed in the performance analysis of the basic vehicle are also shown in Figure 5-12 together with the results obtained from the Gentry and Harris Computer programs. The Harris C_{D_o} values shown include the same skin friction, base, and vertical tail toe-in drag contributions as employed in the basic Phase II method of analysis since only wave drag is computed in the program. No aft body flow separation is assumed for this configuration and the complete fuselage geometry is therefore input to the Harris program.

(U) The C_{D_o} values shown for the Gentry program include the same base and protuberance drag contributions as employed in the basic Phase II method of

(U) FIGURE 5-12
MACH 6 TURBORAMJET
LIFT AND DRAG CHARACTERISTICS
Power On



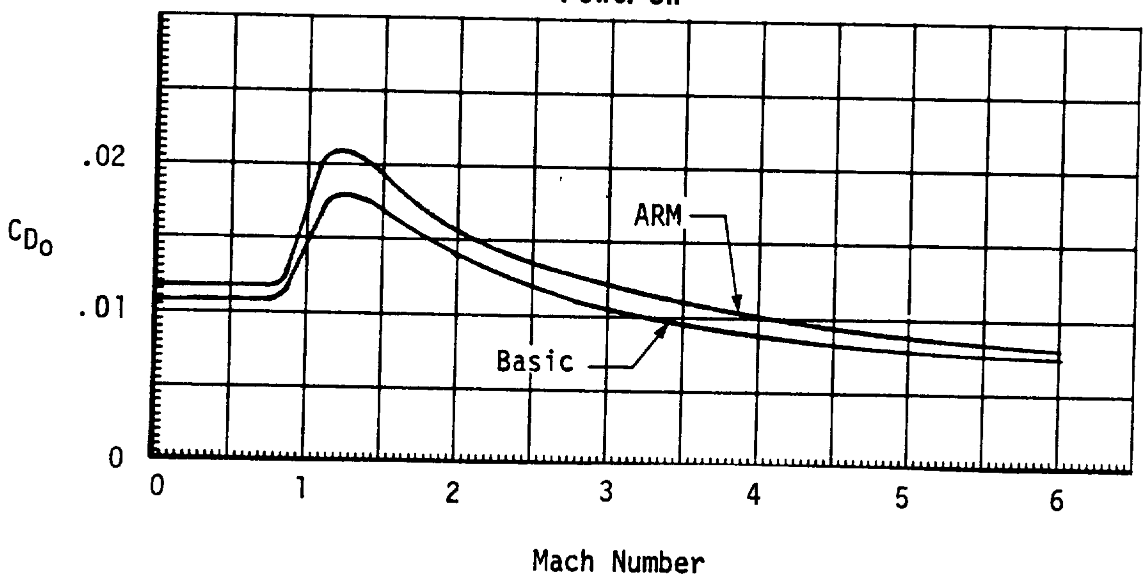
analysis since these are unaccounted for in the program computations. Although there are obvious differences in the three methods of computing C_{D_0}, the results indicate that the drag method employed in the basic vehicle performance analysis yields estimates substantially in keeping with other accepted prediction techniques.

(U) Like the Mach 12 rocket vehicles, a symmetrical outboard deflection of rudder panels is employed for speed brakes on the Mach 6 configuration. The speed brake effectiveness employed in the performance analysis of the Mach 6 vehicles is the same as that shown for the Mach 12 vehicles in Figure 4-28.

5.3.1.2 (U) Configuration Options - Estimated values of C_{D_0} obtained for the armament option to the basic vehicle (Figure 5-5) are presented in Figure 5-13. These values were employed in determining the variation in vehicle performance to be expected when incorporating this configuration option. The values of L' and C_{L_α} are assumed to be unchanged and are therefore the same as shown in Figure 5-12.

(U) A second option requiring an external configuration change is the use of a centerline LH_2 fuel tank (see Figure 5-8). This store is employed only at subsonic speeds and the additional drag is estimated as ΔC_{D_0} = .0015.

(U) FIGURE 5-13
MACH 6 VEHICLE ARMAMENT OPTION
ZERO-LIFT DRAG COEFFICIENT
Power On

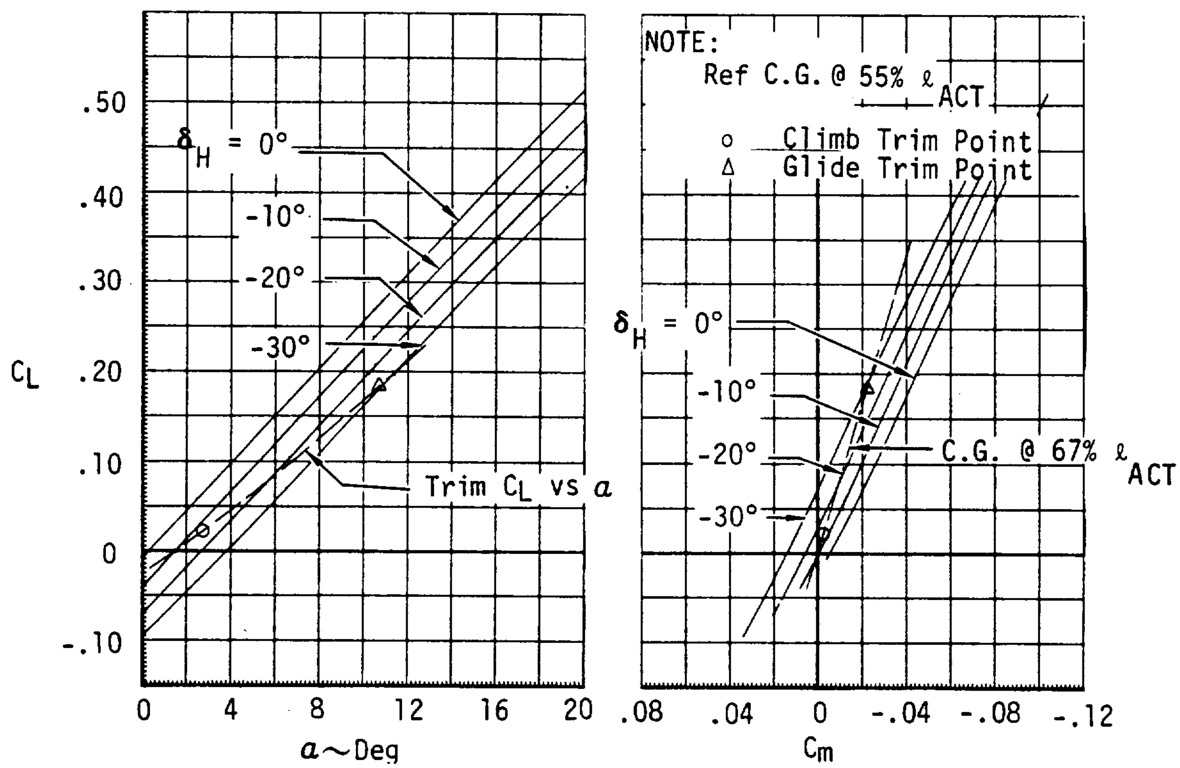


5.3.2 (U) STABILITY AND CONTROL CHARACTERISTICS - The longitudinal static stability characteristics for the basic vehicle are presented in Figures 5-14 and 5-15. The reference moment center employed is located at approximately 55% of the body length. Also shown on these plots are selected climb and glide trim points corresponding to an anticipated aft center of gravity location of 67% length.

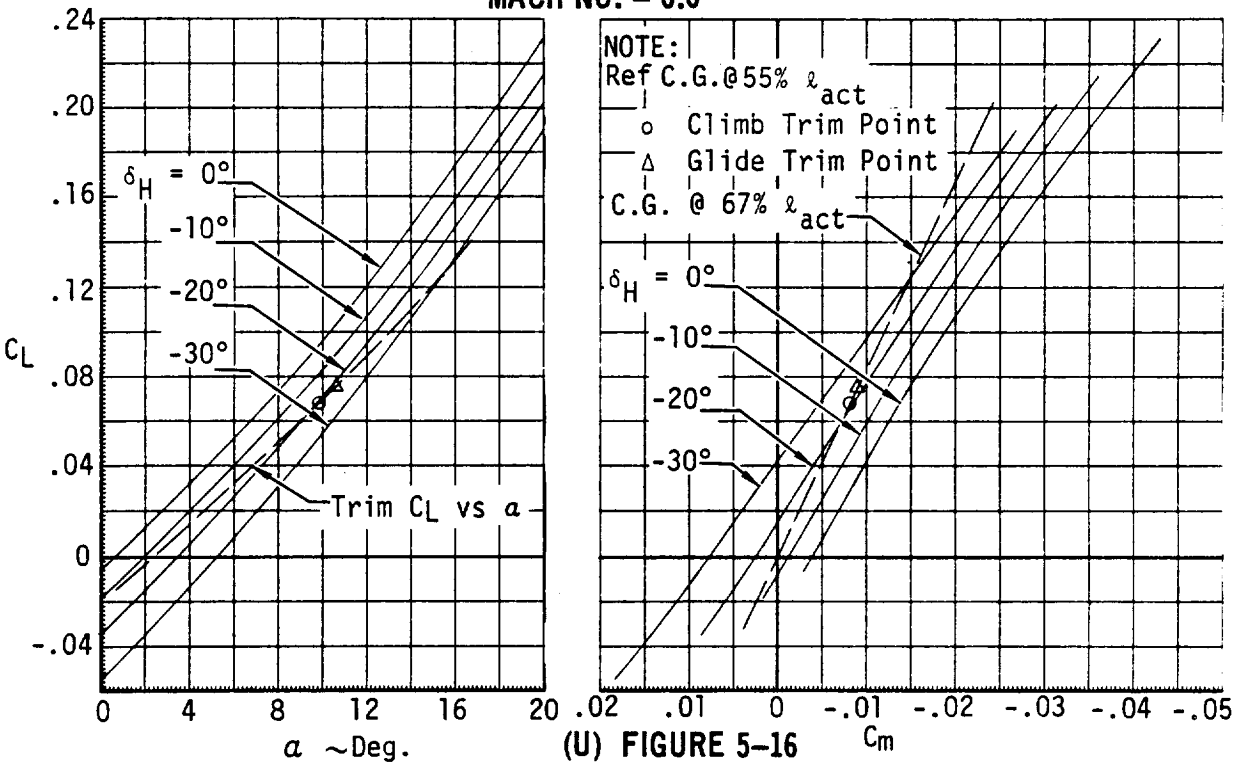
(U) The delta tip controls are shown to be quite effective in providing longitudinal control at all Mach numbers. However, as presently configured, the vehicle exhibits a nose down zero-lift pitching moment coefficient (C_{m_0}) which requires large control deflections to trim out. This in turn results in significant trim drag as shown in Figure 5-16. The nose down C_{m_0} is produced by the nose droop incorporated in the present design to improve the pilot's over-the-nose view during landing. The effect of removing this nose droop insofar as trim drag is concerned is illustrated in Figure 5-16. Another means of reducing trim drag, as illustrated in Figure 5-16, is to reduce static margin. This can be accomplished through cg control or by adding variable controls forward of the cg, e.g. canards, to reduce the rather significant high speed static stability margin shown in Figure 5-17.

(U) Figure 5-18 presents a tabulation of the pertinent longitudinal stability derivatives corresponding to the trim points shown in Figures 5-14 and 5-15. They were obtained from the Gentry analysis using a body axis system. They are employed in the handling qualities analysis presented in Section 5.3.3.

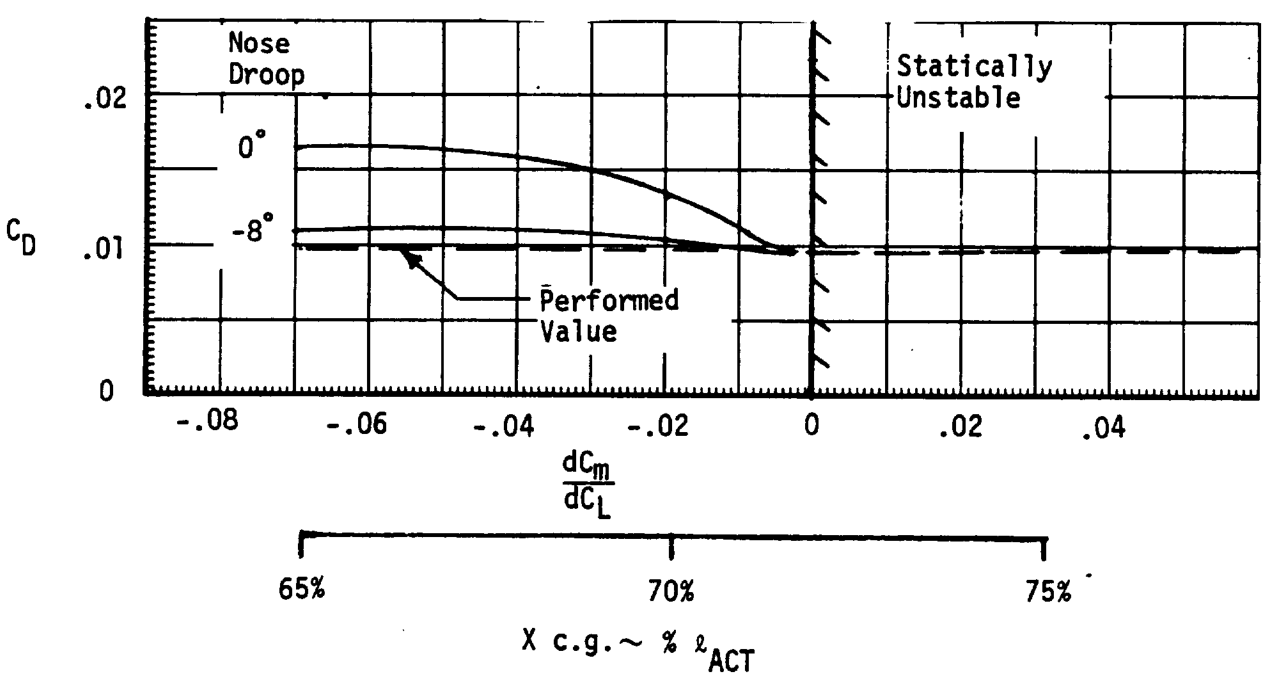
(U) FIGURE 5-14
MACH 6 TURBORAMJET
LONGITUDINAL STATIC STABILITY AND CONTROL CHARACTERISTICS
MACH NO.=2.0



(U) FIGURE 5-15
MACH 6 TURBORAMJET
LONGITUDINAL STATIC STABILITY AND CONTROL CHARACTERISTICS
MACH NO. = 6.0

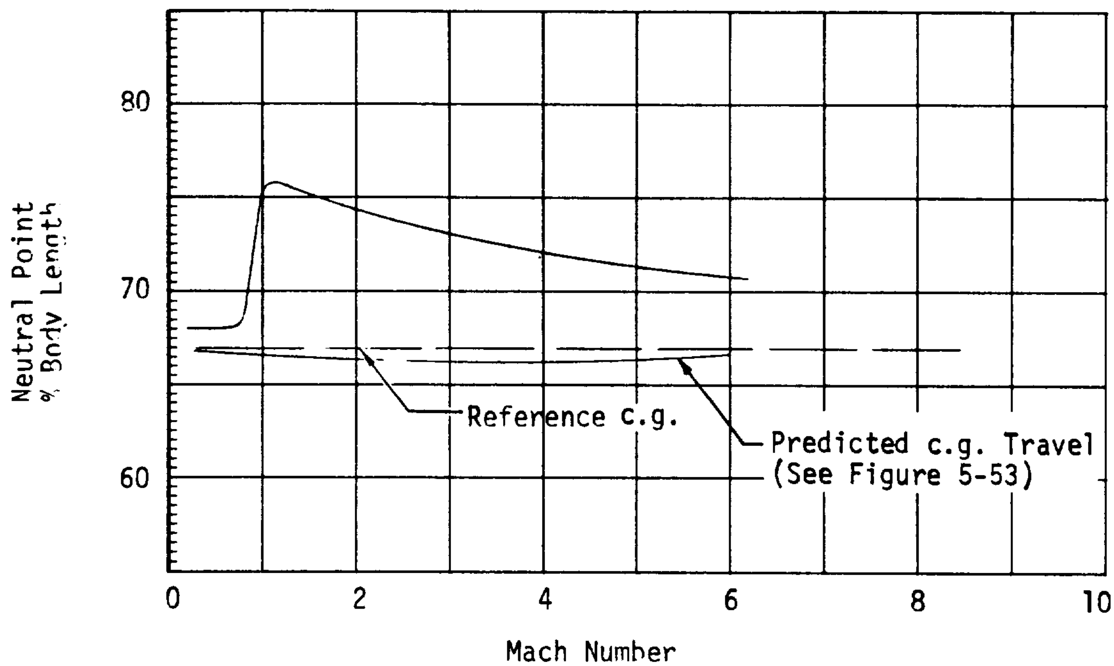


(U) FIGURE 5-16
MACH 6 TURBORAMJET
TRIMMED DRAG COEFFICIENT vs STATIC MARGIN
CLIMB AT MACH 4.0

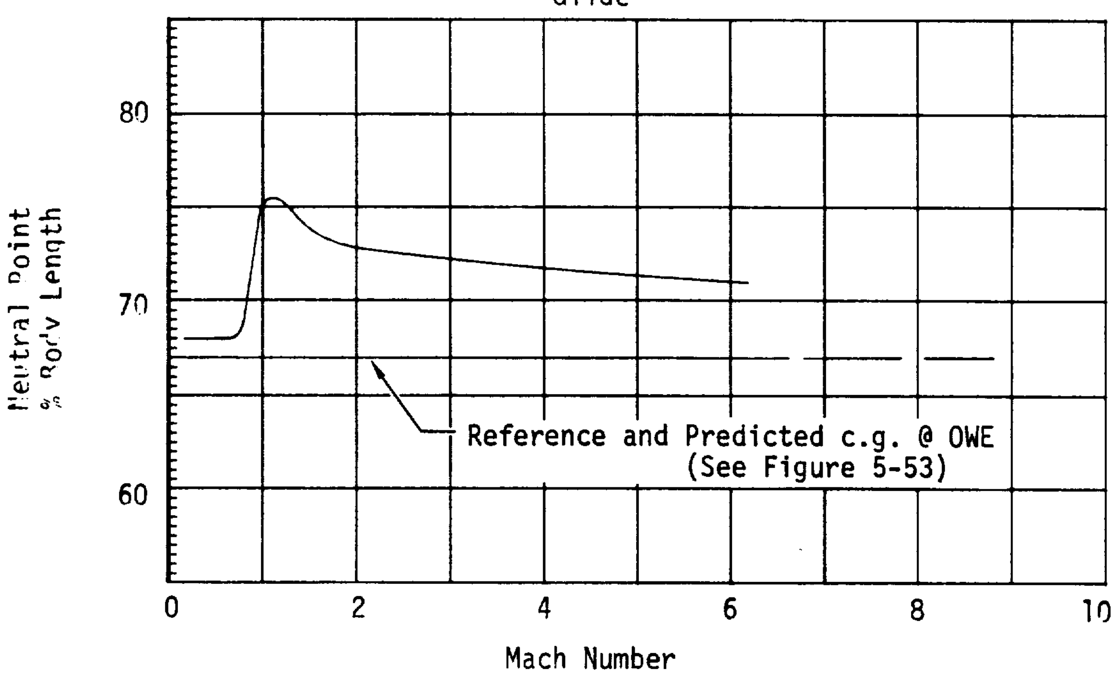


(U) FIGURE 5-17
MACH 6 TURBORAMJET
NEUTRAL POINT vs MACH NUMBER
TRIM FLIGHT CONDITIONS

Climb



Glide

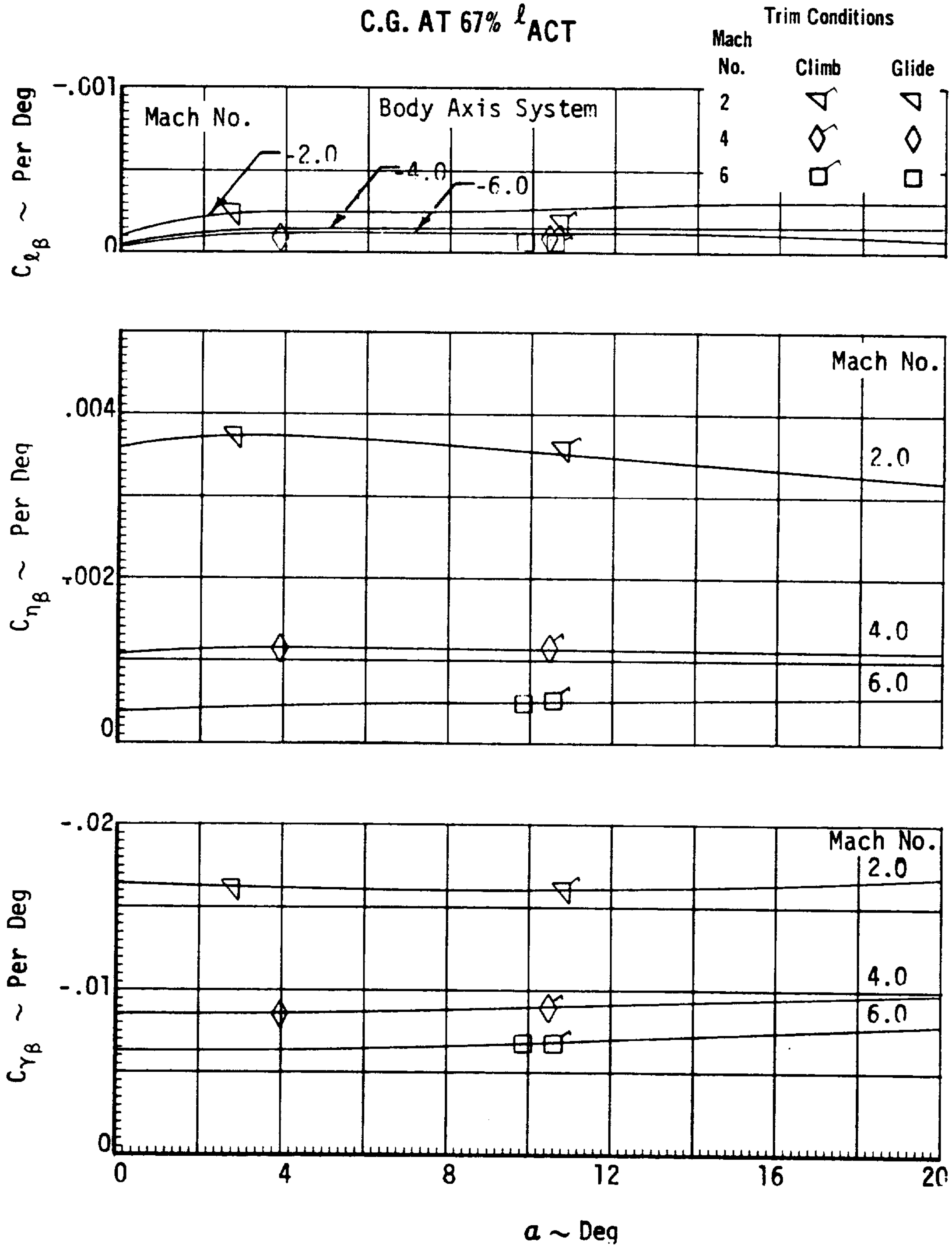


(U) FIGURE 5-18
MACH 6 TURBORAMJET
LONGITUDINAL DERIVATIVES AND
TRIM FLIGHT CONDITIONS

	Climb			Glide		
Mach No.	2.0	4.0	6.0	2.0	4.0	6.0
Pitch Der.						
C_A /Deg.	-.00017	-.00012	-.00002	-.00016	-.00006	-.00001
C_{L_α}/Deg.	.02553	.01366	.01182	.02606	.01512	.01209
C_{N_α}/Deg.	.02601	.01396	.01245	.02814	.01611	.01281
C_{m_α}/Deg.	-.00186	-.00068	-.00044	-.00151	-.00071	-.00048
C_{m_q}/Rad.	-.11934	-.07505	-.07019	-.15211	-.09011	-.07210
C_{A_q}/Rad.	.01370	.00863	.00265	.00912	.00354	.00200
C_{N_q}/Rad.	.20159	.08271	.05223	.18782	.08581	.05676

(U) Figure 5-19 shows the lateral-directional static stability characteristics of the basic vehicle for a stabilator deflection of 0° (untrimmed). The symbols indicate the selected climb and glide trim points previously discussed. These data indicate that the vehicle is statically stable directionally at all Mach numbers. Figure 5-20 presents a tabulation of the pertinent lateral-directional stability derivatives for the selected trim points which are employed in the handling qualities analysis. These were obtained from the Gentry program.

(U) FIGURE 5-19
MACH 6 TURBORAMJET
LATERAL DIRECTIONAL STATIC STABILITY
Horizontal Stabilator Deflection = 0
C.G. AT 67% ℓ_{ACT}



(U) FIGURE 5-20
MACH 6 TURBORAMJET
LATERAL-DIRECTIONAL DERIVATIVES
AT TRIM FLIGHT CONDITIONS

	Climb			Glide		
Mach No.	2.0	4.0	6.0	2.0	4.0	6.0
C_{y_r}/RAD	.22085	.07206	.03219	.21496	.07159	.03302
C_{n_r}/RAD	-.57581	-.35008	-.29180	-.58246	-.36026	-.29311
C_{l_r}/RAD	.03351	.01681	.01394	.03314	.01762	.01357
$C_{y_{\delta_r}}$/RAD	.0854	.0575	.0523	.0763	.0516	.0480
$C_{n_{\delta_r}}$/RAD	-.0574	-.0387	-.0352	-.0513	-.0347	-.0375
$C_{l_{\delta_a}}$/RAD	.0721	.0403	.0325	.0746	.0449	.0338
C_{l_p}/RAD	-.1083	-.0582	-.0479	-.1134	-.0658	-.0495

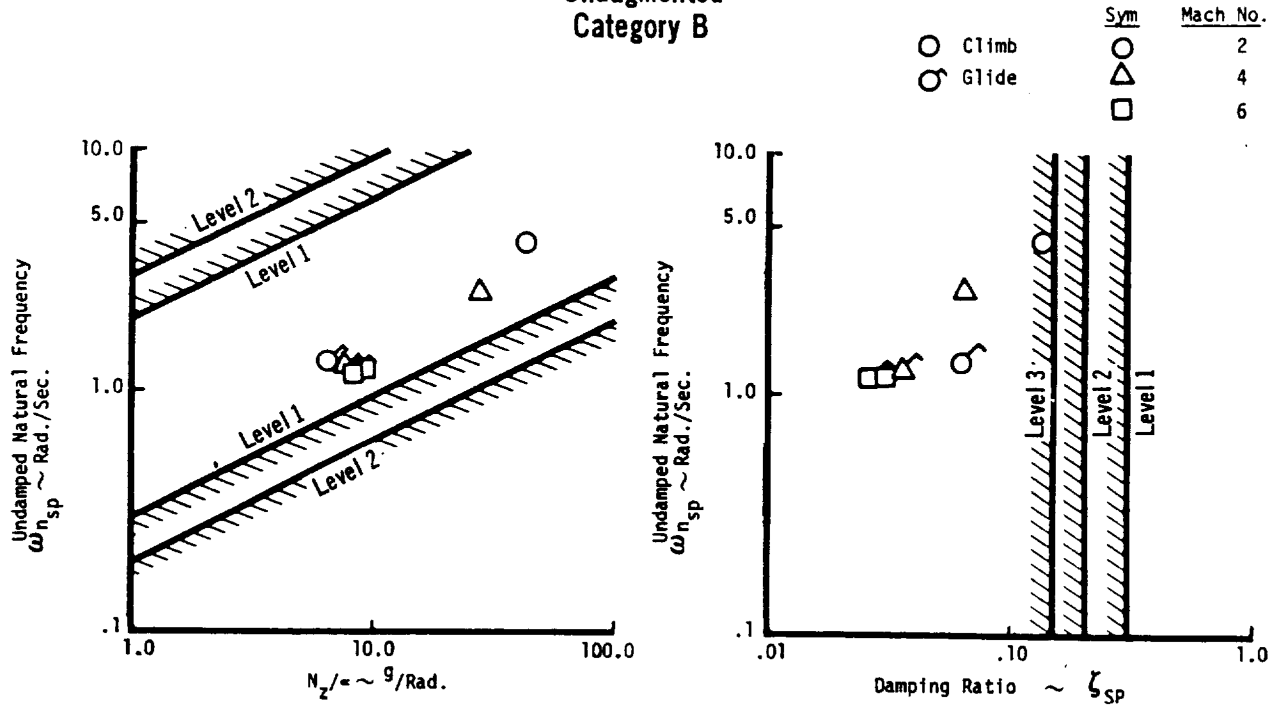
5.3.3 (U) HANDLING QUALITIES - The unaugmented handling qualities of the basic Mach 6 vehicle were examined in a manner similar to that described for the Mach 12 vehicle in Section 4.3.3. The trim flight conditions investigated are those shown in the preceding section. Comparisons of the results attained are again made with the Level 2 and Level 3, Category B and C, Class IV handling qualities requirements of Reference 6. These requirements, although not directly applicable to a research aircraft, provide a frame of reference. The bare aircraft flying qualities shown are representative of an inoperative augmentation system.

(U) Figure 5-21 presents the longitudinal short period dynamic stability characteristics of the unaugmented vehicle. This plot indicates that the undamped short period natural frequency, $\omega_{n_{sp}}$, versus normal load factor per unit angle of attack, Nz/α, is well within limits throughout the Mach range of the vehicle. As with the Mach 12 vehicle, however, the short period damping ratio, ζsp, is less than Level 3 in all cases and the vehicle will require artificial damping in pitch as would be expected. The strong effect of density (altitude) on ζsp is apparent in Figure 5-21 when the differences in climb and glide altitude are compared. (See Figure 2-3.)

(U) Figures 5-22 and 5-23 show the lateral-directional characteristics of the unaugmented vehicle. The Dutch roll characteristics shown in Figure 5-22 are generally quite good. The plot of undamped natural frequency, ωnd, versus the damping ratio, ζd, indicates that Level 2 requirements are met by all but the Mach 6 points. This is also true for the criteria relating the product of frequency and damping to roll-sideslip coupling, ϕ/β.

(U) The roll-sideslip coupling characteristics produced by aileron rolls are examined in Figures 5-23. The sideslip excursions, $\Delta\beta_{max}$, are well within the Level 2 requirement. The roll-rate oscillations, P_{osc}/P_{av}, for this vehicle are negligible and, therefore, meet all requirements.

(U) FIGURE 5-21
MACH 6 TURBORAMJET
LONGITUDINAL SHORT PERIOD DYNAMIC STABILITY
Unaugmented
Category B

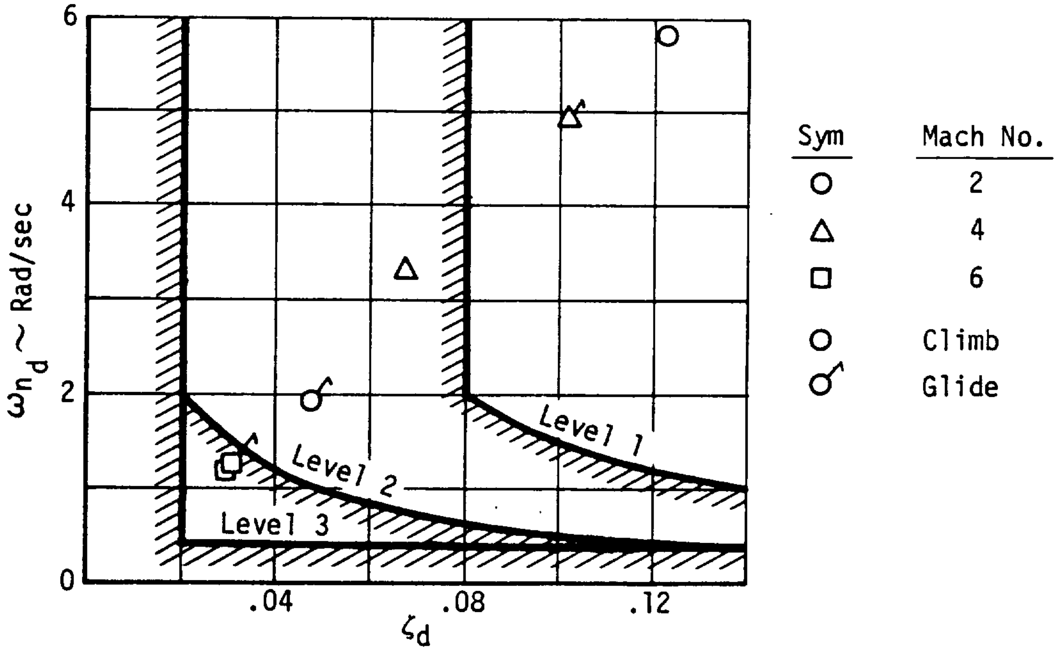


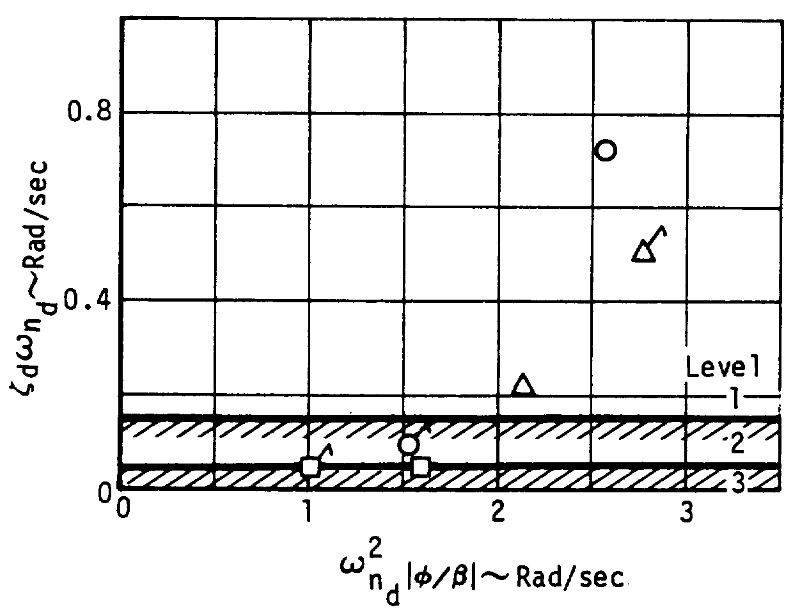
5.3.4 (U) TAKEOFF AND LANDING CHARACTERISTICS - The low speed longitudinal static stability and control characteristics are presented in Figure 5-24. It is evident that the longitudinal control effectiveness is sufficient to trim the aircraft at an angle of attack of 15°, the maximum value anticipated to be required for landing.

(U) The longitudinal short period dynamic stability characteristics are presented in Figure 5-25. The low speed longitudinal handling qualities for the unaugmented airframe are indicated to be quite acceptable.

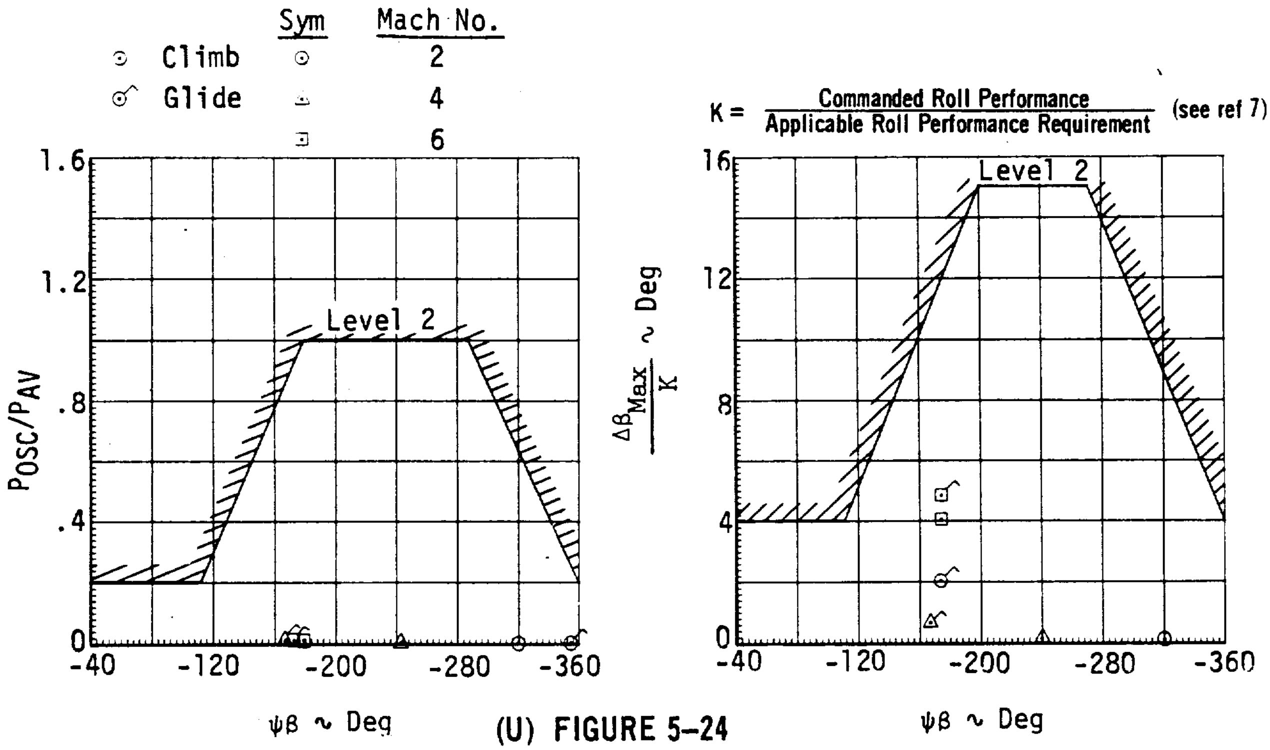
(U) In Figure 5-26, the low speed lateral control characteristics are presented. The Mach 6 aircraft, like the Mach 12 vehicle, employs differential deflections of the delta tip controls to provide required rolling moments. The deflection authority must be shared with the pitch control mode and this sharing is most critical at low speeds. Two differential deflections, 10° and 20°, are shown in Figure 5-26 and it is apparent that ample lateral control effectiveness is available. With the differential control deflections shown, the collective deflections available for longitudinal control are -20° and -10° respectively. It is evident in Figure 5-25 that with proper sharing of control mode authority sufficient longitudinal control, as well as lateral control, can be provided.

(U) FIGURE 5-22
MACH 6 TURBORAMJET
DUTCH ROLL CHARACTERISTICS
UNAUGMENTED
CATEGORY B

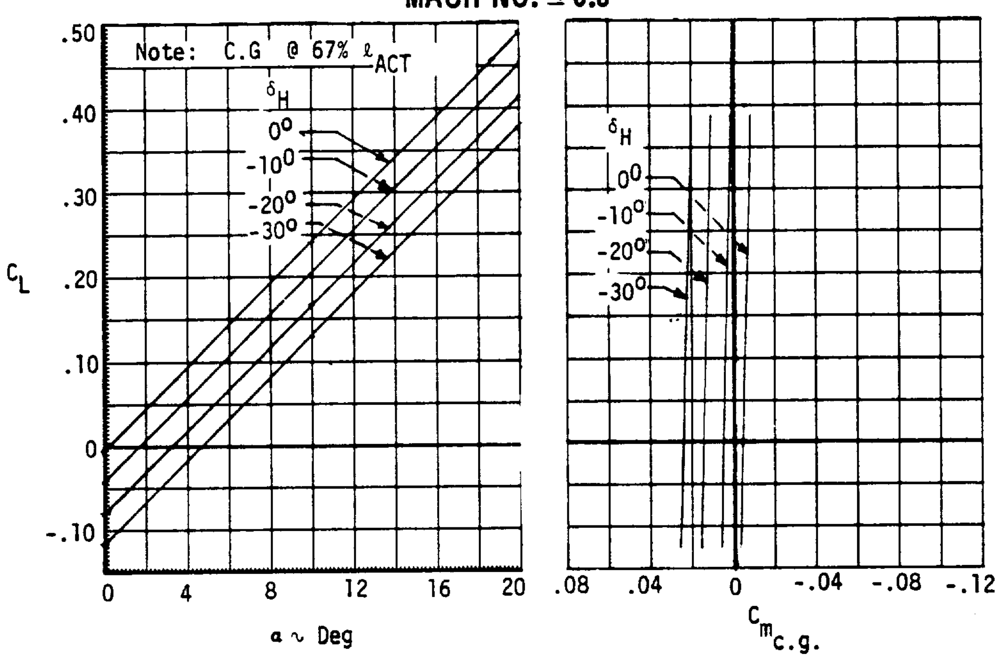




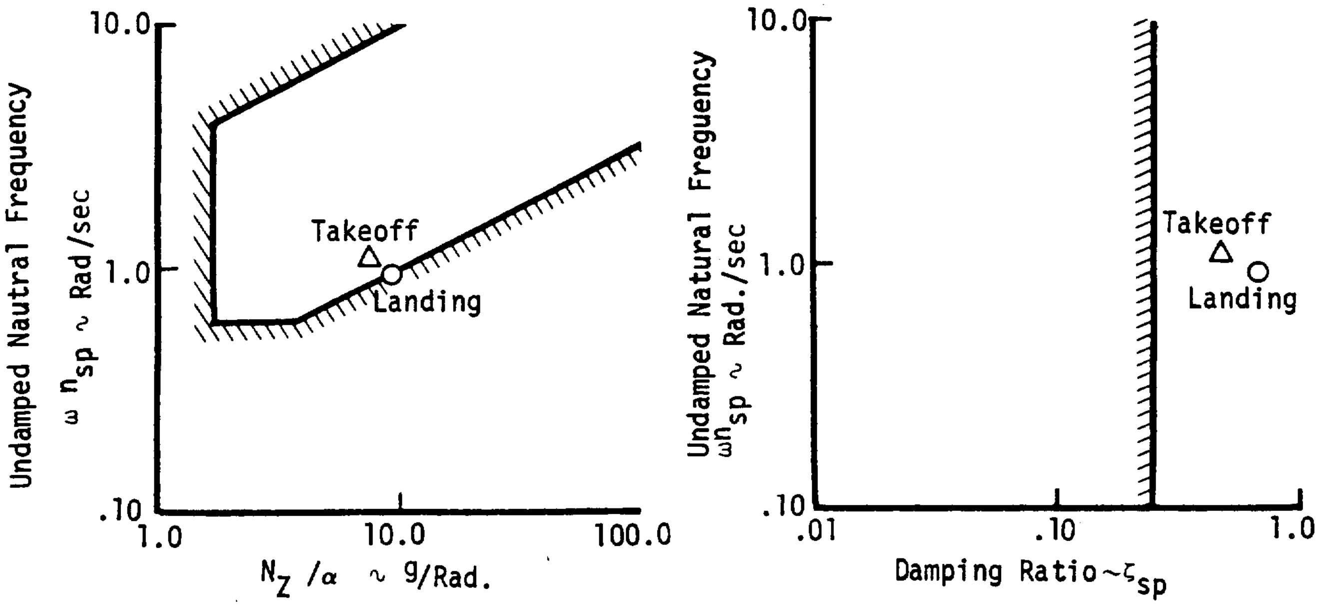
(U) FIGURE 5-23
MACH 6 TURBORAMJET
ROLL – SIDESLIP COUPLING
Unaugmented Category B



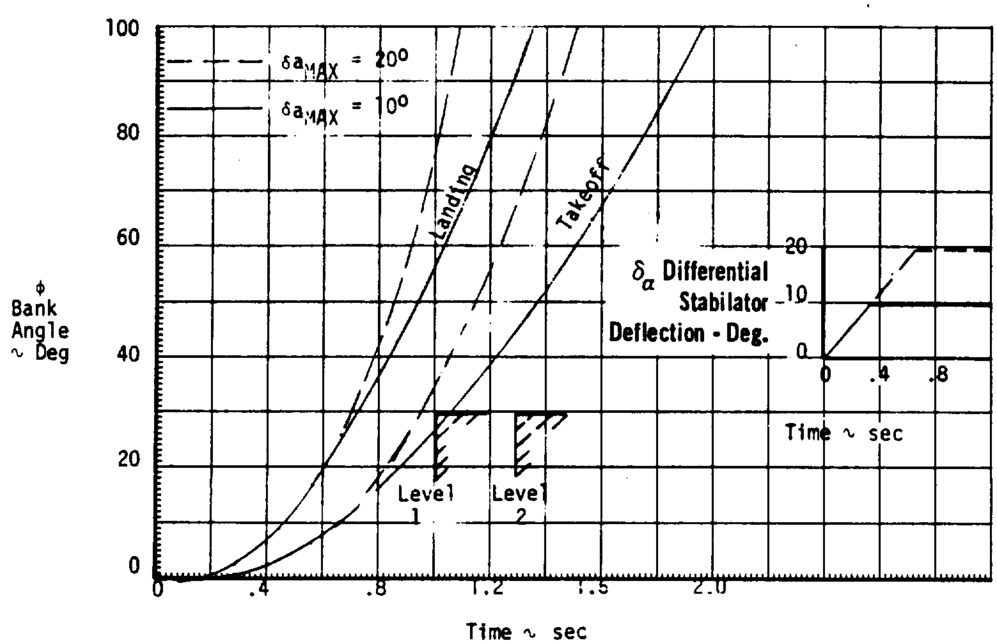
(U) FIGURE 5-24
MACH 6 TURBORAMJET
LONGITUDINAL STATIC STABILITY AND CONTROL CHARACTERISTICS
MACH NO. ≤ 0.8



(U) FIGURE 5-25
MACH 6 TURBORAMJET
LONGITUDINAL SHORT PERIOD DYNAMIC STABILITY
UNAUGMENTED AIRFRAME
CLASS IV – CATEGORY C – LEVEL 2
MACH NO=0.4



(U) FIGURE 5-26
MACH 6 TURBORAMJET
BANK ANGLE vs TIME
TAKEOFF AND LANDING



5.4 PROPULSION

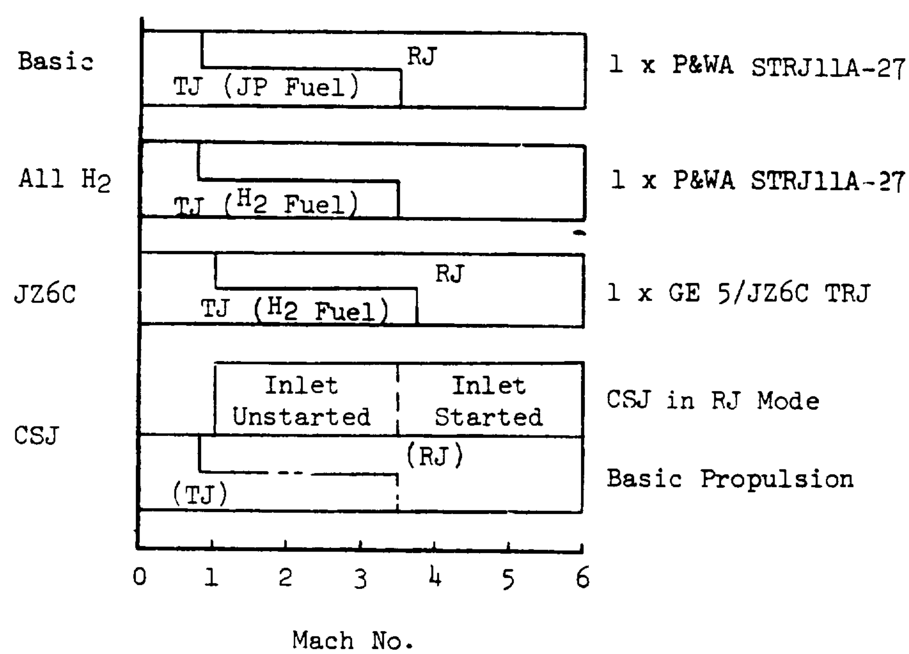
(U) The Phase III propulsion efforts, with regard to the Mach 6 vehicle, were directed to the following areas:

- Engine selection
- Definition of inlet design and operating characteristics
- Inlet-engine matching
- Definition of engine operating characteristics
- Determination of installed engine performance
- Analysis of propulsion research options.

In the following paragraphs, the propulsion system of the basic vehicle is discussed, followed by a description of the propulsion system research options added to the basic vehicle. Figure 5-27 summarizes the Mach number capabilities of the various propulsion systems.

5.4.1 (U) BASIC VEHICLE - The basic Mach 6 vehicle uses a single P&WA STRJ11A-27 wraparound turboramjet engine installed at the rear of the vehicle. Air is supplied by a bifurcated inlet arrangement shown in Figure 5-3. The TJ and RJ engine subsystems are operated simultaneously at speeds above the RJ minimum operating limit, and below the TJ maximum operating limit, to minimize acceleration time. The synthesis of the turboramjet and inlet, and the matching of these components for the propulsion system of the Mach 6 vehicle, are described in the following sections.

(U) FIGURE 5-27
PROPULSION SYSTEMS AND OPERATING REGIMES
Mach 6 Vehicle



5.4.1.1 (U) Engine Selection - In selecting the engine for this vehicle, two sequential tasks were involved. First, a choice was made between using an advanced technology TRJ engine, or an engine which would incorporate an existing TJ core and require developing only the RJ subsystem (near-term technology). Phase I results indicated that the selection of an advanced TRJ engine would necessitate large engine development costs and thereby increase the cost of the overall program significantly. At the same time, both Phase I and Phase II results point to the research value to be gained by flight experience with advanced airbreathing systems. To reduce the engine development costs and preclude pacing the aircraft development to that of an advanced engine, the near-term technology approach was selected.

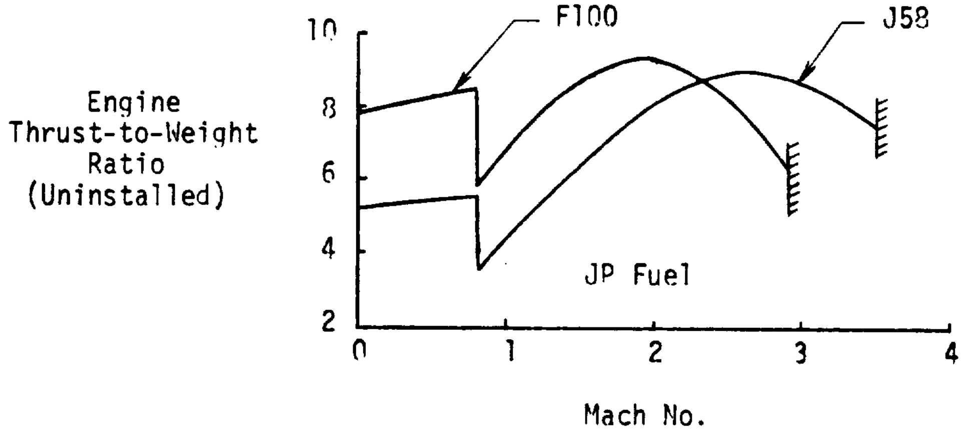
(U) Having made this decision, the second task involved selecting the TJ core to be employed from existing TJ engines. Five candidate TJ engines were considered: J58, F-100, GE 4/J5P, J93, and J97. The selection was based on thrust-to-weight ratio, thrust, availability, and maximum Mach number capability. Figure 5-28 presents a comparison of these engines and an evaluation of their suitability. Three engines--the GE 4/J5P, J93, and J97--were eliminated due to the considerations noted in Figure 5-28. In choosing between the J58 and F-100, a comparison was made of the thrust-to-weight characteristics of the two engines across the applicable speed range, Figure 5-29. The J58 was selected as the core TJ for the basic Mach 6 based on the combination of a higher T/W at high supersonic speeds and a higher maximum Mach no. capability.

(C) FIGURE 5-28
NEAR-TERM TJ COMPARISON

ENGINE	USE	SLS THRUST klb (kN)	MAX. MACH	REMARKS	EVALUATION
J58	YF12, SR71	34 (150)	3.3+	Uses large ejector nozzle not suitable for TRJ.	Continued
F100	F15	23 (101)	2.7	Uses convergent-divergent nozzle.	Continued
GE 4/J5P	SST	67 (295)	3.0	Thrust large for research vehicle application.	Dropped, thrust
J93	B70	28 (123)	3.2	No engines or tooling available.	Dropped, availability
J97	(Classified)	7 (31)	2+	Capability above M = 2.2 not substantiated.	Dropped, Mach capability

(U) The resulting TRJ engine is designated the P&WA STRJ11A-27 and is comprised of a modified P&WA J58 (JT11) afterburning turbojet, with a new design wraparound ramjet. The TJ engine burns JP fuel. Engine modifications include using a fixed-geometry convergent nozzle and closure doors to seal off the TJ at flight speeds above Mach 3.5. The regeneratively cooled RJ burns LH_2 fuel and uses a variable geometry convergent-divergent nozzle. The RJ size as defined in the engine specification was used for the propulsion system of the Mach 6 vehicle, and resulted in satisfactory performance without need for scaling.

(C) FIGURE 5-29
TJ THRUST-TO-WEIGHT COMPARISON FAVORS J58
Along Flight Path of Figure 5-38

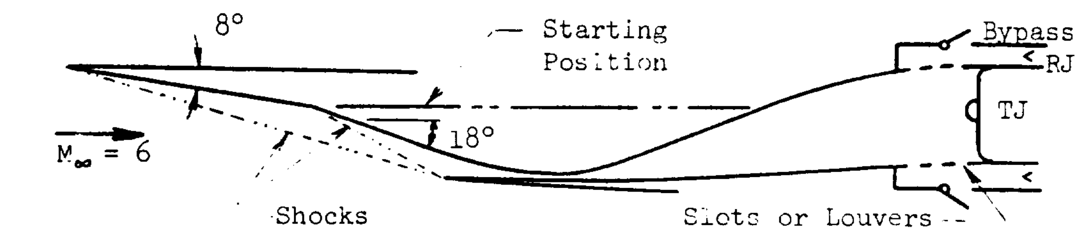


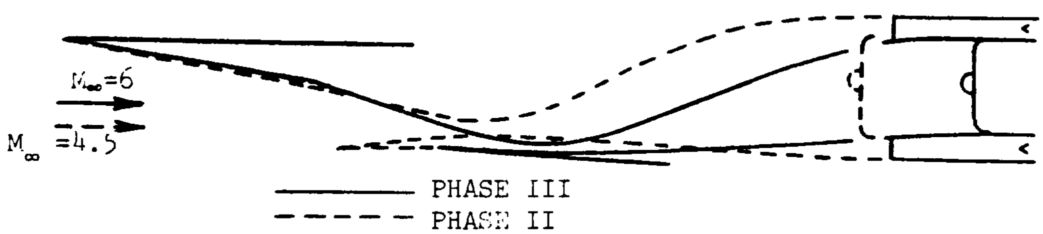
5.4.1.2 (U) Inlet Synthesis - A two-dimensional, mixed compression, inlet configuration was defined for the Mach 6 TRJ propulsion system. A bifurcated arrangement was chosen for efficient integration of inlet and airframe. The design point for inlet contours is Mach 6 at 0° angle of attack and freestream conditions as a result of the inlet installation selected for the Mach 6 vehicle. The inlet bowshock angle relative to the ramp surface has a minimum value within the anticipated range of angles of attack. By picking the design point at this minimum ($\alpha = 0$) ingestion of the inlet bowshock is prevented throughout the range of angles of attack. The inlet has two horizontal external-compression ramps. Design ramp angles were selected to provide equal total pressure drop across the two external oblique shocks in order to maximize the design point pressure recovery of the external supersonic diffuser, per Reference (19).

(U) Geometry of the Mach 6 TRJ inlet was based on several two-dimensional mixed compression inlets, with design Mach numbers up to 5.0, obtained from References (20) and (21). From these references, a data base was established for the Phase III inlet design. By extrapolating the trend of contraction ratio as a function of design Mach number, the design point external contraction ratio of 4.0 and the overall geometric contraction ratio of 22 were established for a Mach 6 inlet. A first ramp angle of 8° and second ramp angle of 10° satisfied the dual design point requirements of equal pressure drop plus specified contraction ratio. The second ramp is hinged at its leading edge and is varied from 0° to 10° relative to the fixed first ramp to provide the turning angle required to maintain high pressure recovery at off design conditions. The leading edge of the cowl was located at the design point intersection of the two oblique shock waves generated by the external ramps. An initial cowl turning angle of 6° was selected on the basis of the data shown in Reference (22) for low supersonic speed operation of a started, mixed-compression inlet. Sideplates were provided to reduce lateral spill.

(U) The overall length of the mixed compression supersonic diffuser is dependent on the design Mach number. On the basis of the Reference (22) data, a value of 4.75 was established for the ratio of the supersonic diffuser length to the inlet capture height on the Mach 6 inlet. For Phase III, an aspect ratio (height to width) of 2.0 was chosen on the basis of fuselage/inlet integration. The subsonic diffuser design reflects reasonable goals based on MCAIR inlet investigations. Figure 5-30 shows the Phase III Mach 6 inlet design in comparison with the Mach 4.5 inlet design employed in Phase II.

(U) FIGURE 5-30
PHASE III TURBORAMJET INLET

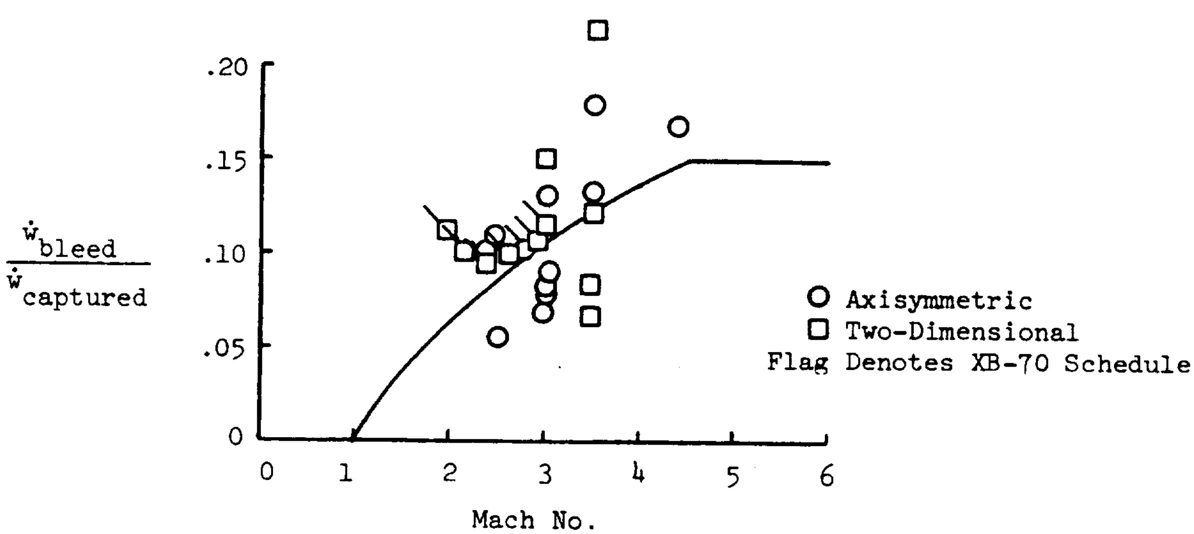




(U) The TRJ inlet boundary layer bleed airflow schedule used in Phase III was based on a collection of experimental data, Figure 5-31. Since data were not available beyond Mach 4.5, the bleed airflow used in Phase III was assumed constant from Mach 4.5 to 6.

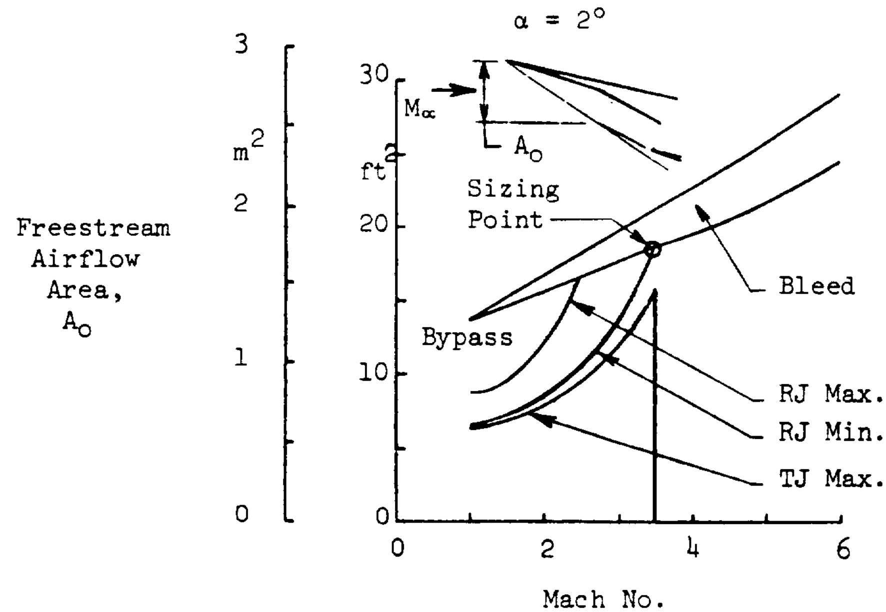
5.4.1.3 (U) Inlet/Engine Matching - This matching was accomplished to define the minimum inlet size that would be compatible with the engine airflow requirements. The turbojet core engine airflow demand at maximum power was determined, as a function

(C) FIGURE 5-31
TRJ INLET BLEED SCHEDULE



of Mach number, along the acceleration/climb path for $\alpha = 2°$ (See Figure 5-38). In addition, the minimum airflow demand on the ramjet engine (at minimum burner Mach number of 0.05) was similarly determined. The sum of these two airflows represents the minimum value that the inlet must supply. The inlet capture area required to supply the net airflow, (captured airflow less ramp bleed airflow) needed to meet this minimum demand, was established as 25.6 ft^2 (2.38 m^2). With this size, the airflow supply exactly equals the minimum demand at Mach 3.5. At all other speeds the supply exceeds the minimum demand. The excess airflow is used in the ramjet to provide more thrust, until the RJ burner Mach number exceeds 0.50. At conditions where the inlet supply airflow exceeds the sum of ramp bleed plus TJ demand on maximum power plus RJ demand at burner Mach number of 0.50, the excess air is bypassed through a louvre system and ducted aft where it is dumped overboard in an axial direction. Bypass drag was assessed as described in Section 3.3. Figure 5-32 shows the apportionment of the airflow as a function of flight speed.

(U) FIGURE 5-32
INLET SIZING FOR P&WA STRJ11A-27



5.4.1.4 (U) <u>Engine Operation</u> - The turbojet core engine operates on JP fuel, through the speed range of Mach 0 to 3.5, Figure 5-27. At Mach 3.5 and above, the turbojet is shutdown, protected from the environment by closure doors, and windmilled with a small amount of inlet air which is cooled to 1000°F (810°K) in a hydrogen/air heat exchanger. The wraparound ramjet engine operates on LH_2 fuel through the speed range of Mach 0.8 to 6. For the basic flight profile, the TJ engine operates at maximum afterburning power throughout its speed range, with fuel-rich combustion in the primary burner and lean operation in the afterburner so that the overall amount of fuel consumed corresponds to the stoichiometric amount for the engine airflow. The RJ operates at stoichiometric conditions over the entire acceleration speed range. To regeneratively cool the engine and inlet at Mach 6 requires a stoichiometric fuel flow rate ($\phi = 1.0$). To match cruise thrust to drag, a portion of the fuel is used to operate the engine in a throttled condition while the remainder is used for cooling only, and dumped without burning, thus degrading effective specific impulse.

5.4.1.5 (U) Installed Performance - The installed propulsion system performance used in the basic vehicle is based on the P&WA STRJ11A-27 engine and the 25.6 ft^2 (2.38 m^2) capture area, Mach 6 inlet design discussed in the preceding section. The engine performance (Reference 23) was based on the use of hydrogen fuel in both the TJ and RJ subsystems. These data were adjusted to reflect the use of JP fuel in the TJ. The fuel specific impulse was adjusted by the ratio of the lower heating value of the fuel,

$$(I_{sp})_{JP} = (I_{sp})_{H_2} \times \frac{18{,}500 \text{ Btu/lb (for JP-air)}}{51{,}500 \text{ Btu/lb (for } H_2\text{-air)}} \left(\frac{4.29 \times 10^7 \text{ Joules/kg}}{11.95 \times 10^7 \text{ Joules/kg}}\right)$$

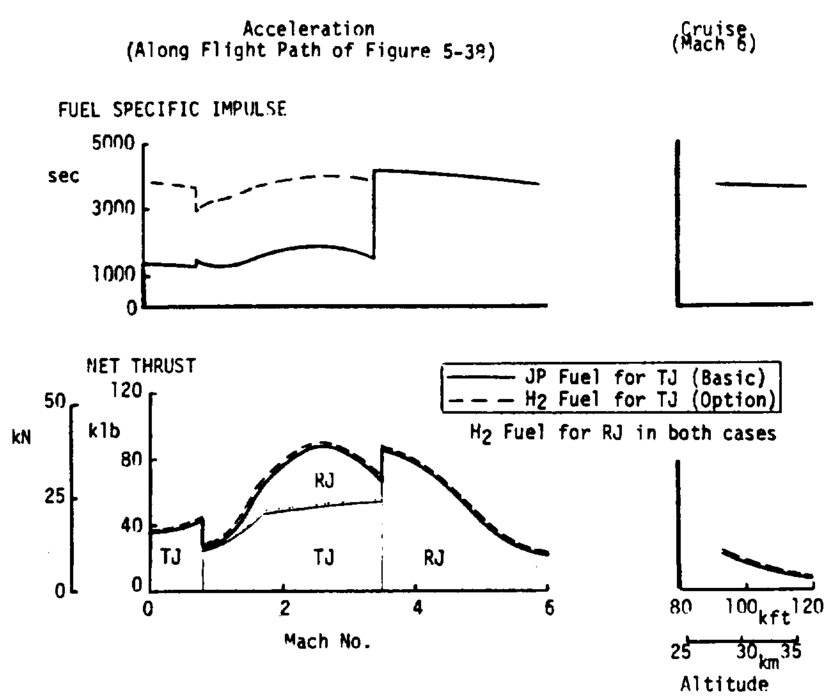
The thrust was adjusted by the ratio of the mass flows, and the square root of the ratio of the afterburner temperatures,

$$(F_N)_{JP} = (F_N)_{H_2} \times \frac{\dot{m}\sqrt{T} \text{ (for JP-air)}}{\dot{m}\sqrt{T} \text{ (for } H_2\text{-air)}} \approx .975 \times (F_N)_{H_2} \qquad (\text{at } \phi = 1.0)$$

(U) Engine thrust was reduced by the amount of inlet drag due to bleed, bypass, and supersonic spill. The net installed thrust and fuel consumption characteristics of the basic vehicle's propulsion system were calculated, Figure 5-33, based on the nominal inlet pressure recovery factors of Figure 3-6.

(C) FIGURE 5-33
P&WA STRJ11A-27 INSTALLED ENGINE PERFORMANCE

Stoichiometric (ϕ = 1)



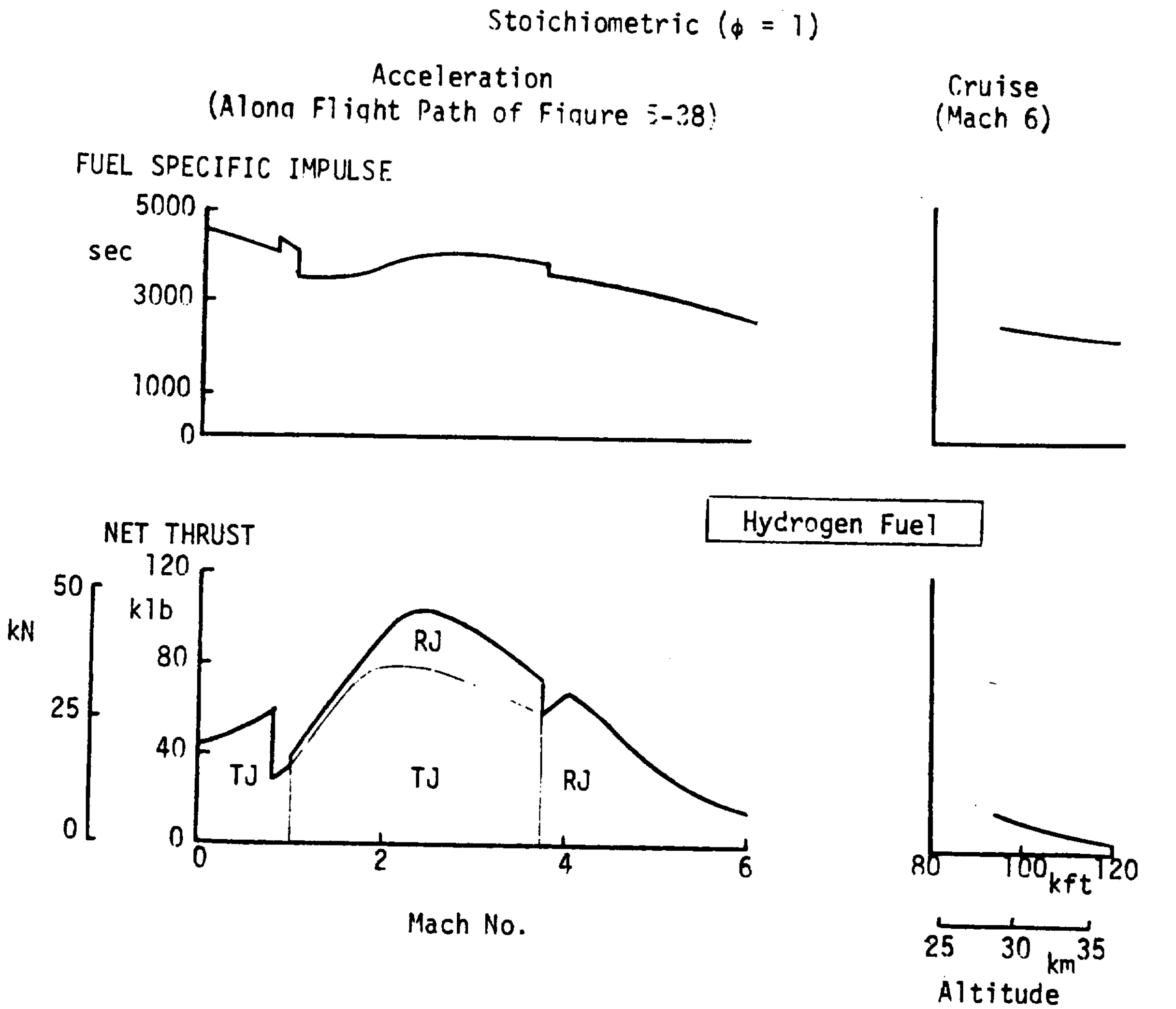
5.4.2 (U) CONFIGURATION OPTIONS - Three modifications to the basic vehicle propulsion system were investigated, to extend the research capability of the Mach 6 vehicle. All of these modifications retain a turboramjet as the primary propulsion system. One of the modifications maintains the basic vehicle propulsion system (STRJ11A-27), and adds a CSJ as an addition to the basic vehicle. The other two modifications replace the basic TRJ engine. One option incorporates an all-hydrogen-fueled version of the basic STRJ11A-27 TRJ. The other replaces the near-term TRJ with an advanced technology TRJ engine, the GE5/JZ6C, to permit research on the hydrogen-fueled TJ core of the advanced engine. Figure 5-34 summarizes the characteristics of the propulsion systems for these vehicle modifications, and Figures 5-35 and 5-36 show the propulsion system installation and installed engine performance for the modifications. Figures 5-7 and 5-9 depict the overall vehicle arrangements for these modifications.

(U) In addition to these propulsion modifications, two other configuration options were investigated. They were an armament installation and a thermal protection system variation. Neither of these require a change in the basic vehicle propulsion system.

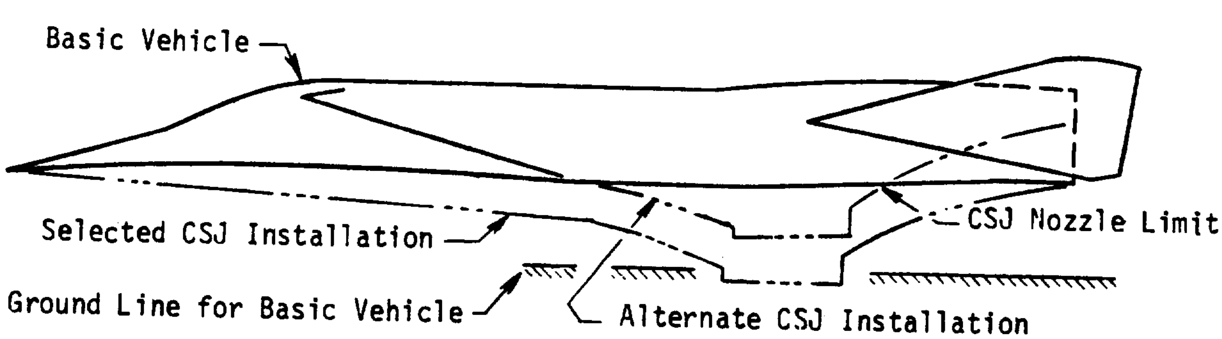
(U) FIGURE 5-34
PROPULSION MODIFICATIONS TO MACH 6 VEHICLE

Type of Modification	Status of Basic STRJ11A-27 System	Characteristics of Added Propulsion System			
		Operating Speed Range	Configuration Source	Installed Performance Presented	Remarks
All H_2	TJ fuel changed to H_2	Same as basic		Figure 5-33	
JZ6C	Replaced with new, advanced technology TRJ	Mach 0-6; TJ: Mach 0-3.75 RJ: Mach 1-6		Figure 5-35	H_2 fueled TJ
CSJ	Maintained, used to accel. to CSJ light-off, & for thrust supplement during CSJ operation	M=1-6	Same as for CSJ on Mach 12 vehicle; installation per Fig. 5-36	Same as for CSJ on Mach 12 vehicle	Available CSJ capture area insufficient for CSJ to accelerate the vehicle
RJ	Maintained, used as primary propulsion	M=2-6	Same as for RJ on Mach 12 vehicle	Not conducted	Test bed installation

(C) FIGURE 5-35
GE5/JZ6C TRJ INSTALLED ENGINE PERFORMANCE



(U) FIGURE 5-36
CSJ INSTALLATION ON MACH 6 VEHICLE

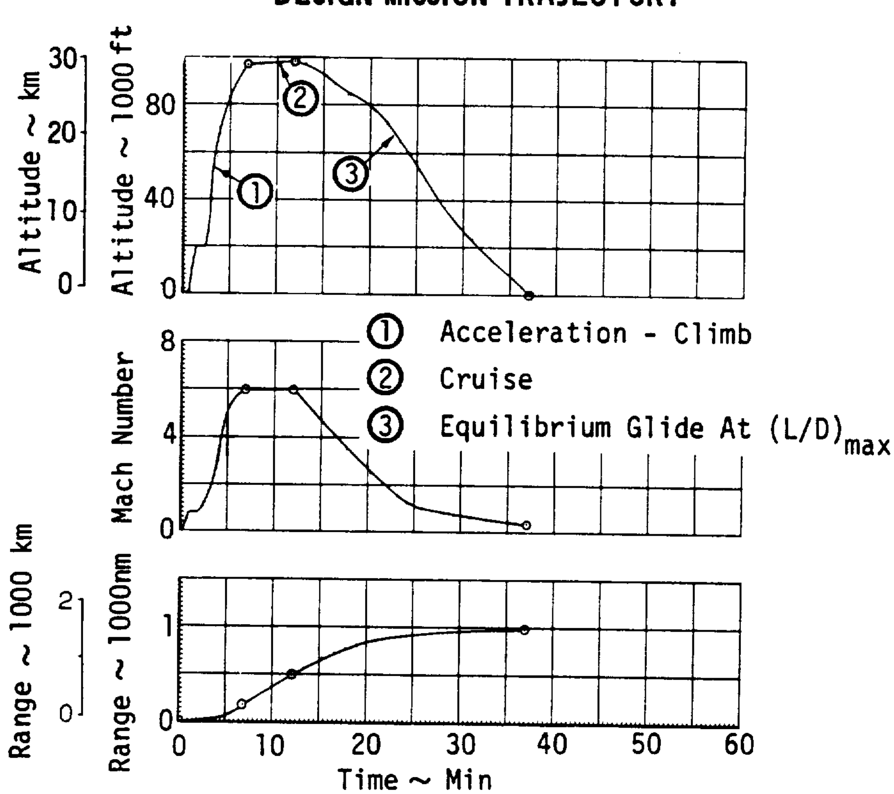


5.5 PERFORMANCE AND TRAJECTORIES

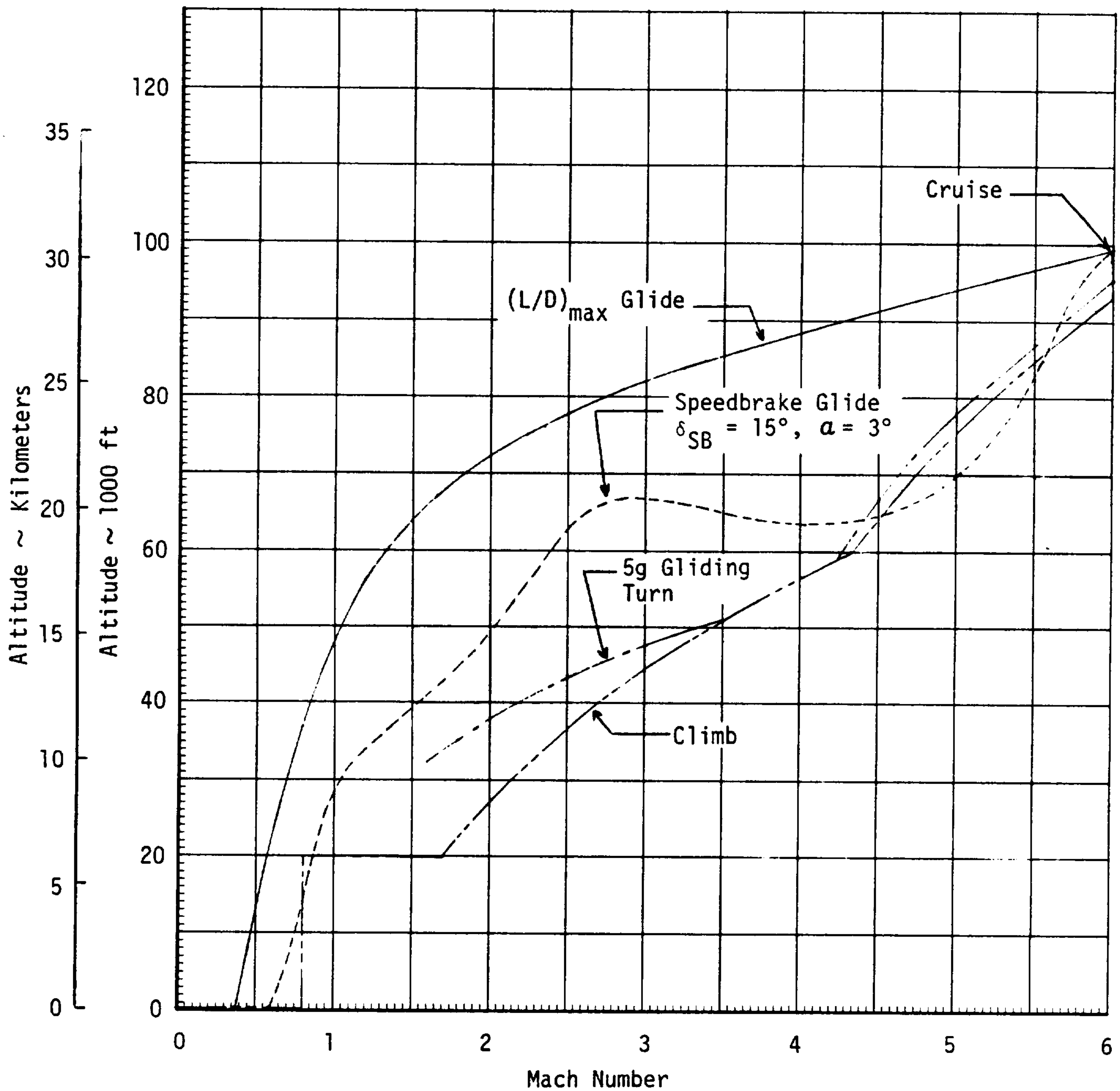
(U) The performance and trajectories for the basic vehicle and several configuration options are presented in the following sections.

5.5.1 (U) BASIC VEHICLE - The design mission trajectory for the basic vehicle as defined in Section 2.3 is shown in Figures 5-37 and 5-38. Following the acceleration - climb and cruise, three modes of unpowered glide are illustrated. Maximum range is provided by gliding at $(L/D)_{max}$. Minimum range will be obtained by deflecting the rudder speedbrakes and reducing the angle-of-attack to 3°. Lower angles-of-attack would result in a trajectory that exceeds the design dynamic pressure limitation of 2000 lb/ft^2 (95,800 N/m^2). The pilot can modulate the use of the speedbrakes to provide a variation in glide range as required for energy management purposes. The third mode of descent shown is a gliding turn performed at a normal load factor of 5 g following the five-minute cruise. A descent to an altitude of 78,000 ft (24,000 m) is required to provide sufficient dynamic pressure to achieve 5 g.

(U) FIGURE 5-37
MACH 6 TURBORAMJET
DESIGN MISSION TRAJECTORY

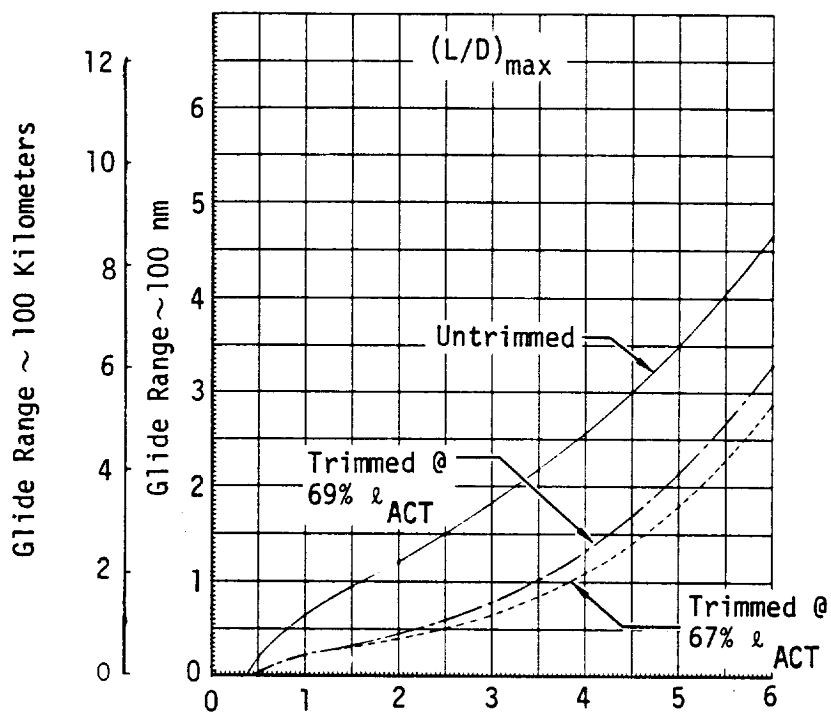


(U) FIGURE 5-38
MACH 6 TURBORAMJET
DESIGN MISSION PROFILE



(U) In Figure 5-39, the glide range obtained when employing the untrimmed $(L/D)_{max}$ values determined by the basic Phase II methods is compared with the glide range obtained using trimmed $(L/D)_{max}$ values from the Gentry analysis. Two center of gravity locations are utilized for the Gentry data to indicate the effects of static stability on trim drag as discussed in Section 5.3.2. The more aft center of gravity location results in less static stability and therefore, requires less stabilator deflection to trim. This results in a higher value of $(L/D)_{max}$ and more range. It also indicates the further configuration refinement required to reduce trim drag, as discussed in Section 5.3.2.

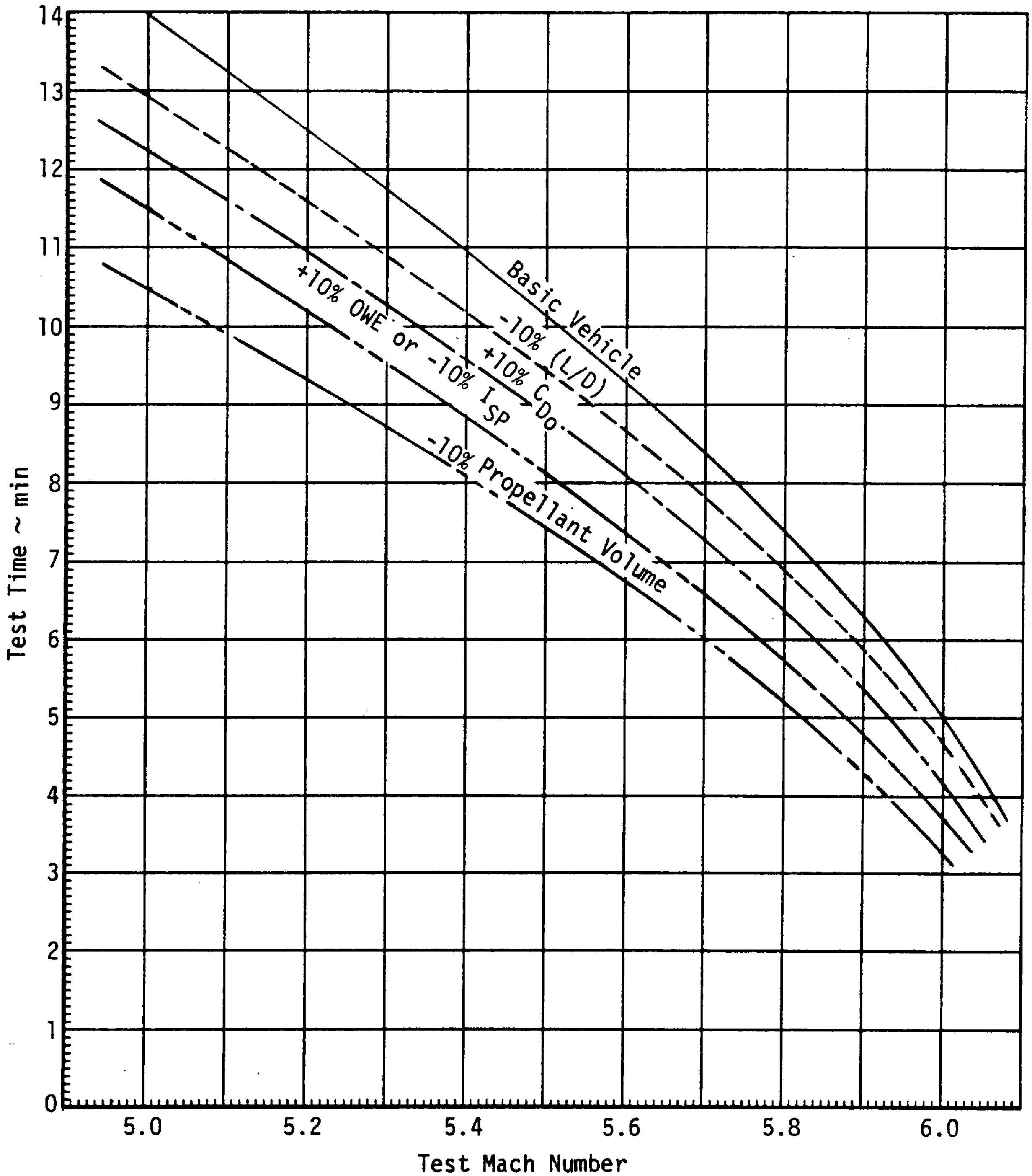
(U) FIGURE 5-39
MACH 6 TURBORAMJET
GLIDE RANGE CAPABILITY



(U) The basic vehicles performance sensitivities to changes in vehicle physical characteristics are shown in Figure 5-40. OWE, propellant weight, rocket engine specific impulse, $(L/D)_{max}$ and zero lift drag are varied 10% in a direction that will degrade performance. The results are shown in Figure 5-40 in terms of test time and test Mach number achievable.

(U) The sensitivity of each parameter on performance can be determined by comparing to the basic vehicle performance also shown in Figure 5-40. The most sensitive parameters are OWE, engine specific impulse, and propellant volume. The aerodynamic parameters are of greater impact to the Mach 6 configuration than to the Mach 12 rocket because of the higher q climb profile employed for the airbreather.

(U) FIGURE 5-40
MACH 6 TURBORAMJET PERFORMANCE SENSITIVITIES
Individual Sensitivities – Not Cumulative



However, the performance sensitivity to variations in the aerodynamic parameters remains significantly less than for other variations investigated. The rapid decline of test time with increasing Mach number is due to the fall-off of engine I_{SP} resulting from cooling requirements.

5.5.2 (U) CONFIGURATION OPTIONS - Figure 5-41 presents the performance capabilities in terms of test time vs test Mach number, for three research options available to the basic configuration. They are the armament modification, the all hydrogen STRJ11A-27 modification, and the all hydrogen advanced technology GE 5/JZ6C engine modification. A fourth option involving changes to the thermal protection system has no effect on the basic vehicle performance.

(U) The armament modification increased the OWE by 2192 lb (993 kg) and increased C_{D_0} as shown in Figure 5-12. The resulting performance of the configuration is 3.2 minutes of cruise at Mach 6 and 10.7 minutes of cruise at Mach 5.

(U) The performance of the all hydrogen fueled engine modifications can be increased by employing an external drop tank to provide increased fuel volume as shown. The drop tank carries sufficient LH_2 to climb to an altitude of 20,000 ft (6100 m) at .8 Mach number.

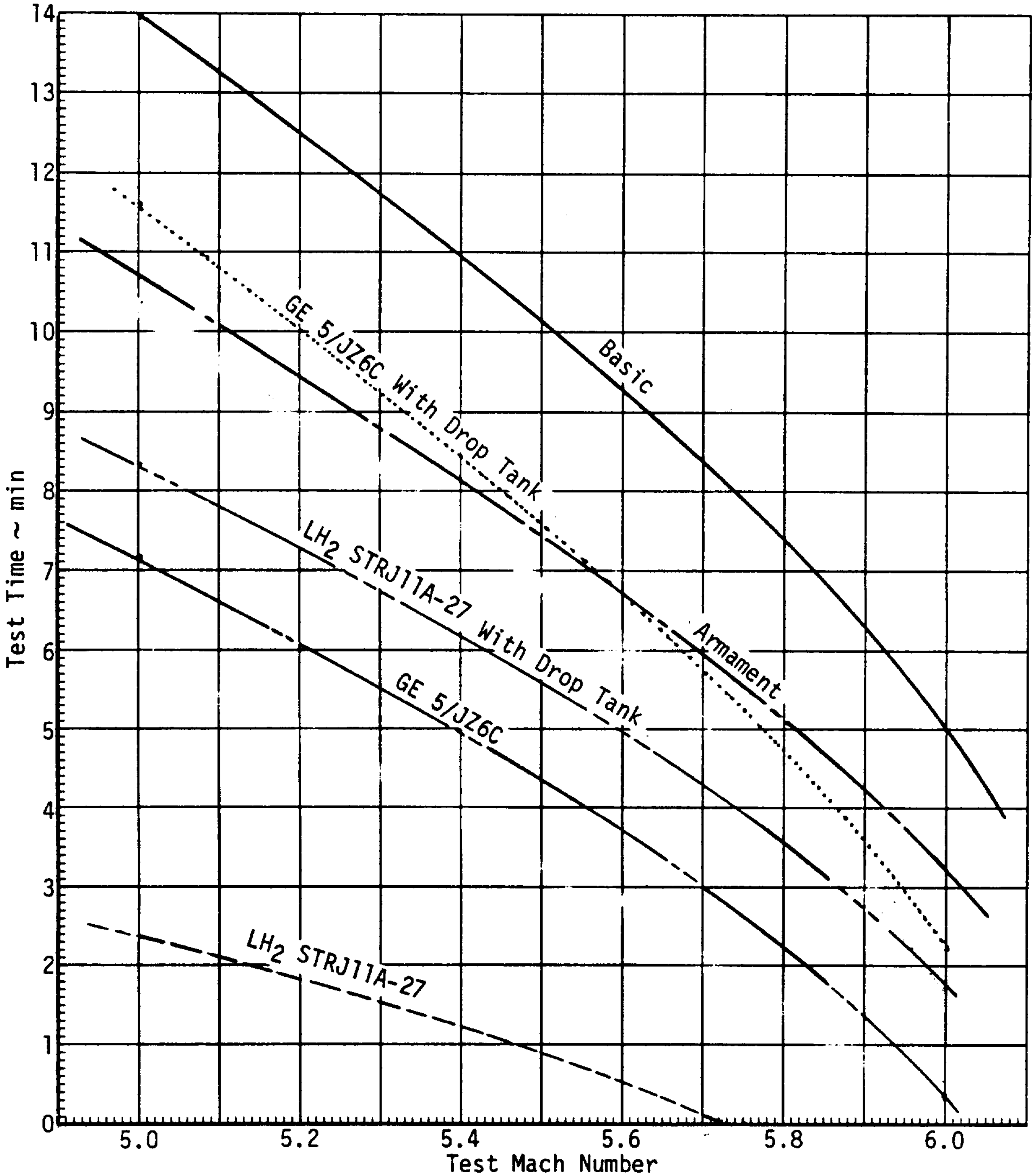
(U) Converting the JP/LH_2 STRJ11A-27 to all hydrogen involves no external changes to the basic vehicle. Internally, however, 30 ft^3 (.85 m^3) of fuel volume in the wings is unusable for LH_2. Operating on the internal fuel volume, the all hydrogen STRJ11A-27 version can attain a Mach number of 5.7 and can provide 2.4 minutes of test time at a Mach number of 5. Operating with a drop tank, 1.8 minutes of cruise time is available at Mach 6 and 8.3 minutes at Mach 5.

(U) The installation of the GE 5/JZ6C in the vehicle reduces the OWE by 2312 lb (1048 kg) and increases the available fuel volume by 43 ft^3 (1.21 m^3). With internal fuel, the GE 5/JZ6C version can attain a Mach number of 6 and can provide 7.2 minutes of cruise at Mach 5. With the drop tank, the vehicle provides 2.2 minutes of cruise at Mach 6 and 11.6 minutes at Mach 5.

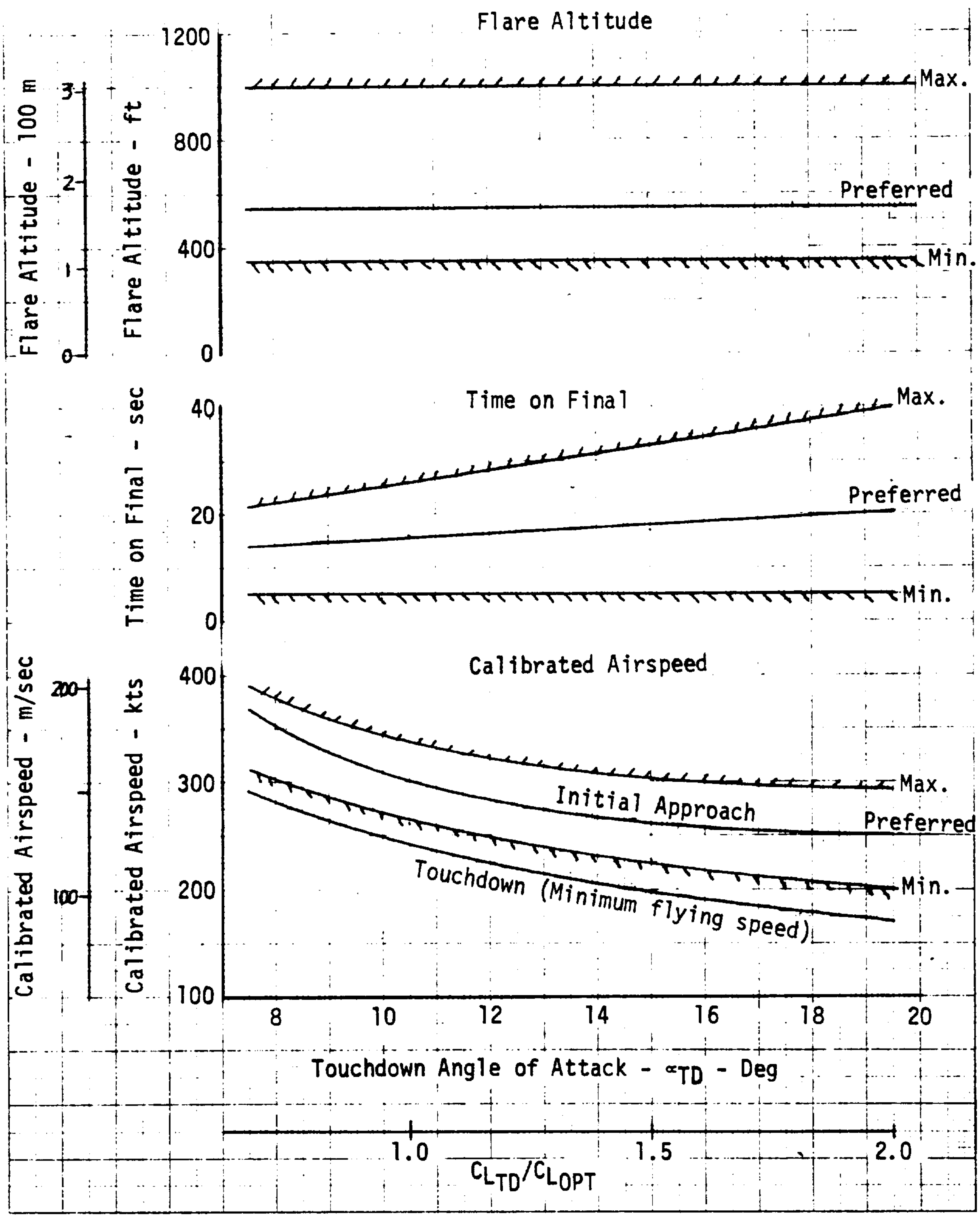
(U) All of the above performance is computed utilizing maximum turbojet power from takeoff through shutdown. A slight improvement can be obtained by using military power from takeoff up to 20,000 ft (6100 m) and Mach .8. This will yield a 35% saving in fuel consumed to that point. In the case of the STRJ11A-27 JP fueled turbojet, this amounts to 600 lb (272 kg) or only 12 ft^3 (.34 m^3). For the LH_2 turbojet, however, 200 lb (91 kg) or 43 ft^3 (1.22 m^3) less fuel is required to reach Mach .8 and 20,000 ft (6100 m). With the hydrogen fueled STRJ11A-27 as an example, the performance capability will be increased from 0 time at Mach 5.7 to 14 sec at Mach 6.

5.5.3 (U) TAKEOFF AND LANDING CHARACTERISTICS - Due to its high fineness ratio and low $VOL^{2/3}/S_p$, the subsonic performance is comparable to conventional fighter aircraft in terms of takeoff and landing characteristics. Liftoff at the design gross weight of 61,426 lb (27,860 kg) is accomplished at 230 knots (119 m/sec) with approximately a 4500 ft (1870 m) ground roll. The power-off approach and landing characteristics in terms of calibrated airspeed prior to flare initiation, flare altitude, time on final, and touchdown velocity are presented in Figure 5-42 as a function of touchdown angle-of-attack. The boundary conditions shown for the Mach

(U) FIGURE 5-41
MACH 6 TURBORAMJET
PERFORMANCE CAPABILITY WITH VARIOUS OPTIONS



(U) FIGURE 5-42
MACH 6 TURBORAMJET
APPROACH AND LANDING CHARACTERISTICS
POWER OFF



6 aircraft are a function of $(L/D)_{max}$, C_{L_α}, $C_{L_{OPT}}$, $C_{L_{TD}}$, and W/S and are obtained from the Reference (17) study. The preferred conditions shown correspond to the power off approach conditions receiving the best pilot ratings in the Reference (17) study for an aircraft exhibiting characteristics similar to the Mach 6 vehicle.

5.6 THERMODYNAMICS

(U) An assessment of the thermal environment for the basic Mach 6 research aircraft and its options has shown that radiation cooled external surfaces can be constructed of long life titanium and superalloy materials and do not require the use of refractory metals. The general decrease in external moldline temperatures experienced at Mach 6, relative to the Mach 12 vehicle, results in only a slight weight advantage (3% OWE) with the use of an active TPS and, hence a passive TPS was selected. Although uncooled refractory metal inlet liners could be utilized at Mach 6, regenerative cooling is employed because of a nominal weight advantage plus vehicle enhancement by allowing dash capability to speeds greater than Mach 6.

(U) Configuration options to increase the vehicle's research capability were found to have an insignificant effect upon the thermal characteristics of the basic vehicle. Like the Mach 12 version, provisions for conducting TPS research (passive and active systems) with the Mach 6 aircraft are recommended.

5.6.1 (U) BASIC VEHICLE - Maximum external temperatures, moldline materials, and thermal protection requirements for the basic Mach 6 research vehicle are discussed in the following paragraphs.

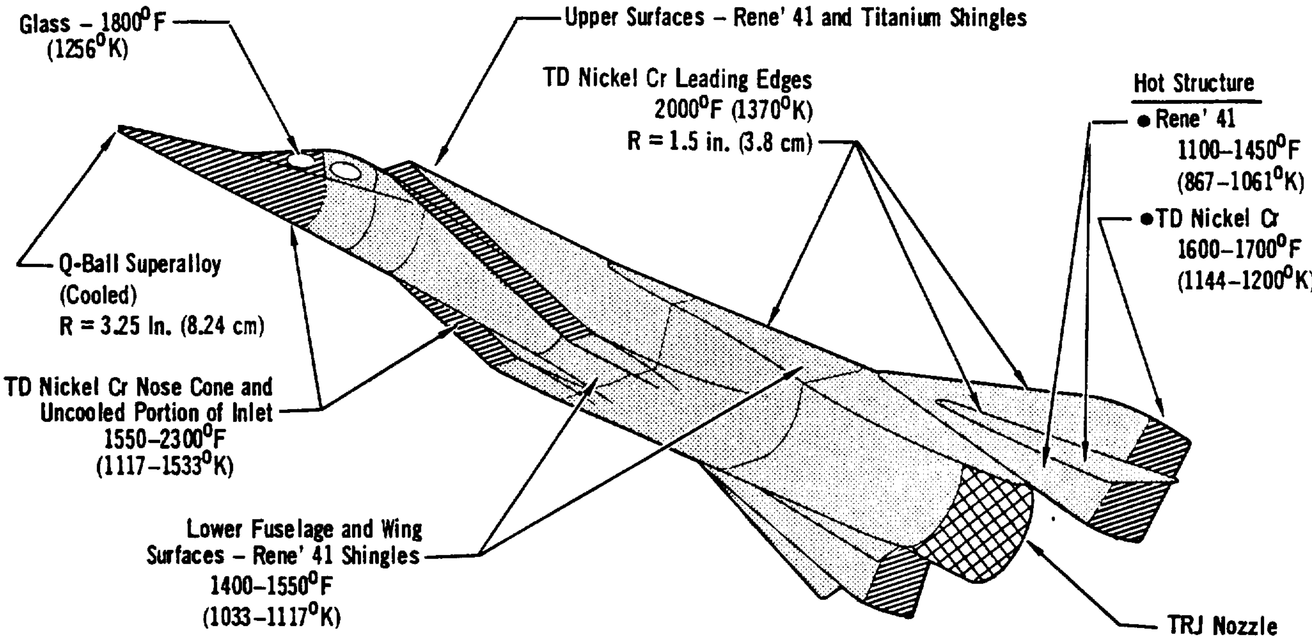
5.6.1.1 (U) Vehicle Temperatures - Moldline materials (shingles and insulation) for the basic Mach 6 vehicle have been selected to be compatible with maximum temperatures anticipated for the flight profiles presented in Figure 5-38. No significant change in moldline materials is required when the basic vehicle is reconfigured to perform the various research options (TPS, propulsion, armament, etc.) considered during this study.

(U) Maximum temperatures and corresponding shingle materials are presented in Figure 5-43. As shown in this figure, maximum external surface temperatures are below 2400°F (1590°K) so that conventional and superalloy materials can be used exclusively. Furthermore, since these are maximum temperatures they are only experienced for short times during those flights when the vehicle is pushed to design limits. Temperatures experienced during a nominal Mach 6 test run will range from 150°F (338°K) to 250°F (395°K) less than the maximum values presented herein.

(U) With the exception of stagnation regions, maximum surface temperatures were determined based upon the turbulent heating correlation of Spalding and Chi, and reradiation with a surface emissivity of 0.8. The Fay and Riddell heating correlation was used in predicting nose tip and leading edge equilibrium temperatures. Lower surface and delta tip temperatures are maximum during the 5 g turn condition. During the turn, the delta tips are deflected -30 degrees with maximum temperatures occurring on their upper surface. Speed brake temperatures are a maximum at initiation of the minimum range descent, Mach 6 at 98,000 feet (29,850 m). Maximum upper surface temperatures were determined for a push over angle of attack of -4° (-0.5 g) upon completion of the Mach 6 test run.

5.6.1.2 (U) Thermal Protection System (TPS) - Results of a trade study conducted during Phase II have shown that incorporation of an active TPS would not significantly enhance (3% reduction in OWE)the performance capability of the Mach 6

(U) FIGURE 5-43
MAXIMUM SURFACE TEMPERATURES
(Basic Mach 6 TRJ Aircraft)



Surface Material Distribution

Material	% of Surface Area
Titanium	18
Rene' 41	78
TD Nickel Cr	4

research vehicle and, hence, a passive TPS has been selected. The general arrangement and insulation thickness requirements for typical locations on the vehicle are presented in Figure 5-44. Flexible Min-K type insulation was used whenever practicable (maximum temperature capability of 1800°F, 1260°K) to minimize insulation thicknesses and the length of shingle attachments. The passive TPS defined in Figure 5-44 has been designed to limit maximum structural temperatures to the following:

- o Aluminum fuselage structure; 250°F (394°K) in fuel tank areas, 300°F (422°K) in non-fuel areas.
- o Titanium inlet and engine support structure; 300°F (422°K).

5.6.1.3 (U) Inlet Cooling - During ramjet operation, internal inlet walls are regeneratively cooled to 1550°F (1118°K) using the liquid hydrogen fuel as a coolant. Regenerative cooling of the inlet avoids the need for a refractory metal inlet liner (uncooled inlet walls reach a temperature of about 2600°F, 1700°K, at Mach 6), reduces insulation requirements, reduces total inlet weight by approximately 10%, and provides growth capability to higher Mach numbers. The Pratt and Whitney

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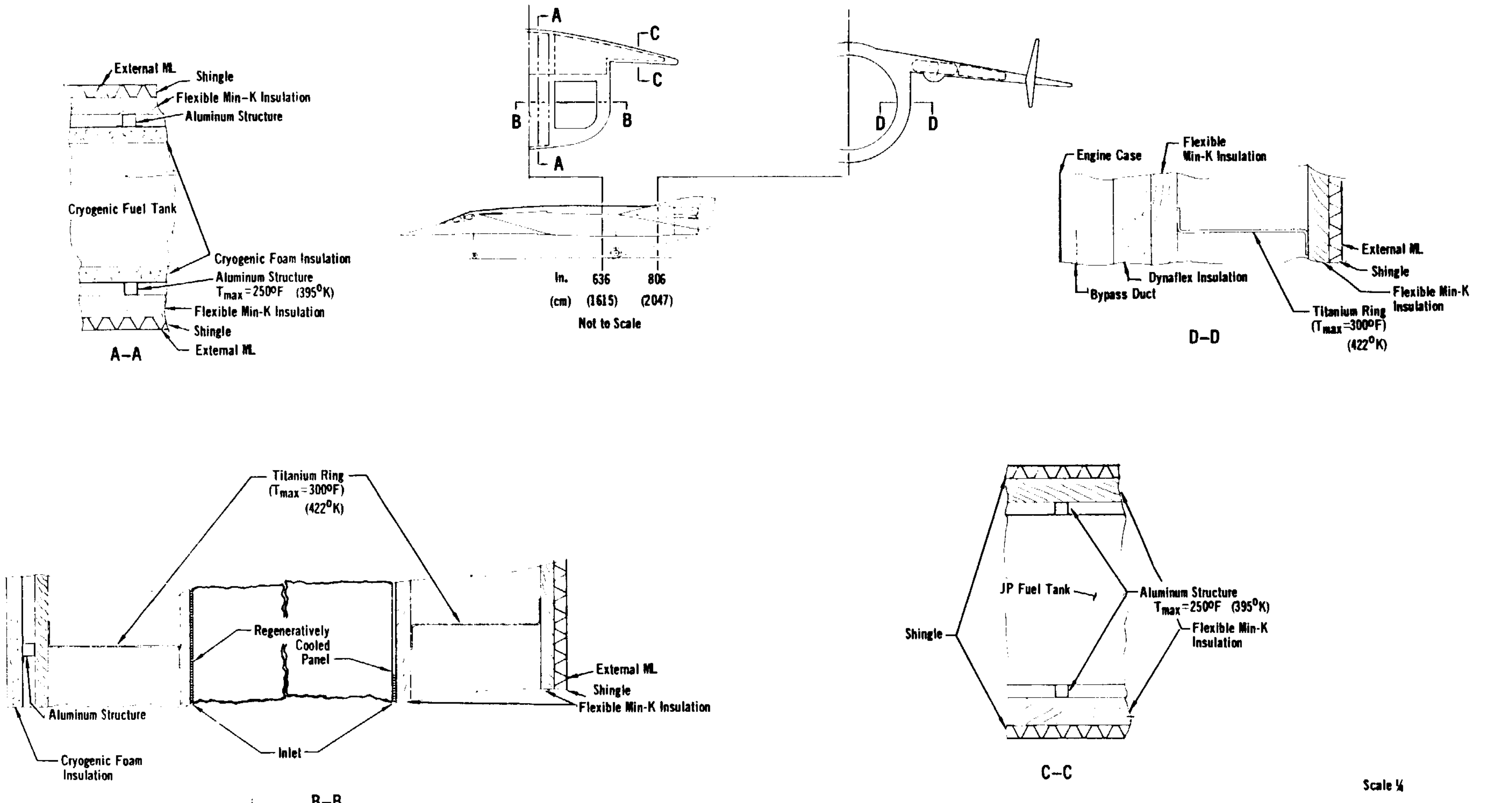
(U) FIGURE 5-44
PASSIVE THERMAL PROTECTION REQUIREMENTS
(Basic Mach 6 TRJ Aircraft)



FOLDOUT FRAME 1

FOLDOUT FRAME 2

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STRJ11A-27 turboramjet selected for the Mach 6 research aircraft presently is throttled to an equivalence ratio of about 0.5 to match thrust equal to drag at Mach 6. However, operation at an equivalence ratio of 1.0 (in terms of hydrogen flow rates) is still required because of inlet (ϕ = 0.25) and engine (ϕ = 0.75) cooling requirements. Hydrogen flow, in excess of thrust requirements, by-passes the combustor and is dumped overboard.

5.6.2 (U) CONFIGURATION OPTIONS - The thermal aspects of reconfiguring the basic vehicle to enhance its research value are discussed below.

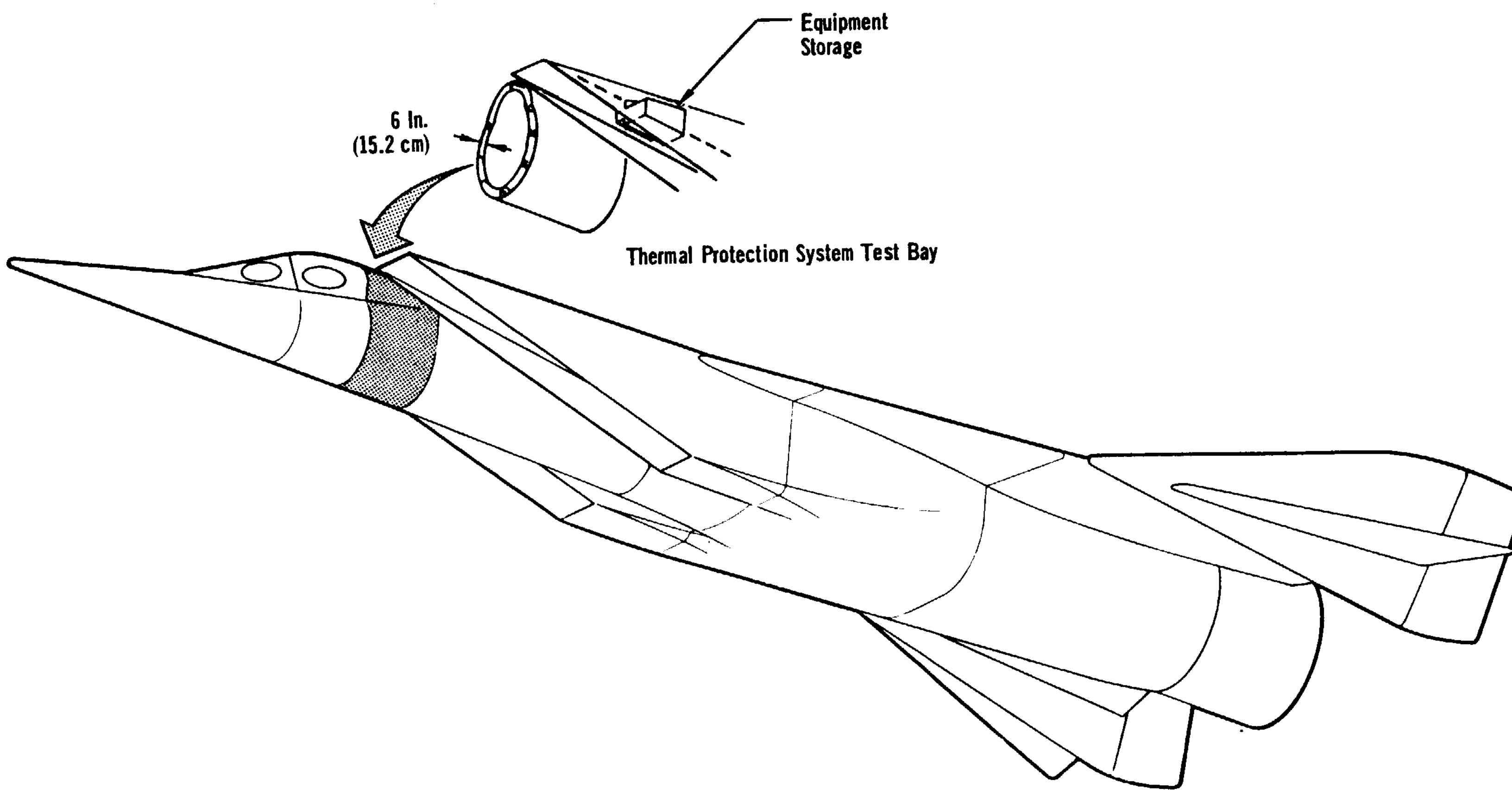
5.6.2.1 (U) Armament (ARM) Option - Research in the area of high speed weapon delivery can be accomplished with the addition of a missile bay on the lower surface of the aircraft as described in Section 5.2. During these tests (doors open), the interior of the bay will be subjected to locally high heating rates due to separation and re-attachment of the boundary layer. In the extreme situation, weapon delivery at Mach 6, it was estimated that local heating rates would approach 100 Btu/ft^2 sec (1140 kw/m^2). If unprotected, these heating rates would seriously overheat the aircraft's structure and test bay in considerably less time than the 30 to 60 seconds that the armament doors are open. Addition of a 0.125 inch (0.318 cm) layer of ablation material (0.55 psf, 2.68 kg/m^2) was determined to be adequate to maintain structural temperatures within design limits.

4.6.2.2 (U) Thermal Protection System (TPS) Option - The need for conducting TPS research, contemporary thermal protection concepts, and aircraft modifications required to conduct TPS research were discussed previously in Section 4.6. Although aerodynamic heating rates are considerably lower on the Mach 6 vehicle than on the Mach 12 aircraft, meaningful TPS research can still be conducted at this lower Mach number. As indicated in Figure 5-45, the recommended research option for the Mach 6 aircraft reserves a 5 foot (1.53 m) long test section around the periphery of the vehicle aft of the forward LH_2 fuel bulkhead. As shown in this figure, a 6 inch (15.3 cm) deep bay for testing various TPS concepts plus an additional 9 cubic feet (0.255 m^3) for equipment storage has been provided. Two hundred pounds (90.8 kg) of provisions have been allocated for conducting TPS research and are included in performance calculations. Considerations leading to the recommendation of this TPS modification concept over alternate concepts are essentially the same as those listed in Section 4.6 for the Mach 12 vehicle. However, the location available for TPS modification on the Mach 6 aircraft exhibits two disadvantages relative to the Mach 12 aircraft. Inlet duct structure adjacent to the fuselage moldline on both sides at this location reduces the area available to obtain meaningful TPS research. Also, since basic fuel requirements necessitate utilizing as much fuselage volume for cryogenic fuel storage as possible, the TPS modification is located solely in a cryogenic tankage area. However, if it is desired to conduct such research it appears that reasonable alternative concepts could be developed without compromising the basic airplane.

4.6.2.3 (U) Propulsive Research Options - Three engine modifications have been considered for the Mach 6 aircraft. These are:

- GE 5/JZ6 TRJ
- CSJ
- RJ

(U) FIGURE 5-45
CONFIGURATION OPTION FOR CONDUCTING TPS RESEARCH
(Mach 6 TRJ Aircraft)



The first two modifications results in minimal changes in moldline geometry, surface temperatures, and shingle material distributions presented in Figure 5-43. Major modifications to the lower surface of the Mach 6 aircraft result with the CSJ modification to achieve the necessary ramp angles. In comparison with similar fuselage stations on the basic aircraft, the resulting maximum lower surface temperatures are increased by 60°F (33°K) along the first ramp and by 110°F (61°K) along the second ramp. A slight alteration of shingle material distribution given in Figure 5-43 also results.

(U) The shorter length of the JZ6 is accommodated by the addition of a constant section of inlet to fill the gap between the STRJ11A-27 engine face and the GE5/JZ6 engine face. This additional surface areas must be regeneratively cooled and this increases the inlet coolant flow requirements by about 15%. However, the shorter length of the GE5/JZ6 proportionately reduces engine coolant flow rate. The combined inlet plus engine cooling requirement is 4.4 lb/sec (2.0 kg/sec), resulting in a cooling equivalence ratio of 0.8.

(U) The convertible scramjet module has a ratio of wetted area to capture area approximately one half that of the bifurcated TRJ inlet on the basic aircraft. The associated heating rates, while operating in the supersonic combustion mode at Mach 6, require an equivalence ratio of 0.3 to maintain internal CSJ wall temperature below 1550°F (1118°K).

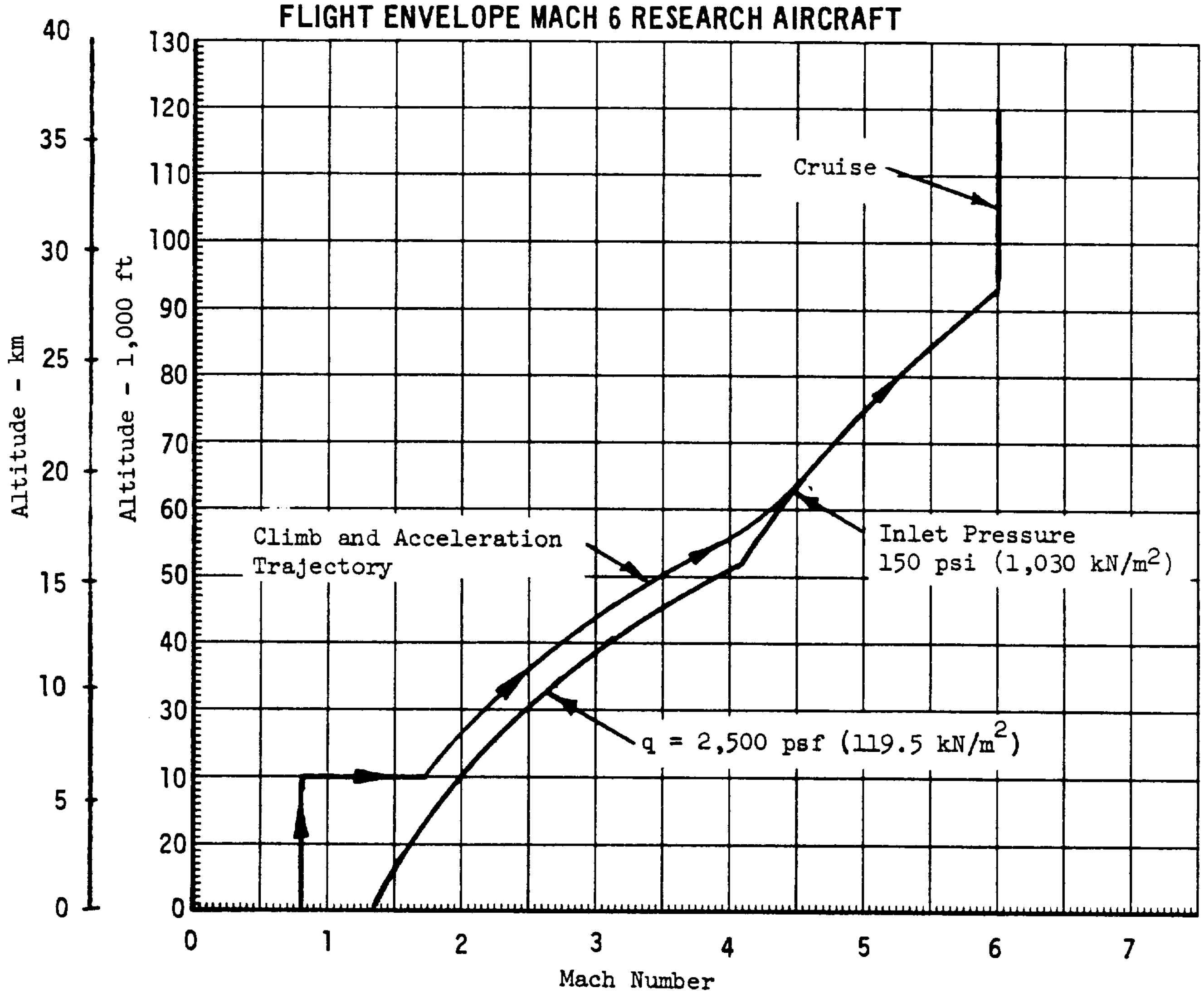
(U) The ramjet module for the Mach 6 aircraft is identical to that for the Mach 12 aircraft. Adequate cooling of the inlet plus engine is achieved at an equivalence ratio of 0.5.

5.7 STRUCTURES

(U) The goals, scope, and depth of the structural analysis of the Mach 6 aircraft are the same as those described for the Mach 12 aircraft in Section 4.7. Primary consideration was given to the fuselage structure, thermal protection system, and engine inlet. The basic vehicle structural arrangement shown in Figure 5-1 was used to make concept selections and structural comparisons. Some of the important considerations in the basic concept selections are integration of the engine/inlet and the airframe, the LH_2 fuel tank, and the flight environment.

5.7.1 (U) ENVIRONMENT - The loads and temperatures used in the Phase III evaluations were based on the flight profile in Figure 5-46. The most severe temperatures and loads for most of the structure occur during the 5.0 g wind-up turn at Mach 6. This maximum temperature condition results in temperatures as shown on Figure 5-43.

(U) FIGURE 5-46
FLIGHT ENVELOPE MACH 6 RESEARCH AIRCRAFT



(U) Loads used for structural comparisons are based on a loading environment resulting from the conditions described in MIL-A-8861(ASG) and MIL-A-8862(ASG). These are used as a guideline. However, some variations have been made for purposes of simplification of analysis and comparison. The variations to these specifications are the same as shown for the Phase II trade studies and are noted in Figure 5-3.

(U) For the Mach 6 aircraft, as for the Mach 12 aircraft, the significant loading conditions are taxi, takeoff, and landing for fuselage bending and gear loads, whereas the high speed maneuver is the major contributor to inlet pressure loads and control surface loads.

5.7.2 (U) MATERIALS - This aircraft uses the same material and fabrication technology as the Mach 12 aircraft described in Section 4.7.3. The material candidates used in the structural concepts comparisons were selected on the basis of material efficiencies. It is understood, however, that in actual design the material selected is dependent also on the specific application, gage, type of construction, local environment, weight, and cost.

5.7.3 (U) FABRICATION METHODS - The fabrication methods and basic philosophy for material application described for the Mach 12 aircraft in Section 4.7.3 are also applicable to the structural concepts of the Mach 6 aircraft.

5.7.4 (U) CONCEPT COMPARISON - The following paragraphs discuss the concept comparisons used as the basis for the selection of the structure for the basic research aircraft.

5.7.4.1 (U) Primary Structure - The fuselage structure in the LH_2 fuel tank area is very similar to that of the Mach 12 aircraft. The major exception is that the tank surface area has very few large uninterrupted panels and, therefore, is fabricated of stiffened skin construction rather than the corrugated sandwich. The stiffened skin results in lower weight for this application than the sandwich, primarily because the fuselage loads are more localized and highly concentrated than those of the Mach 12 vehicle. This is due to the many cutouts, concentrated masses, and load path direction changes. There will be greater radial bending in the Mach 6 fuselage structure because of the internal pressure in the non-circular tanks. This requires very little additional structure because of the relatively small cross sectional area (low bending moment) and the pre-existence of a highly loaded structure (dual function). Selection of aluminum over titanium is, as for the Mach 12 aircraft, based on lower cost, and slightly lower weight. The selection was also influenced by cryogenic propellant compatibility considerations and the greater experience background with aluminum cryogenic tanks than with titanium.

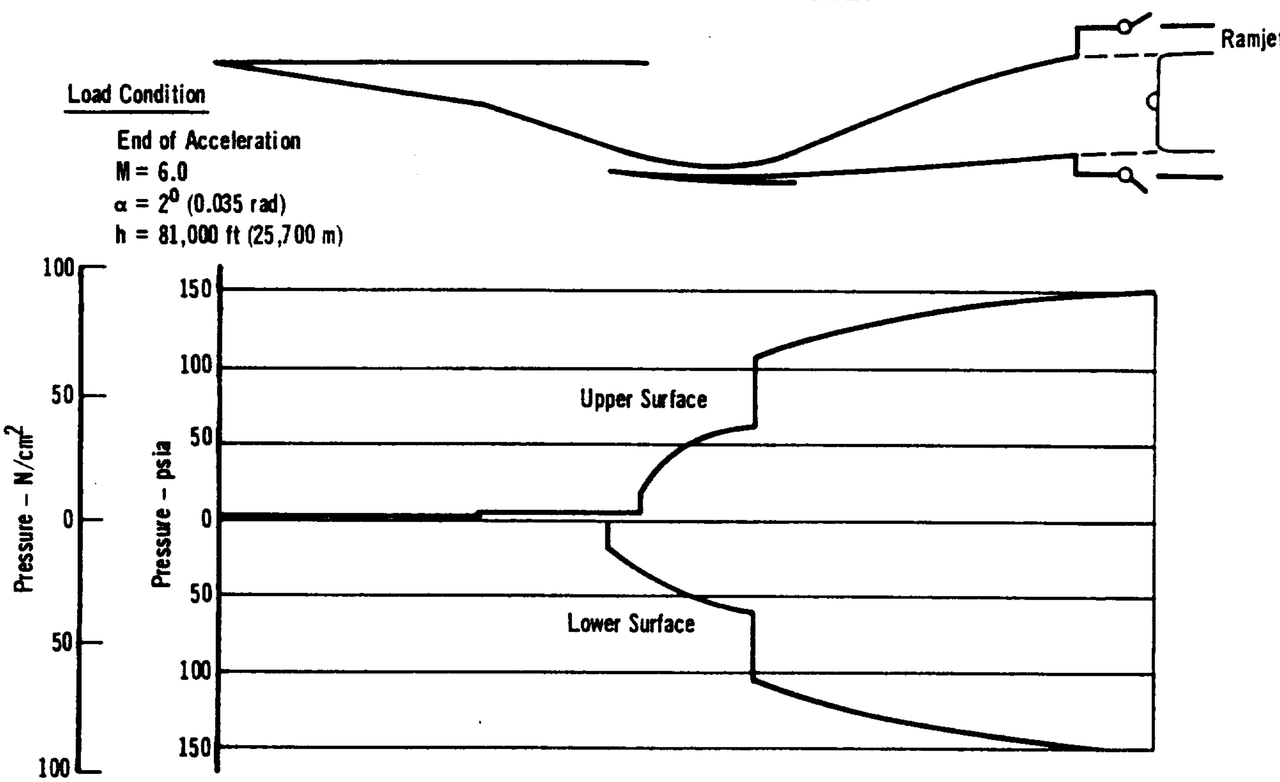
(U) Non-fuel tank areas of the fuselage are integrated with the engine/inlet and the selection of the structural concept is influenced primarily by the inlet environment.

5.7.4.2 (U) Wing Structure - The main wing box is located above the engine/inlet cavity where the material selection, as in the fuselage, is influenced by the high temperature in the inlet. The wing structure is insulated titanium rib and spar construction. Because of the high sweep angle there is no single major carry-through; rather, the spars are located parallel to fuselage station lines and are continuous between wings.

5.7.4.3 (U) Engine Inlet - The engine inlet is the most challenging of the structural development programs on the Mach 6 research aircraft. The combination of high temperature, high loads, movable structure, and integration with the fuselage requires structurally sophisticated concepts for low weight design. Selection of the structural concept and thermal protection system are based on the load and temperature conditions occurring during the 5.0 g maneuver at Mach 6 and 90,000 ft (27.4 km) altitude. Temperature of the liner (uncooled) in the inlet approaches the total temperature of 2600°F (1700°K) during the Mach 6 flight. During the maneuver, the inlet pressure reaches a peak of 150 psi (103 N/cm^2) as shown in Figure 5-47, which is over three times the pressure, 40 psi (27.6 N/cm^2), at normal cruise conditions.

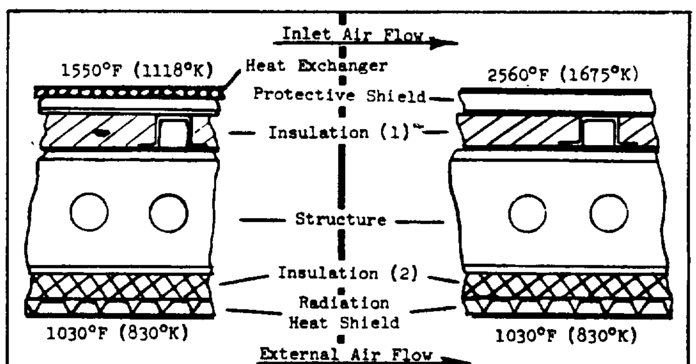
(U) The two basic concepts considered for inlet were hot structure of columbium alloy and insulated structure of titanium alloy. For the insulated structure there are two concepts that were considered: passive insulation with columbium shingles and passive insulation with a regeneratively cooled liner made of a superalloy. During the preliminary consideration, the hot structure columbium concept was discarded for two primary reasons. First, the weight was over six times greater for structure of 2500°F (1640°K) columbium than for structure of cooled titanium. Second, thermal deflections and distortions are a severe detriment to the efficient operation of the inlet because of the movable ramps and the geometric tolerances of the throat.

(U) FIGURE 5-47
TURBORAMJET INLET PRESSURES



(U) A comparison of the insulated concepts required consideration of the trade-offs of structural temperature vs insulation weight, and fuel cooling capacity vs heat exchanger temperature. A summary of the results is shown in Figure 5-48. The comparison shows that there is about a 20% difference in total system weight between the regenerative cooled surface and the shielded concept. The major difference in weight is in the protective shield which must transmit the high inlet pressures into the cooled substructure. Reduced temperature of these load bearing surfaces due to cooling is the main factor contributing to the reduced shield and insulation weight. The regeneratively cooled concept has been selected for the inlet on the basis of weight and growth capability. The shingled concept is limited to about 2800°F (1810°K) because of the temperature limit of the columbium shingle which limits the speed to no higher than Mach 6.4. The regeneratively cooled liner is made of Rene' 41 tubular construction having a wall gage of .004 inch (.0102 cm). This is based on an independent study comparing several shapes and materials for heat exchanger design (Reference (9)). Reducing the temperature of the substructure in the cooled concept to 300°F (422°K) from 800°F (700°K) not only reduced the structural weight 18%, but reduces the thermal deflection of the inlet shape as well. However, the increased insulation weight resulted in a similar total weight for the inlet structure/thermal protection system. Because of the lower deflection, slightly lower weight, and higher structural confidence, the 300°F (422°K) system was selected.

(U) FIGURE 5-48
THERMAL/STRUCTURAL CONCEPT COMPARISON
Mach 6 TRJ Inlet



Structure Material Temperature	Titanium 300°F (422 °K)	Titanium 800°F (700 °K)	Titanium 300°F (422 °K)	Titanium 800°F (700 °K)
Weight-lb/ft^2-(kg/m^2)				
Heat Exchanger	3.20 (15.60)	3.20 (15.60)	-	-
Protective Shield	-	-	5.20 (25.40)	5.20 (25.40)
Insulation (1)	.73 (3.56)	.40 (1.95)	2.11 (10.30)	1.06 (5.17)
Structure	4.10 (20.00)	4.98 (24.30)	4.10 (20.00)	4.98 (24.30)
Insulation (2)	.67 (3.27)	.32 (1.56)	.32 (1.56)	.07 (.34)
Radiation Shield	.67 (3.27)	.67 (3.27)	.67 (3.27)	.67 (3.27)
Total	9.37 (45.70)	9.57 (46.68)	12.90 (60.53)	11.98 (58.48)
Thickness-in.-(cm)	5.45 (13.8)	5.19 (13.2)	7.14 (18.1)	5.70 (14.5)

5.7.4.4 (U) Thermal Protection System - The TPS study performed in Phase II showed that the active system (water wick) is lower weight than the passive system by about .2 lb/ft^2 (8.41 kg/m^2). However, the Phase III investigation of the inlet structure concept for the Mach 6 aircraft indicates that it may be advantageous to have a passive TPS on the external surface. This provides an internal insulation concept (protecting the structure from the inlet air) that is the same as the external insulation concept (protecting the structure from the boundary layer air). The result is a uniform temperature in the structure with a maximum of about 300°F (422°K). The water wick system, if it were used on the outer surface of the inlet structure, would result in temperature gradients of 200-300°F (366-422°K) with the cool side on the outside. In the area where the structural mass is relatively large, this would not allow maximum use of the available heat sink (max structural temperature would be about 100°F (310°K) in the outer cap).

(U) The heat shield analysis and selection for the Mach 6 aircraft is the same as described for the Mach 12 aircraft. The critical condition for both aircraft occurs during the 5 g maneuver at the maximum allowable shingle temperature.

5.7.4.5 (U) Control Surfaces - Load intensity for the control surfaces is similar to that of the Mach 12 aircraft, since they both are capable of 5.0 g maneuvers. Size and thickness relationships are also similar and, therefore, as on the Mach 12, the hot structure, stiffened skin concept is selected for the basic vehicle cost and weight estimates as discussed in Section 4.2.2.

5.8 PROPELLANT SYSTEMS

(U) The propellant system design is a combination of a current technology JP system and adaptation of existing cryogenic technology to horizontal operations for the LH_2 system. Redundancy is provided in critical elements such as boost pumps and pressure regulators. This approach has resulted in a high confidence definition of major system functional areas as illustrated schematically in Figure 5-49. Location of the major system components is shown in Figure 5-1.

(U) Safety features are incorporated for the LH_2 system similar to those discussed for the Mach 12 vehicle. The use of nitrogen pressurization for the JP tankage provides inert atmosphere, thereby minimizing fire and explosion hazards.

(U) The propellant system is comprised of the following major subsystem/ operational areas:

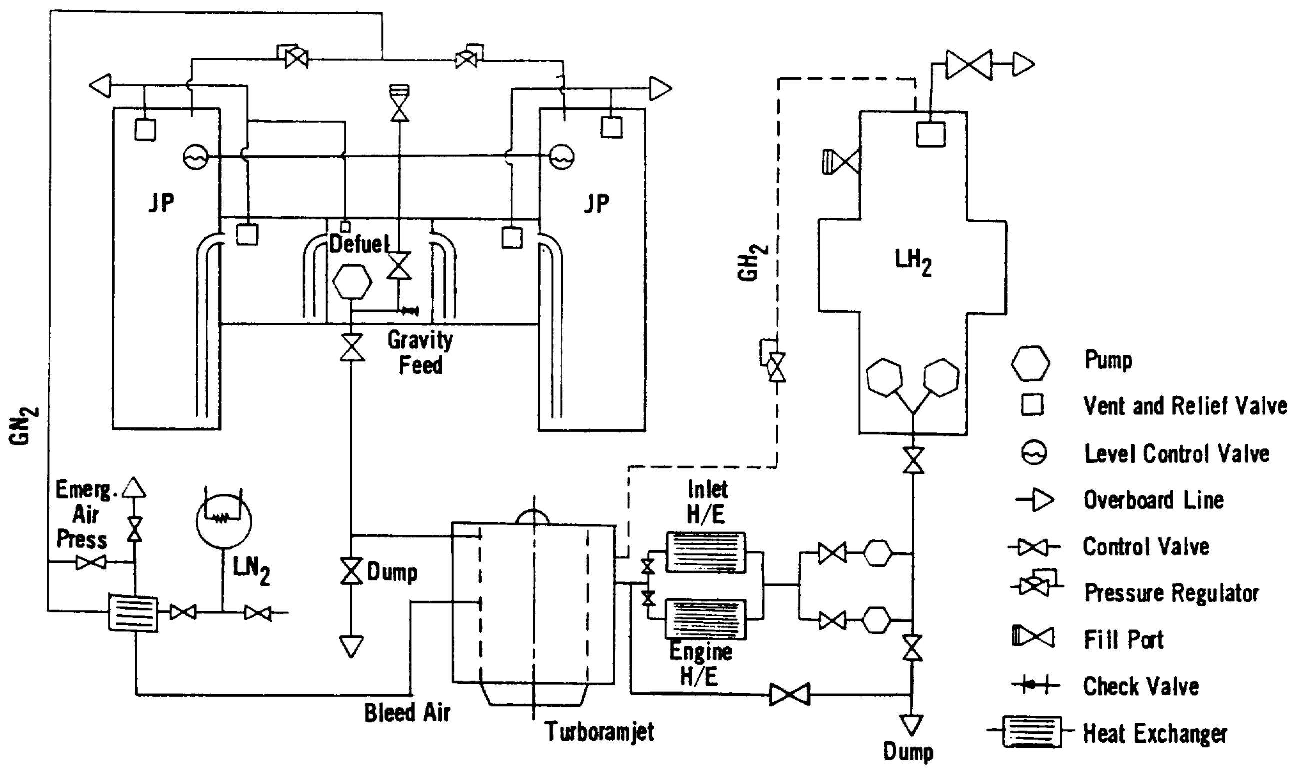
- JP System
  - Tankage
  - Feed System
  - Pressurization
- LH_2 System
  - Tankage
  - Feed System
  - Pressurization

5.8.1 (U) BASIC VEHICLE - The propellants for the basic Mach 6 flight vehicle are JP and subcooled LH_2. JP is utilized in the turbojet propulsion mode for takeoff and acceleration to Mach 3.5. LH_2 is employed in the ramjet mode for acceleration from Mach 0.8 to cruise.

(U) JP Fuel System - The turbojet core engine (J-58) is designed for operation with a high thermal stability fuel. Currently JP-7 is an available fuel which meets the stringent requirements imposed by the J-58 environment. Attractive alternate fuels currently under development for this application are hydrotreated JP-5 or hydrotreated JET A-1, both middle distillate kerosenes. The major advantages of using the hydrotreated JP's (HT-JP) as compared to JP-7 include increased density and availability and a resultant reduction in cost. These fuels have been produced in pilot plant operations and are currently being investigated by both government and private agencies for thermal stability and characterization factors. HT-JP was selected for application on the basic vehicle. Use of this fuel will provide valuable fuel system design and operational data relative to advanced military and commercial aircraft.

(U) Details of both the HT-JP and LH_2 fuel systems are presented schematically in Figure 5-49. The HT-JP is carried in the wing tanks with flow toward the center feed tank. Because of the flat design, the tanks are compartmented to permit pressure transfer to the feed tank. The feed tank is equipped with an electric drive boost pump to provide fuel flow to the engine. Should this pump fail, emergency operation will permit pressurized fuel feed to the fuel system/engine interface. Such an occurrence would require an emergency return to base, but would not endanger either aircraft or personnel.

(U) FIGURE 5-49
PROPELLANT SYSTEM SCHEMATIC



(U) Nitrogen is used to pressurize the tankage during all flight phases with air as an emergency backup. The nitrogen, stored as a liquid, is heated to 530°R (285°K) and fed to the tankage at a regulated 10 psig (6.89 N/cm^2). Advantages of nitrogen pressuization include complete inertion of the vapor space, thereby minimizing fire/explosive hazards, and reduction of fuel deposit formations through reactions between the fuel and oxygen.

(U) <u>LH_2 System</u> - The high speed acceleration and cruise LH_2 fuel system is also shown in Figure 5-49. Basic design philosophy as to tank pressure, boost pumps, and pressurization system are the same as for the basic Mach 12 vehicle; subcooled LH_2, redundant boost pumps, and autogenous GH_2 pressurization. The LH_2 storage tank is an internally insulated, straight walled, single tank. Electric powered dual boost pumps provide engine fuel flow and provide sufficient redundancy in the event of failure. LH_2 dump is accomplished by opening the engine feed bypass valve and dumping out the aft portion of the vehicle. Pressurization is by GH_2 bleed from the engine/inlet regenerative cooling loop, regulated to 10 psig (6.89 N/cm^2), and introduced into the high point of the tank. Required control information includes fuel quantity, flow rate, and tank/boost pump outlet pressure. Leak detection provisions are essentially the same as for the Mach 12 vehicle. Vapor detectors, sniffers and fire suppression equipment will be located in tankage and flow component areas.

5.8.2 (U) CONFIGURATION OPTIONS - The armament option will not result in any propellant system changes.

(U) The JZ6 TRJ Modification is an all LH_2 configuration which incorporates a centerline drop tank to provide increased fuel volume for acceleration and cruise. This tank, consisting of a rather small diameter cylindrical section, can efficiently utilize pressure transfer to the main feed tanks. Operationally, the fuel in the drop tank is used first and the tank is then dropped prior to transonic acceleration to minimize drag and weight.

(U) The CSJ option includes integral LH_2 tanks, pumps, and distribution lines as part of the research pod. This integral feature precludes any impact on the basic flight vehicle propellant system. This fuel system is similar in design and operational aspects to the basic LH_2 system and uses regenerative hydrogen for tank pressurization.

(U) The RJ modification includes two distinct approaches. The fixed inlet ramjet missile application utilizes a self contained, pressure feed JP type fuel system. LH_2 is used for the variable inlet ramjet which requires a LH_2 takeoff downstream from the basic flight vehicle's boost pump. The high pressure fuel pump for the LH_2 ramjet is an integral part of the propulsion research pod.

5.9 SUBSYSTEMS

(U) The vehicle subsystems consisting of avionics and miscellaneous subsystems are described in the following paragraphs.

5.9.1 (U) AVIONICS SYSTEMS - The avionic systems for the Mach 6 turboramjet test vehicle are fundamentally the same as for the Mach 12 rocket vehicle, as discussed in Section 4.9. The principal differences are identified in the following paragraphs. Because no Staging Module tests were considered for the Mach 6 vehicle, the avionic support equipment required for staging is omitted.

5.9.1.1 (U) Mission Description and Operational Sequences - The Mach 6 research aircraft operates in a conventional takeoff and landing mode and follows a normal powered acceleration and climb to test Mach number and altitude (approximately 100,000 ft (30,500 m)). The aircraft can take off Holloman AFB in New Mexico, complete a straight line cruise at Mach 6 for 5 minutes, and then descend and land unpowered at Edwards AFB. Tests in the opposite direction can also be performed. The duration of the long distance flights is approximately 30 minutes. The pilot can perform visible flight throughout the entire mission.

(U) Unlike the Mach 12 vehicle, the Mach 6 aircraft is not air launched, is not required to accomplish VTO, nor handle the Staging Module.

(U) For missile launch tests, this vehicle will also take off within the Edwards AFB tracking range, accelerate to test speed and altitude, release the test missile, turn, and return to Edwards. This flight path is designed to allow the test missile to follow a ballistic path and impact within the Pacific Missile Range.

5.9.1.2 (U) Functional Requirements Description - The functional requirements of the avionic subsystems are essentially the same as for the Mach 12 vehicle. Some of the performance requirements may be somewhat less stringent, but the use of common equipment is a desirable system feature. The avionics subsystems are basically the same as those described in Section 4.9.1.3 and described by the functional block diagram of Figure 4-93. The instrumentation equipment is essentially the same as discussed in Section 4.9.1.4. The equipment summary of Section 4.9.1.5 will meet the requirements of the Mach 6 vehicle.

5.9.2 (U) MISCELLANEOUS SUBSYSTEMS - The electrical, hydraulic, and environmental control systems are described in the following paragraphs.

5.9.2.1 (U) Electrical and Hydraulic Systems - Electrical and hydraulic power are supplied by two (redundant) power distribution gear boxes driven by a single transfer gear box. Each power distribution gear box provides a power takeoff shaft to drive one 20 KVA Integrated Drive Generator (IDG) and one 60 gpm (3800 cc/sec), 3000 psi (2068 N/cm^2) hydraulic pump. Each power distribution gear box is driven by a power takeoff shaft from the transfer gear box, which in turn is driven by a gear train connected to the power source. Depending upon the flight mode, one of three power sources or combination of power sources are used aboard the test aircraft. During the turbojet flight mode, a power takeoff shaft from the turbojet engine is clutched to drive the transfer gear box. When the ramjet mode of operation is initiated, the high pressure, high temperature engine inlet air is bled off the inlet subsonic diffuser and used to drive an air turbine motor (ATM). The drive shaft of the ATM is clutched to the transfer gear box and when the turbojet power decreases the ATM power shaft takes over and powers the transfer gear box. After the ramjet fuel is expended, a chemically fueled auxiliary power unit (APU) is started and the APU power takeoff shaft is clutched into the transfer gearbox drive gear train. As the engine inlet air pressure and temperature decrease during the high speed glide, the ATM power decreases and the APU will take over and drive the transfer gear box. The APU is operated until the aircraft is landed and shut down. The three power drive systems insure the transfer gear box has power during all aircraft modes of flight from takeoff to shut down. The APU is fueled by LH_2 and uses LO_2 as the oxidizer. Both components are stowed in separate tanks adjacent to the APU.

(U) The estimated peak hydraulic and electrical power is 155 hp (115.6 Kw) with an average requirement of 100 hp (74.6 Kw) throughout the flight. The two 20 KVA generators furnish the electrical power for the avionics and test equipment. The two hydraulic pumps furnish the required hydraulic power for the hydraulic systems of the aircraft such as the flight controls, landing gears, and engine inlet ramp actuation and control system.

5.9.2.2 (U) Environmental Control System (ECS) - The ECS is a direct loop heat sink system which rejects heat through intermediate fluid transport loops to the cryogenic fuel system of the engine or APU as a heat sink. During the air turbine motor drive mode for electrical and hydraulic power and after the ramjet engine LH_2 fuel is echausted, the cryogenic fuel of the APU is used as a heat sink for the ECS. This condition requires the APU to be started when the ramjet engine LH_2 fuel supply is depleted. The ECS electrical hydraulic and avionic heat loads are transferred to the cryogenic fuels by heat exchangers located in the discharge lines of the fuel tanks. Cockpit temperature control and liquid cooling of avionics and test equipment are provided by this system. Cockpit pressurization is provided by separate supplies of LO_2 and LN_2 mixed in gaseous form and supplied to the cockpit under pressure to provide an acceptable environment during the entire flight.

5.10 WEIGHTS

(U) The results of the parametric studies conducted during Phase II have been incorporated into the Phase I 210 Mach 6 aircraft. This configuration was then refined to a vehicle size of S_P = 1103 ft^2 (102.5 m^2) and a takeoff gross weight of 61426 lbm (41450 kg). In Figure 5-50 a summary of the incremental weight changes is presented to show the progression in weight growth from Phase I to Phase III starting with the engine change and progressing to the larger vehicle structure required to maintain aircraft performance.

(U) FIGURE 5-50
WEIGHT SUMMARY – PHASE I TO PHASE III

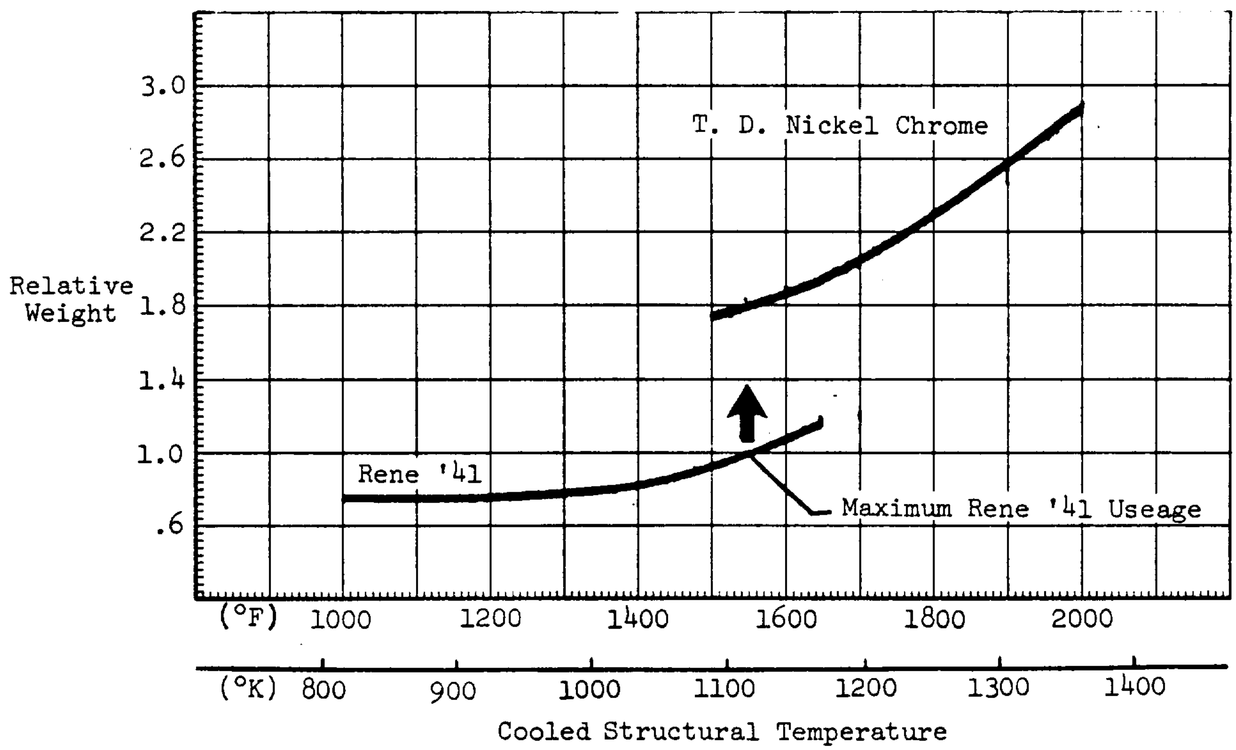
	lbm	kg
Operating Weight Empty, Phase I-210 Sp=895 ft^2 (83.2 m^2)	36450	16533
Engine	6804	3086
Air Induction	-1100	-499
Fuel System	656	298
Systems	300	136
Structure	(5376)	(2439)
Wing/Body	4100	1860
Tails and Landing Gear	1276	579
Operating Weight Empty, Phase III C210 Sp=1103 ft^2 (102.5 m^2)	48456	21993

5.10.1 (U) MACH 6 BASIC VEHICLE - The basic Mach 6 vehicle studied in Phase III is derived from the concept formulated for configuration 210 in the Phase I study effort. This concept is a manned, horizontal takeoff, wing body, turboramjet vehicle. The Phase I aircraft propulsion utilized a rubberized GE5/JZ6C all hydrogen stoichiometric engine sized to achieve thrust requirements for acceleration and 5 minutes of Mach 6 cruise time. The Phase III aircraft is a point design sized for 5 minutes of cruise with a JP/LH_2 fuel P&WA STRJ11A-27 turboramjet engine. This engine is basically a J58 JP fueled turbojet with a LH_2 fueled wraparound ramjet.

(U) The best inlet concept was determined through trade studies on various inlet designs and their integration to the aircraft. The three inlet configurations studied consisted of two inlet concepts which were located underneath the fuselage and one horizontal ramp, bifurcated, shoulder configuration. The two concepts carried underneath the body were a single inlet with horizontal ramps and a bifurcated vertical ramp inlet.

(U) During the inlet trade study, all design concepts investigated utilized passively cooled structure. For the Phase III study effort however, a regeneratively cooled inlet was used. This produced a small weight savings, as compared to hot inlet structure, and also enabled a growth in Mach number capability by being able to tailor the added heat load with the cooling fuel available. In future programs weight savings through regenerative cooling will play an important role in the overall weight of an aircraft. Figure 5-51 illustrates the relative weight variation for the primary inlet structure over the various temperature ranges investigated.

(U) FIGURE 5-51
AIR INDUCTION RELATIVE WEIGHT vs VARIOUS STRUCTURAL TEMPERATURES



The design cooling point of 1550°F (1116°K) for the regeneratively cooled inlets is the maximum for Rene' 41 usage. Maintaining lower temperatures would increase the fuel flow schedule and thus require a larger aircraft to obtain the increased fuel volume required.

(U) Figure 5-52 is the group weight statement for the Phase III aircraft. The structural weight is greater than that of the Phase I Mach 6.0 vehicle because the length and diameter of the P&W STRJ11A-27 engine are greater than the Phase I engine. The accommodation of this engine requires a longer aft fuselage with a larger fuselage cross section than the Phase I vehicle. When the aircraft was resized to maintain constant performance a planform area of 1103 ft^2 (102.5 m^2) was required. The increased planform area, 208 ft^2 (19.3 m^2) over the Phase I vehicle, requires growth in vertical tails to maintain stability. A larger, stronger landing gear is required for added vehicle size and strength requirements. Operation of larger, heavier vehicle requires more power and hence subsystems such as surface controls, hydraulics, and electronics must be increased, since they are a function of size and weight of the vehicle.

(U) FIGURE 5-52
GROUP WEIGHT SUMMARY – BASIC MACH 6 VEHICLE

Group	lbm	kg
Wing/Body	15900	7212
Rotating Tips	1231	558
Vertical Tails	2331	1057
Landing Gear - Nose	295	133
Main	1276	579
Surface Controls	598	271
Propulsion Group	(21621)	(9807)
Engine	11604	5263
Air Induction	8447	3831
Fuel System	1056	479
ADS	394	179
Controls	50	23
Lube and Cooling	70	32
Instruments	175	79
Hydraulics	553	251
Electrical	300	136
Electronics	715	324
Furnishings	400	182
ECS	250	113
Contingency	929	421
Weight Empty	46574	21123
Crew and Equipment	240	109
Payload	1290	585
Vented and Trapped Fuels	162	73
Fuel Pressurant	40	18
APU Fuel	150	68
Operating Weight Empty	48456	21976
Fuel - LH_2	4310	1955
JP	8660	3928
Takeoff Gross Weight	61426	27859

(U) Propellant tankage locations and sequencing were selected to maintain the vehicle c.g. forward of the aft aerodynamic c.g. limit throughout the flight profile. At the takeoff and landing conditions, where control requirements are the most critical, the aircraft c.g. is far aft as permissible for minimal control forces. The JP fuel is located in two wing tanks and one fuselage tank. Liquid hydrogen is located in two fuselage tanks. Figure 5-3 depicts the Mach 6.0 vehicle tank available for hydrogen which for fuel sequencing is partitioned at F.S. 470. The c.g. travel which, through fuel sequencing, spans but .8% body length, is as follows. The takeoff c.g. is at 66.8% body length. The vehicle c.g. is moved forward as far as possible as all JP fuel is expended, first from wing (2650 lbm (1202 kg)) and then from the fuselage tank, 6010 lbm (2726 kg), to arrive at the Mach 3.5 condition, 52,766 lbm (23,935 kg), at a c.g. of 66.2% body length . The aft hydrogen tank fuel, 1840 lbm (835 kg), and 800 lbm (363 kg) from the forward hydrogen tanks is expended to obtain the 62.3% body length c.g. at the Mach 6.0 begin cruise weight 50,126 lbm (22,735 kg). The c.g. moves aft as the remaining hydrogen, 1670 lbm (758 kg) is expended from the forward hydrogen tank with the vehicle landing c.g. at 67% body length, which is the aft aerodynamic limit.

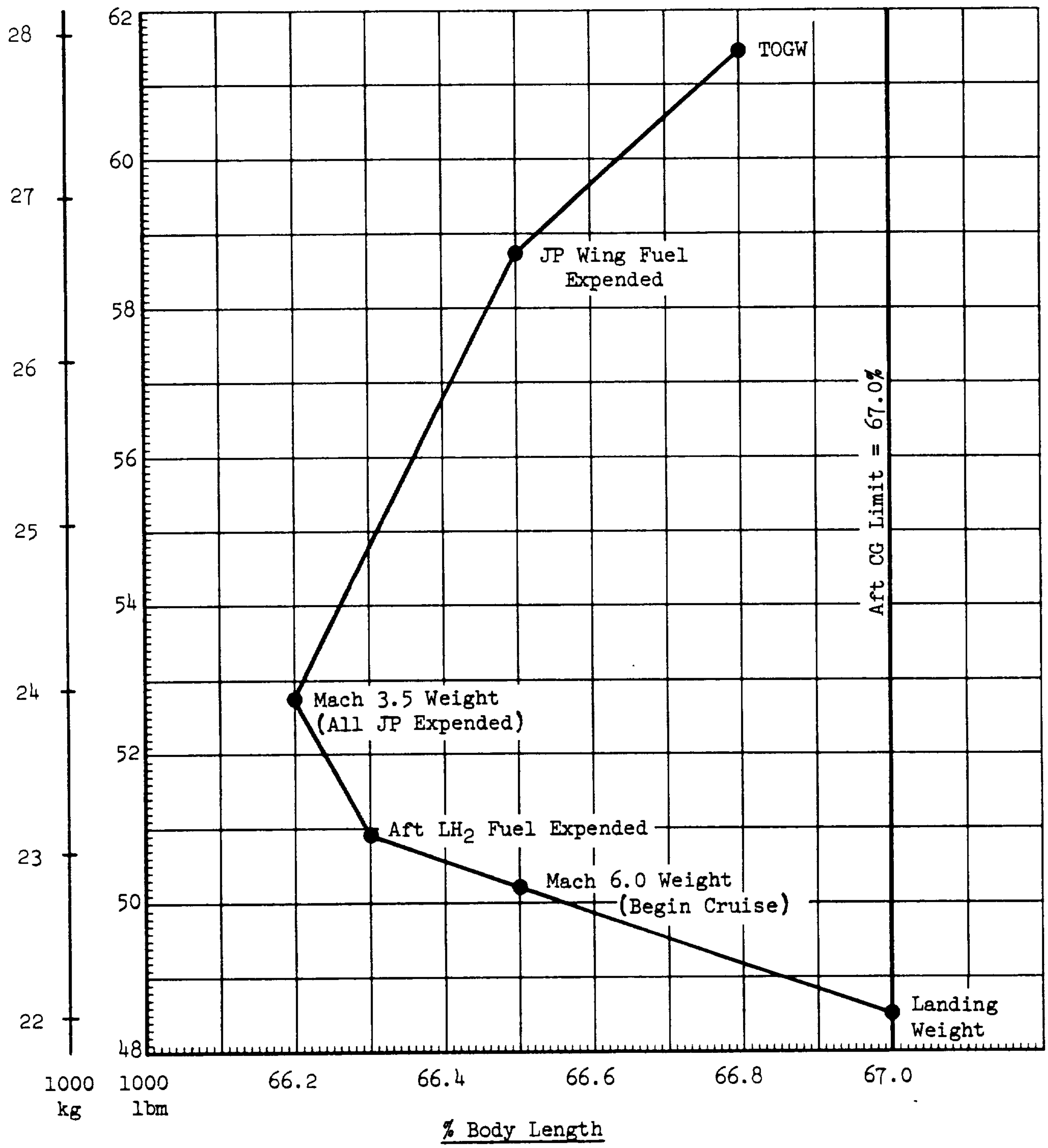
(U) Moments of inertia and principle axis results for the basic Mach 6 aircraft are presented below.

	Slug - ft^2		kg - m^2	
Condition	OWE	TOGW	OWE	TOGW
I_x Roll	22223	67990	30134	92194
I_y Pitch	464819	508733	630295	689842
I_z Yaw	511808	562767	694012	763112
I_{xz} Product	16233	18763	22012	25443
ϕ Principle Axis	1°54'	2°10'	1°54'	2°10'

(U) The Mach 6.0 aircraft uses protected primary (load carrying) structure because of its thermal environment. The passively insulated structural concept features radiation shingles on the vehicle surface with insulation in between the shingles and the primary structure. However, because of cooling problems, structural depth limitations, and general vehicle integration, the following structural components are designed as unprotected hot structures:

- Direct access engine doors and shrouds
- Rotating tips
- Vertical tails
- Air induction primary structure
- Air induction internal ramps.

(U) FIGURE 5-53
CENTER OF GRAVITY TRAVEL
Mach 6 Aircraft



These components are constructed of high temperature materials such as Rene' 41. Thermal stresses and attachment methods are calculated into the overall structure to account for "hot" and "cold" structural compatibility.

(U) A material breakdown for the main structural components, wing/body, rotating tips, vertical tails, landing gear, and air induction is presented in Figure 5-54.

(U) In summary, the Phase III Mach 6 aircraft concept is larger in size and weight than the Phase I design because of the larger and heavier STRJ11A-27 engine. However, the lower development costs of the P&WA STRJ11A-27 in the Phase III aircraft makes it more attractive. Of the total weight change, 12,006 lbm (5440 kg), the engine accounts for 6804 lbm or 3085 kg of the weight difference.

5.10.2 (U) MACH 6 DESIGN OPTION WEIGHTS - Several modification kits were proposed for the Mach 6 vehicles. Incremental weights were derived for both the installation provisions and the basic research package. Figure 5-55 summarizes the changes in operating weight and the empty balance effect on the basic Mach 6 configuration.

(U) ARMAMENT (ARM) OPTION - The ARM option concept utilizes a lower fuselage fairing consisting of the basic shell structure, insulation and shingles. An ablation material is applied to the inside walls of the fairing shell to protect the structure from the high temperatures and thermal shock that are incurred as the missile is fired. During missile ejection, the door is open for a cycle time of 30 seconds which allows temperatures to reach 2600°F (1716°K) inside the pod. The missile door is designed to sustain the added air loads that it is subjected to during its operating cycle. Actuation of the door is accomplished hydraulically. Weight for fuselage mounting provisions, electronic missile initialization, and thrust ejection provisions are added to complete the installation. Figure 5-56 summarizes the weight increments for the armament modification and its installation.

(U) THERMAL PROTECTION SYSTEM (TPS) OPTION - This modification consists of taking a 60 inch (152.5 cm) length of fuselage structure and designing it to accept various thermal protection systems. Systems or concepts with depths of up to 6 inches (15.25 cm) can be accommodated for test times consistent with the aircraft flight time. Each system will be designed as a removable package concept thereby necessitating installation provisions in the aircraft.

(U) The installation weight is calculated on a maximum thermal protection package weight of 200 lbm (91 kg). The actual installation weight of 136 lbm (62 kg) is for cutouts, kickload and equipment mounting loads that are encountered and is in addition to the 200 lbm(91 kg) package weight. 126 lbm (57 kg) is installed at the same time as the basic package. A tabular summation of the TPS modification is shown in Figure 5-57.

(U) FIGURE 5-54
MACH 6 PHASE III AIRCRAFT MATERIAL DISTRIBUTION – LBM

	Structure					Fixed Equipment			Thermal Protection					Miscellaneous Expendables	Total
									Shingles			Other			
	Aluminum Alloys	Titanium Alloys	Nickel Alloys	Columbium Alloys	Other	Running Gear	Hydraulic Equipment	Misc. Equipment	Titanium	Rene '41	T.D.NiCr	Nose Cap	Misc. Insulation		
Wing/Body	3905	3966	1160	282	520				289	2610	150	80	2938		15900
Rotating Tips			1231												1231
Vertical Tails			2331												2331
Nose Gear			144			91	60								295
Main Gear			639			487	150								1276
Controls							598								598
Engine – STRJ11A-27								11604							11604
Air Induction			7498				682						267		8447
Fuel System								1056							1056
Auxiliary Power System								394							394
Engine Controls								50							50
Lube & Cooling (Engine)								70							70
Instruments								175							175
Hydraulics							553								553
Electrical								300							300
Electronics								715							715
Furnishings	20							330					50		400
Environmental Control								250							250
Contingency	287	382	260												929
Weight Empty															**46574**
Crew and Equipment								240							240
Payload								1290							1290
Ullage and Vent Fuel														162	162
Pressurant														40	40
APU Propellant														150	150
Operating Weight															**48456**
Propellant – LH_2														4310	4310
Propellant – JP														8660	8660
Takeoff Gross Weight	4212	4348	13263	282	520	578	2043	16474	289	2610	150	80	3255	13322	**61426**

(U) FIGURE 5-54 (CONTINUED)
MACH 6 PHASE III AIRCRAFT MATERIAL DISTRIBUTION – Kg
(Presented in International Standard Units)

	Structure					Fixed Equipment			Thermal Protection					Miscellaneous Expendables	Total
									Shingles			Other			
	Aluminum Alloys	Titanium Alloys	Nickel Alloys	Columbium Alloys	Other	Running Gear	Hydraulic Equipment	Misc. Equipment	Titanium	Rene '41	T.D.NiCr	Nose Cap	Misc. Insulation		
Wing/Body	1771	1799	526	128	236				131	1184	68	36	1333		7212
Rotating Tips			558												558
Vertical Tails			1057												1057
Nose Gear			65			41	27								133
Main Gear			290			221	68								579
Controls							271								271
Engine – STRJ11A-27								5263							5263
Air Induction			3401				309						121		3831
Fuel System								479							479
Auxiliary Power System								179							179
Engine Controls								23							23
Lube & Cooling (Engine)								32							32
Instruments								79							79
Hydraulics							251								251
Electrical								136							136
Electronics								324							324
Furnishings	9							150					23		182
Environmental Control								113							113
Contingency	130	173	118												421
Weight Empty															**21123**
Crew and Equipment								109							109
Payload								585							585
Ullage and Vent Fuel														73	73
Pressurant														18	18
APU Propellant														68	68
Operating Weight															**21976**
Propellant – LH_2														1955	1955
Propellant – JP														3928	3928
Takeoff Gross Weight	1910	1972	6015	128	236	262	926	7472	131	1184	68	36	1477	6042	**27859**

(U) FIGURE 5-55
RESEARCH PACKAGE OPTIONS

Item	Installation Provisions				Combined Research Package			
	Weight		*ΔCG		Weight		*ΔCG	
	lbm	kg	in	cm	lbm	kg	in	cm
Armament	97	44	1	2.54	2192	994	2	5.08
Thermal Protection (TPS)	10	5	0	0	336	152	2	5.08
GE5/JZ6C Engine	**	-	-	-	-2312	-1048	3	7.62
Armament and TPS Combined	107	49	0	0	2528	1147	0	0

* CG change taken at the operating weight empty conditions.
** Not separately defined since the modification involves rework of the aft fuselage and no separate provisions are initially installed in the aircraft.

(U) FIGURE 5-56
ARM OPTION WEIGHT SUMMARY

Item	*Installation Provisions		Complete Research Package	
	lbm	kg	lbm	kg
Structure and Heat Protection	61	28	654	297
Electronics	13	6	63	29
Armament Group	23	10	130	59
Hydraulics	-	-	10	5
Subtotal	97	44	857	390
Missile (useful load)	-	-	1335	606
Total	97	44	2192	996

* Included in research package weight.

(U) FIGURE 5-57
TPS OPTION WEIGHT SUMMARY

Item	Installation Provisions Only		Total Research Package	
	lbm	kg	lbm	kg
Structure				
. Equipment Bay	-	-	126	57
. Equipment Bay Provisions	10	5	10	5
TPS Equipment	-	-	200	91
Total	10	5	336	153

(U) ALTERNATE ENGINE (JZ6) - The stoichiometric GE5/JZ6C engine is the basis for this modification. The physical geometry of the GE5/JZ6C is smaller than the P&WA STRJ11A-27 and the capture area requirement is less for its air induction system. Installation of the engine entails the reworking of the rear fuselage. Since the GE5/JZ6C engine is an all hydrogen burning design, the JP fuel system is replaced with a LH_2 system. Figure 5-58 presents the weight changes involved in installing the GE5/JZ6C engine.

(U) FIGURE 5-58
JZ6 OPTION WEIGHT SUMMARY

Item	Weight	
	lbm	kg
Δ Engine Provisions	-463	-210
Δ Basic Structure	1149	522
Δ Engine	-4434	-2010
Δ Air Induction	1795	815
Δ Fuel System	- 359	-163
Total Δ Weight	-2312	-1046

(U) The basic aircraft does not contain adequate fuel volume to provide 5 minutes of test time at Mach 6. Therefore, an external fuel tank, designed for liquid hydrogen fuel, is installed. Thus, in addition to the previous changes, an additional 500 lbm (227 kg) is added for the tank and its installation on the aircraft. The capacity of the tank is 236 ft^3 (6.7 m^3) or 1100 lbm (499 kg) of hydrogen. All of the external fuel is consumed by the time the aircraft climbs to 20,000 ft (6100 m) at Mach 0.8 and the tank is then dropped. The incremental weight is shown in Figure 5-59.

(U) FIGURE 5-59
EXTERNAL FUEL TANK INSTALLATION

Item	Weight	
	lbm	kg
Initial Engine and Installation Change	-2312	-1048
External Tank and Installation	500	227
Final Increment to OWE	-1812	- 821
Base OWE	48456	21976
Resulting OWE	46644	21155

(U) COMBINED OPTIONS - The Phase III aircraft was designed to have the capability to incorporate both the ARM and TPS modifications if required. In this case, both of the systems installation provisions must be incorporated in the basic vehicle. Figure 5-60 lists the overall combination of installations.

(U) FIGURE 5-60
ARM AND TPS MODIFICATIONS COMBINED INSTALLATION WEIGHT

Item	Modification			
	Armament		TPS	
	lbm	kg	lbm	kg
Fuselage Armament Provisions	61	28	-	-
Armament Ejector System Provision	23	10	-	-
Electronics Equipment Installation	13	6	-	-
Equipment (Test) Bay Provisions	-	-	10	5
Subtotal	97	44	10	5
Total Combination	107 lbm		49 kg	

5.11 COSTS

(U) Total system costs were derived for the Mach 6 vehicle and several configuration options designed to accomplish research in areas of specific interest. The acquisition cost for the basic Mach 6 vehicle is 398M dollars, while the 5 year operating cost for 200 flights is 92M dollars. On a per flight basis, the operating cost is 46M dollars respectively. The acquisition costs represent approximately 81% of the total system cost which is 490M dollars.

(U) The costs associated with the selected configuration options were developed in the same manner as described in Section 4.11 while the cost format employed is the same as the one used for the Mach 12 basic vehicle. The Mach 6 vehicle does not require a launch platform; hence, the cost associated with this element has been eliminated from the investment cost category.

(U) The development of the basic vehicle costs with an explanation of the important cost factors is presented in the following section and is followed by development of the costs associated with the research options.

5.11.1 (U) BASIC VEHICLE COSTS - The Mach 6 vehicle costs are shown in tabular and bar chart form in Figure 5-61 and were derived in accordance with the methods, data and CER's used in the development of the Mach 12 vehicle costs discussed in Section 4.11.1.

(U) The refurbishment cost development is discussed in Section 3.6.2 and is 0.64% of the flight research vehicle's investment cost shown in Item II-1 of Figure 5-61. This percentage is based on the refurbishment cost per flight as a percentage of the flight vehicle's investment cost.

(U) The major cost segments of the Mach 6 program cost are depicted in bar chart form shown in Figure 5-61 together with their respective percentages of their major cost categories. The airframe is the dominate cost in all three major categories of costs. This was also true for the Mach 12 vehicle.

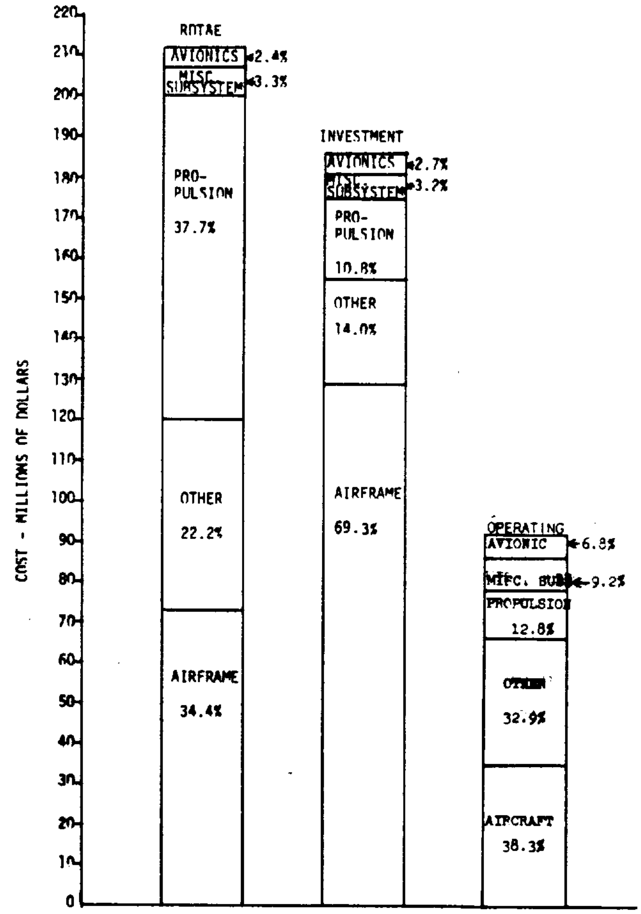
5.11.2 (U) PROGRAM COST INFLUENCES - A discussion of the factors which influence the total program cost of the Mach 6 vehicle is presented in the following sections. These factors are the same as those discussed in Seciton 4.11.2 All of the factors presented exert essentially the same impact on the total program cost for the Mach 6 vehicle as for the Mach 12 vehicle with the exception of the propulsion system for the Mach 6 vehicle which is approximately 3 times more expensive than the propulsion system for the Mach 12 vehicle. This is due to the type of propulsion system employed by the Mach 6 vehicle which is a turboramjet propulsion system.

5.11.2.1 (U) Airframe Cost Influence - The RDT&E, investment and operating costs associated with the airframe represent 48% of the total system cost of the Mach 6 flight research vehicle. As was the case for the Mach 12 vehicle, the airframe is the dominant factor in the total program cost.

(U) FIGURE 5-61
TOTAL PROGRAM COST SUMMARY AND COMPARISON FOR BASIC MACH 6 VEHICLE
(200 Flight Program – 5 Year Duration)
Total Program Cost = 490 $\bar{M}$ Dollars

Cost Categories and Elements	Basic Vehicle Costs (1)
I. RDT&E Costs	
1. Airframe Design and Development	
A. Airframe Design	56.5
B. Miscellaneous Subsystem Design & Development	7.1
C. Development Tests (Including Wind Tunnel)	4.1
D. Test Hardware	8.8
E. Pre-Delivery Flight Test	3.3
Sub-Total	79.8
2. Tooling	19.8
3. Avionics Development	4.7
4. Propulsion Development	80.0
5. Support Equipment Design & System Integration	27.4
6. Ground Test Facilities	-
Total	211.7
II. Investment Costs	
1. Flight Vehicles	
A. Airframe	43.0
B. Miscellaneous Subsystems	1.8
C. Propulsion	2.0
D. Avionics	1.4
Unit Cost (1) Vehicle	48.3
Unit Cost (3) Vehicles	144.8
2. Support Costs	
A. AGE	21.7
B. Training Equipment	0.8
C. Initial Stocks (Engines & AGE Spares)	16.2
D. Initial Training	1.0
E. Initial Transportation	1.7
Sub-Total	41.4
3. Launch Platform Cost	-
Total	186.2
III. Operating Cost	
1. Range User Cost	5.6
2. Escort Aircraft & Logistics	2.2
3. Vehicle Maint./Repair Cost	
A. Airframe	
a. Material	25.8
b. Labor	9.4
B. Miscellaneous Subsystems	
a. Material	8.0
b. Labor	.4
C. Propulsion Systems	
a. Material	9.1
b. Labor	2.7
D. Avionics	
a. Material	5.4
b. Labor	.9
Total Maintenance/Repair Cost	61.7
4. Propellant and Pressurant Cost	0.4
5. AGE Maintenance Cost	2.5
6. General Purpose Maintenance Support	1.0
7. Transportation Cost	-
8. Pilot Pay & Support Personnel Pay	18.6
9. Launch Platform Operation Cost	-
Total Operating Cost	92.0
Grand Total	489.9

(1) Costs In Millions of 1970 Dollars



(U) Approximately 60% of the Mach 6 vehicle's airframe structure is made up of advanced materials while the corresponding percentage for the Mach 12 vehicle is 49%. Thus, the percentage of advanced materials exerts a greater degree of influence on the Mach 6 airframe's weighted production complexity factor than on the Mach 12 airframe. The weighted production complexity factor is 4.0073 for the Mach 6 vehicle whereas it is 3.9015 for the Mach 12 vehicle. Another important parameter that influences the acquisition cost of the Mach 6 vehicle is the DCPR weight. The DCPR weight of the Mach 6 vehicle is approximately twice the DCPR weights of the Mach 12 vehicle. The airframe structure for the Mach 6 vehicle accounted for 91% of the DCPR weight whereas the shingles accounted for the remaining 9% of the DCPR weight. The comparable percentages for the Mach 12 vehicle are 86% and 14% respectively.

(U) The DCPR weight composition and the development of the weighted production complexity factor for the Mach 6 vehicle are shown in Figures 5-62 and 5-63.

(U) FIGURE 5-62
DCPR WEIGHT COMPOSITION – MACH 6 CONFIGURATION

Weight

Material Type	Basic Structure		Equipment		Total	
	lb	kg	lb	kg	lb	kg
Aluminum	4,212	1,910			4,212	1,910
Titanium	4,637	2,103			4,637	2,103
Steel	3,564	1,616			3,564	1,616
Rene 41	11,957	5,422			11,957	5,422
TDNiCr	502	226			502	266
Columbium	282	128			282	128
Nose Cone	80	36			80	36
Insulation	3,255	1,476			3,255	1,476
Other	520	236			520	736
Equipment	-	-	4,380	1,986	4,380	1,986
Total	29,009	13,155	4,380	1,986	33,389	15,142

(U) FIGURE 5-63
PRODUCTION COMPLEXITY - AIRFRAME

Material	DCPR Wt.		% of DCPR Wt.	Complexity	Weighted Complexity
	lb	kg			
Advanced					
1)Columbium	282	126	.97	4.00	3.88
2)T.D.Nickle	502	226	1.73	9.00	15.57
3)Rene 41	11,957	5,422	41.22	7.50	309.15
4)Titanium	4,637	2,103	15.98	2.00	31.96
5)Nose Cone	80	36	.28	1.25	.35
Sub-Total	17,458	7,917	60.18		360.91
Conventional					
1)Insulation	3,255	1,476	11.22	1.00	11.22
2)Aluminum	4,212	1,910	14.52	1.00	14.52
3)Other	520	236	1.79	1.00	1.79
4)Steel	3,564	1,616	12.29	1.00	12.29
Sub-Total	11,551	5,238	39.82		39.82
Total	29,009	13,155	100.00		400.73
Total/100					4.0073

5.11.2.2 (U) Engine Cost Influence - The propulsion development cost for the STRJ11A-27 turboramjet is 80 M dollars while the investment cost is 2 M dollars for one engine. Acquisition costs attributed to the propulsion system accounted for 100 M dollars which is 25% of the total acquisition cost. The propulsion system exerts a greater influence on the acquisition cost for the Mach 6 vehicle than for the Mach 12 vehicle. This is due to the fact that the Mach 6 vehicle employs an airbreathing propulsion system whereas the Mach 12 vehicle employs a rocket propulsion system. There are 10 engines required in the Mach 6 vehicle program, 3 installed and 7 spares. Currently, there are a number of J-58 engines available in the government inventory (Air Force and NASA). The J-58 overall line at the Pratt and Whitney Aircraft facility in West Palm Beach, Florida is presently in operation.

5.11.2.3 (U) Miscellaneous Subsystems and Avionics System Cost Influences - Acquisition costs for the avionics and miscellaneous subsystems are not significant as was the case for the Mach 12 vehicle. This is due to the use of off-the-shelf hardware. The total acquisition cost for these two systems amounted to 22 M dollars.

5.11.2.4 (U) Operating Cost Influences - The maintenance/repair cost is the largest of all the operating cost elements and is 61.8 M dollars. This cost accounts for 67% of the total operating cost and is shown in summary form in Figure 5-64.

(U) FIGURE 5-64
MACH 6 VEHICLE MAINTENANCE/REPAIR COST SUMMARY
(200 FLIGHTS)

Major System	Cost-Millions of Dollars		
	Material	Labor	Total
1. Airframe	25.8	9.4	35.2
2. Propulsion	9.1	2.7	11.8
3. Avionics	5.3	1.0	6.3
4. Misc. Subsystems	8.0	.4	8.4
Total	48.2	13.5	61.7
% of Total	78.0	22.0	100.0

The operating cost for the Mach 6 vehicle (92.0 $\overline{M}$ dollars) ammortized over 200 flight research program is .460 $\overline{M}$ dollars per flight.

5.11.3 (U) RESEARCH VEHICLE OPTIONS - Three major options were priced for the Mach 6 basic vehicle and are as follows:

(a) ARM - Armament
(b) TPS - Thermal Protection System
(c) JZ6C - LH_2 Turboramjet Engine

The Mach 6 vehicle option costs are shown in Figures 5-65 and 5-66. It can readily be seen that options 1 and 2 add a small increment to the basic vehicle's program cost. However, option 3 requires a substantial increment to the basic vehicle's program cost which is due to the engine development cost of 647 $\overline{M}$ dollars.

(U) The costs associated with the installation provisions were derived on the basic of installing them at the contractor's facility and at the flight research center. The last column in Figures 5-65 and 5-66 shows the costs of installing the installation provisions for options 1 and 2 at the contractor's facility. The difference in cost between installation at the contractor's facility and at the flight research center is insignificant. Options 1 and 2 add approximately 11 $\overline{M}$ dollars to the basic vehicle's total program cost which is approximately 2% of stated cost. The magnitude of the cost increments related to the basic vehicle's program cost incurred by the two options is shown in the bar chart in Figure 5-67.

(U) It was found that no additional refurbishment costs would be incurred by adding options 1 through 3 to the basic Mach 6 vehicle.

(U) The cost to install the options is shown in Figure 5-68. All costs are based on installation at the flight test center by NASA personnel.

(U) FIGURE 5-65
DETAILED TOTAL SYSTEM COST BREAKDOWN FOR BASIC MACH 6 VEHICLE AND ASSOCIATED OPTIONS (MILLIONS OF 1970 DOLLARS)

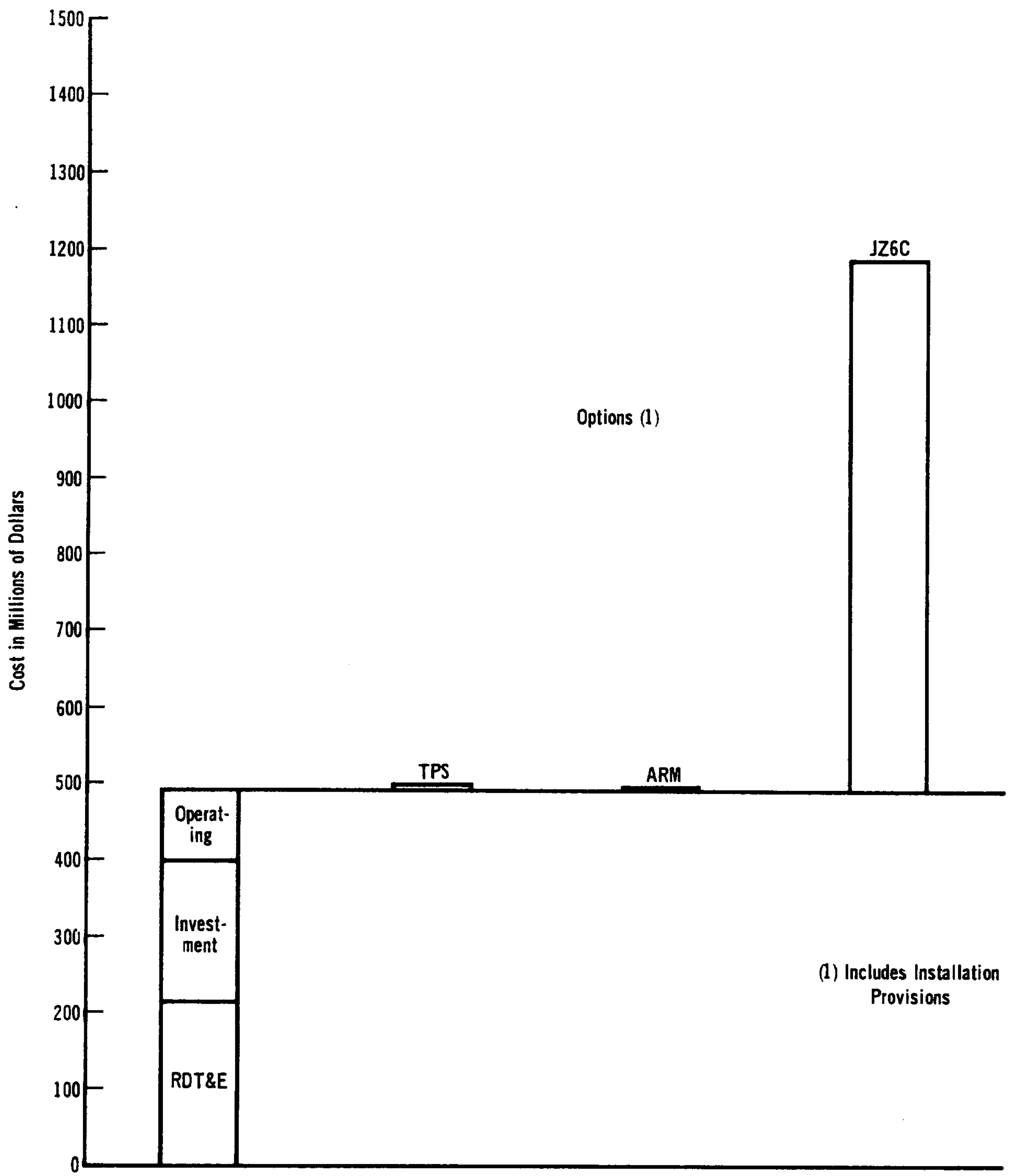
Cost Categories and Elements	Basic Vehicle Cost (1)	Options						Installation Provisions for Options 1 and 2
		Option No. 1 ARM		Option No. 2 TPS		Option No. 3 JZ6C		
		Installation Provisions	Research Package	Installation Provisions	Research Package	Installation Provisions	Research Package	
I. RDT&E Costs								
1. Airframe Design and Development								
A. Airframe Design	56.471	0.165	1.401	0.020	0.811	0.229	11.790	0.183
B. Misc. Subsystem Design and Development	7.147	0.029	0.050	-	-	-	0.195	0.029
C. Development Tests (Including Wind Tunnel)	4.063	0.005	0.072	0.001	1.312	0.017	2.405	0.006
D. Test Hardware	8.833	0.016	0.113	0.002	0.452	0.028	2.813	0.018
E. Pre-Delivery Flight Test	3.334	-	-	-	-	-	-	-
Sub-Total	79.848	0.215	1.636	0.023	2.575	0.274	17.203	0.236
2. Tooling	19.755	0.015	0.520	0.007	0.996	0.093	5.133	0.055
3. Avionics Development	4.669	-	-	-	-	-	-	-
4. Propulsion Development	80.000	-	-	-	-	-	500.000	-
5. Support Equipment Design & System Integration	27.401	0.047	0.705	-	0.094	0.235	0.940	0.047
6. Ground Test Facilities	-	-	-	-	-	-	147.085	-
Total	211.673	0.277	2.861	0.030	3.665	0.602	670.366	0.338
II. Investment Costs								
1. Flight Vehicles								
A. Airframe	42.963	0.075	0.699	0.012	2.262	0.139	14.038	0.087
B. Miscellaneous Subsystem	1.849	0.009	0.014	-	-	-	0.034	0.009
C. Propulsion	2.000	-	-	-	-	-	3.500	-
D. Avionics	1.446	-	-	-	-	-	-	-
Unit Cost (1) Vehicle	48.258	0.084	0.713	0.012	2.262	0.139	17.572	0.096
Unit Cost (3) Vehicles	144.774	-	-	-	-	-	-	0.288
2. Support Costs								
A. AGE	21.716	-	0.107	-	0.339	-	2.636	-
B. Training Equipment	0.750	-	-	-	-	-	-	-
C. Initial Stocks (Engines & AGE Spares)	16.176	-	0.011	-	0.034	-	0.614	-
D. Initial Training	1.000	-	-	-	-	-	-	-
E. Initial Transportation	1.746	-	0.002	-	0.007	-	0.065	-
Sub-Total	41.388	-	0.120	-	0.380	-	3.315	-
3. Modification Installation Cost	-	-	0.042	-	0.019	-	-	-
Total	186.162	0.084	0.875	0.012	2.661	0.139	20.887	0.288
III. Operating Cost (200 Flights - 5 Yrs.)								
1. Range User Cost	5.566	-	-	-	-	-	-	-
2. Escort Aircraft and Logistics	2.220	-	-	-	-	-	-	-
3. Vehicle Maintenance Repair Cost								
A. Airframe								
a. Material	25.778	-	-	-	-	-	-	-
b. Labor	9.450	-	-	-	-	-	-	-
B. Miscellaneous Subsystems								
a. Material	8.039	-	-	-	-	-	-	-
b. Labor	.405	-	-	-	-	-	-	-
C. Propulsion Systems								
a. Material	9.104	-	-	-	-	-	-	-
b. Labor	2.700	-	-	-	-	-	-	-
D. Avionics								
a. Material	5.344	-	-	-	-	-	-	-
b. Labor	.945	-	-	-	-	-	-	-
Total Maintenance Repair Cost	61.765	-	-	-	-	-	-	-
4. Propellant and Pressurant Cost	0.398	-	-	-	-	-	-	-
5. AGE Maintenance Cost	2.498	-	0.016	-	0.051	-	-	-
6. General Purpose Maintenance Support	1.000	-	-	-	-	-	0.395	-
7. Transportation Cost	-	-	-	-	-	-	-	-
8. Pilot Pay & Support Per Pay	18.600	-	-	-	-	-	-	-
9. Launch Platform Operation Cost	-	-	-	-	-	-	-	-
Total Operating Cost	92.047	-	0.016	-	0.051	-	0.395	-
Grand Total	489.882	0.361	3.752	0.042	6.377	0.741	691.648	0.626

(1) Millions of 1970 Dollars

(U) FIGURE 5-66
TOTAL SYSTEM COSTS FOR THE MACH 6 CONFIGURATION AND ASSOCIATED OPTIONS
(Millions of 1970 Dollars)

COST CATEGORIES	BASIC VEHICLE	OPTIONS		INSTALLATION PROVISIONS FOR OPTIONS 1 & 2	OPTION 3			BASIC VEHICLE PLUS INSTALLATION PROVISIONS FOR OPTIONS 1 & 2
		1. TPS	2. ARM		RESEARCH PACKAGE	INSTAL. PROV.	TOTAL	
1. RDT&E	211.7	3.7	2.9	.4	670.4	.6	671.0	212.1
2. INVEST-MENT	186.2	2.7	.8	.3	20.9	.2	21.1	186.5
3. OPERAT-ING	92.0	-	-	-	.4	-	.4	92.0
TOTAL	489.9	6.4	3.7	.7	691.7	.8	692.5	490.6

(U) FIGURE 5-67
OPTION COST COMPARISON SUMMARY



(U) FIGURE 5-68
OPTION INSTALLATION COSTS

Options	Time (Mo's)	No. of Men	Hr/Rate $/Man Hr(1)	Total Man Hrs (2)	Total Opt. Inst. ($)
TPS	2	8	6.66	2,768	18,500
ARM	3	12	6.66	6,228	41,500
JZ6C	6	12	6.66	12,456	83,000

(1) Based on $13,000/yr (1970) NASA shop personnel cost and 1,952 working hrs per yr.
(2) Based on 173 manhours per month.

5.12 DEVELOPMENT SCHEDULE

(U) The development schedule for the Mach 6 flight research vehicle is presented in Figure 5-69 which is a milestone chart depicting the important events and their time of occurrence measured from "go-ahead".

(U) The schedule reflects the cost ground rules stated in Section 3.6.1.1; that is, (1) minimum cost-to-fly program, (2) soft tooling, (3) limited reliability program, (4) "zero defects" program not employed, (5) five year operational test program, (6) limited pre-delivery flight test program and (7) maximum use of existing equipment.

(U) The time allocated for the engineering design of the airframe is 36 months with design freeze occurring 12 months after "go-ahead". Development tests are conducted for a period of 18 months and terminate at the mid-point of the airframe engineering design phase. Aircraft No. 1 undergoes a series of taxi tests during the 39th month after "go-ahead" which lasts for a period of 1 month. Taxi tests are required for HTO (Horizontal-Take-Off) aircraft.

(U) Three major categories of tests are performed during the 18 month development test period; namely, (1) wind tunnel tests, (2) structural and thermal tests and (3) component and sub-systems tests. The assembly of the structural test article begins 16 months after "go-ahead" and is completed 27 months after "go-ahead". Assembly of the three flight research vehicles begins 18 months after "go-ahead" and terminates with the completion of the third vehicle 52 months after "go-ahead".

(U) Due to the use of soft tooling, the tool design and fabrication program is only 13.5 months in duration.

(U) The pre-delivery flight test program is 11 months in duration. Only two of the three flight research vehicles participate in the pre-delivery flight test program. The first flight occurs at the beginning of the 40th month after "go-ahead". Vehicle No. 2 participates in the program during the last four months of the program.

(U) Delivery of the first two flight research vehicles to NASA occurs 50 months after "go-ahead" while the delivery of the third vehicle occurs two months later.

(U) The flight research program is 5 years in duration during which 200 missions are flown. The program starts 51 months after go-ahead and terminates 5 years later.

(U) The pre-delivery flight test phase consists of 18 powered flights in which the envelope is expanded to Mach 3-3.5 to verify the ramjet operation. For the envelope expansion phase, 44 powered flights are scheduled to expand the Mach number envelope from 3 to 6. In this phase, no tests are conducted other than those associated with the expansion of the Mach number. The 44 flights consist of: (1) 20 flights allocated for inlet development, (2) 20 flights allocated for expansion of q with Mach number and (3) 4 flights allocated for maneuvering tests. During the research phase, 38 flights are scheduled to be flown by the basic vehicles in which the test time is varied with Mach number. All flights conducted by the Mach 6 vehicles are powered flights. There are 100 flights allocated for the modification phase of the research program.

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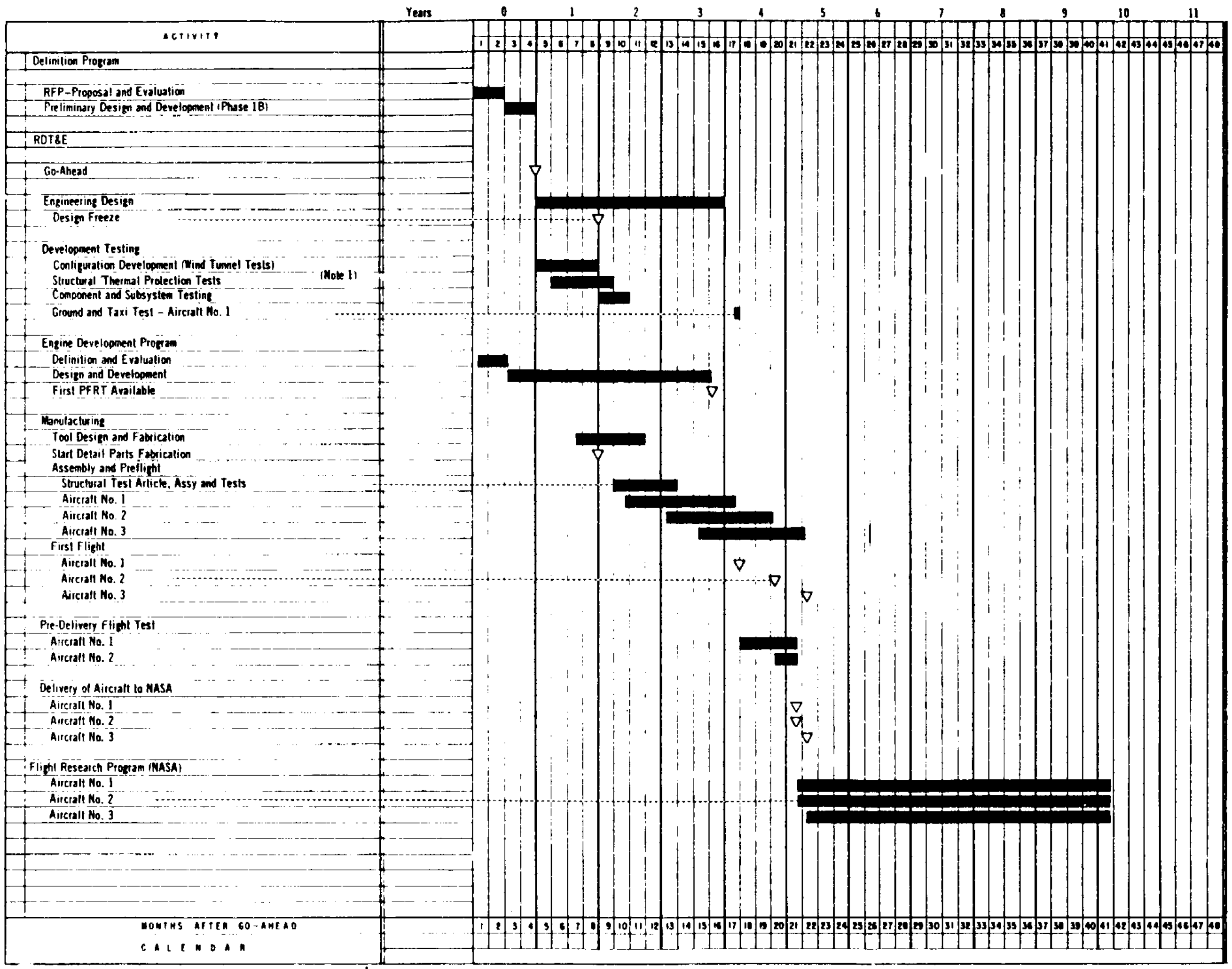
(U) FIGURE 5-69
MACH 6 VEHICLE
DEVELOPMENT SCHEDULE



Note (1)
Includes Fuel
System Integral Tank
Design and Development

FOLDOUT FRAME 1

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FOLDOUT FRAME 2

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6. MISSION STUDIES

(U) This section presents the mission studies for the two basic research vehicles and the modification option versions. Volumes II and III discussed some of the aspects related to mission studies for the various candidate hypersonic research vehicles, and identified other areas for future consideration. The results reported here reflect: (1) a refinement in vehicle design and performance; (2) a more detailed analysis of research missions; (3) investigation of areas previously identified for further study; and (4) treatment of some new areas.

(U) Basing and test ranges are recommended which can accommodate a wide spectrum of flight profiles during the speed buildups to the maximum design missions for the basic as well as the optional vehicle configurations. The Mach 12 vehicle mission will require test ranges up to 1900 nm (3520 km) in length which will span the continental United States from Florida to California as depicted in Figure 6-1. Test ranges were selected across the southern portion of the country in order to take advantage of the best year round weather conditions, less population density, existing tracking and communication stations, and available emergency landing sites. The Mach 6 vehicle mission, illustrated in Figure 6-2, with its less demanding range requirements can be operated in the southwestern area where the weather and population density conditions are even more attractive.

(U) The Mach 12 vehicle has good mission flexibility because of the airlaunch mode of operation. Three main operating bases can be used to accommodate many launch sites encompassing a wide variety of test missions. Flight operations with the Mach 6 vehicle will be more conventional because of the turbojet engine and HTO mode of operation. This vehicle has good single base mission potential, particularly for its lower speed flights, if additional JP fuel is provided to extend the outbound cruise range. This subject is treated in more detail in Section 6.1.

(U) Candidate emergency landing sites, identified during earlier studies, were re-evaluated to determine their suitability to the Phase III vehicles, mission profiles, and test ranges. Study results show that the research vehicles have a high probability of reaching an emergency landing site from any point in the flight profile.

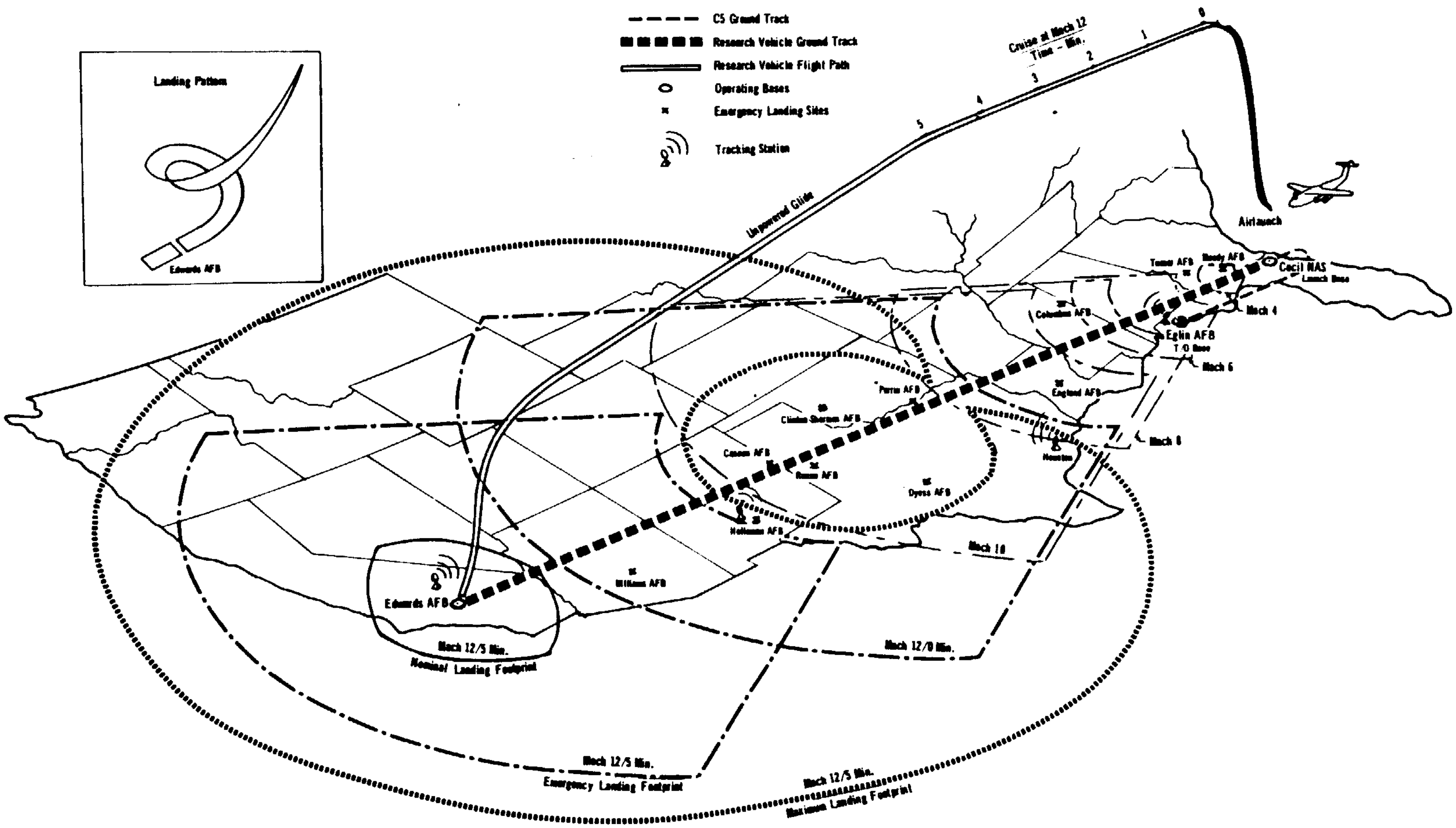
(U) Pilot escape and survivability were included in the mission studies. Various escape concepts were evaluated with respect to the vehicle designs and mission profiles, including the research vehicle optional configurations. The study results indicated that the open ejection seat afforded the most reasonable concept considering the probable ejection envelope, simplicity of design, experience level, and cost factor.

(U) The following subsections discuss these and other considerations pertaining to the vehicle flight operational program.

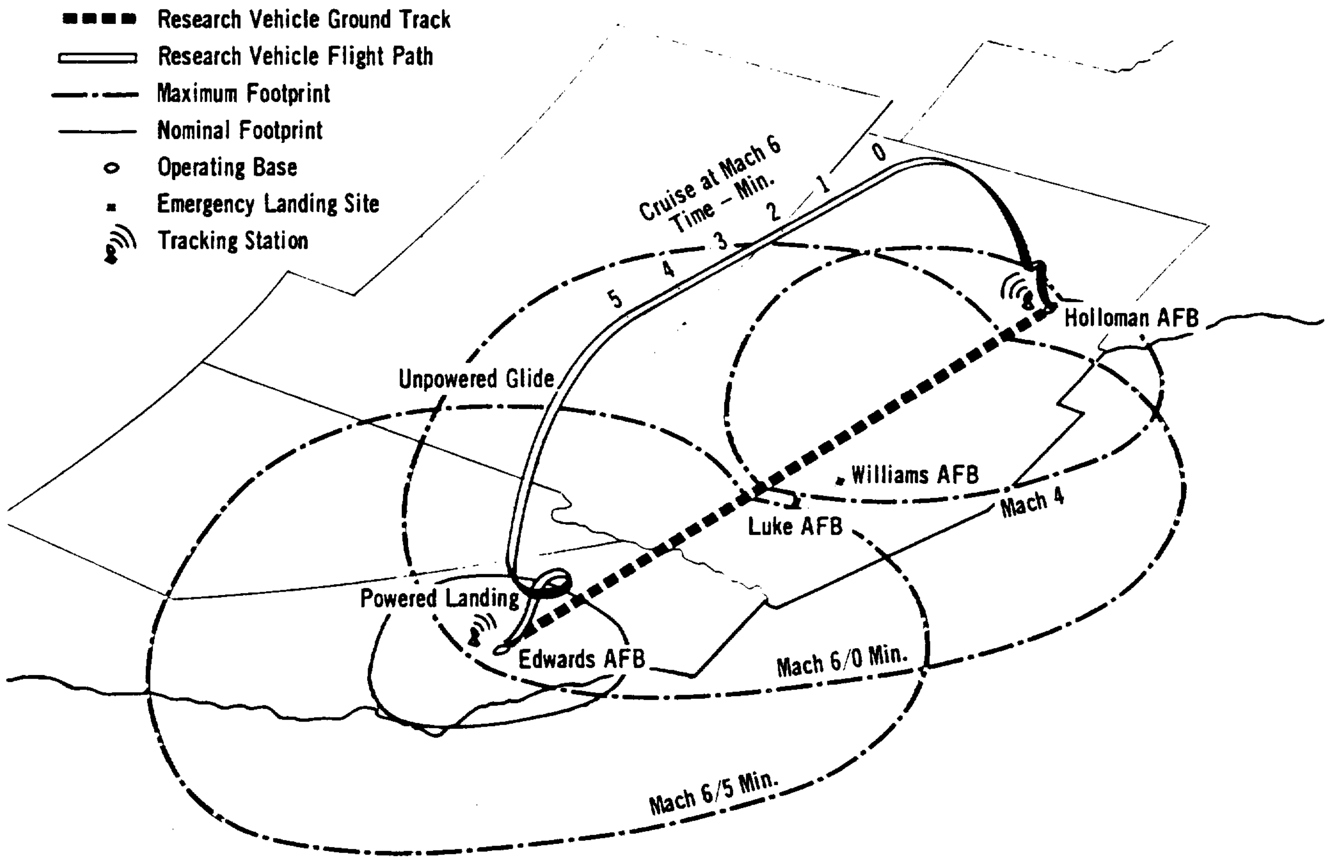
6.1 BASING

(U) Preliminary basing studies performed during Phases I and II established the basing and operating philosophies for the HYFAC research vehicle concepts under consideration. The Phase III basing study involved the investigation of the requirements for the two final candidate vehicles to conform to the established philo-

(U) FIGURE 6–1
MACH 12 VEHICLE DESIGN MISSION



(U) FIGURE 6-2
MACH 6 VEHICLE – TYPICAL MISSION



sophies, including the ramification of testing research options. These requirements were:

- o Define the minimum number of bases and test ranges capable of accommodating incremental expansion of the flight envelope and design mission.
- o Permit test missions to be flown inbound toward the landing site using a nominal glide range without the requirement for high Mach number turn.
- o Provide good emergency landing sites along the test range.
- o Permit maximum utilization of existing facilities (maintenance, tracking, communications, etc).

These requirements were met for all the research missions except for the Mach 12 and Mach 6 armament modes which could require a high Mach turn to return to base. The following paragraphs present the results of the basing study.

6.1.1 (U) MACH 12 VEHICLE - The recommended basing and operating plan for the Mach 12 - all rocket - air launched vehicle is essentially the same as that reported in Phase II, i.e., perform initial flights on the X-15 test range at Edwards AFB followed by two-base operations as the requirement for increased Mach No. and/or test time dictate. Some of the bases and test ranges recommended for certain missions in Phase II have been revised reflecting the results of the refinement of the vehicle design and performance. The nominal range of the 5 minute Mach 12 mission which previously required approximately 2,000 nm (3710 km) has been revised in Phase III to 1,900 nm (3520 km). This range is best accommodated by launching near Cecil NAS which is in northeast Florida, approximately 6 nm (11 km) southwest of Jacksonville, Florida and is located in restricted zone (R-2903A). Facilities include a 12,500 ft (3800m), hard surface runway, surveillance and precision radar, and conventional navigation aids (VOR, TACAN, UHF/DF). This base appears more attractive than Homestead AFB, Florida which previously was recommended as the candidate launch site for this mission, and provides an improved situation in regards to available emergency landing sites.

(U) The recommended basing and operating plan for the Mach 12 research aircraft shown in Figure 6-3 is designed to accommodate test missions over a wide range of test Mach numbers and test times. Test ranges that offer mission versatility are necessary in order to incrementally expand the flight envelope, and to obtain research data throughout the flight envelope. The basing and operating plan selections were based on the following considerations:

- o Edwards AFB, California is the main base of operations and the intended landing site for all missions.
- o Staging bases, from which the C-5A with the test vehicle aboard takes off, were selected on the basis of available facilities, personnel familiar with test operations, and location both with respect to remoteness and within nominal ferry range to suitable drop zones for various missions.
- o Air launch sites were selected on the basis of providing a suitable landing site should the mission be aborted immediately following the drop, remoteness of location, and the availability of emergency landing sites along the ground track between the drop point and Edwards AFB.

(U) The basing plan shows that the air launch mode of operation gives flexibility to mission planning and test range utilization. Three main staging bases can accommodate many launch sites and provide test missions over a wide range of Mach numbers and/or test time. The three recommended staging bases are all Air Force test facilities.

(U) In general, the recommended bases and test ranges of the basic vehicle will also accommodate missions with the vehicle optional configuration installed. With some mods all missions can be flown using a single base operations; other vehicle options require the full complement of test ranges. The recommended basing plans for the vehicle options, also shown in Figure 6-3, are discussed in the following paragraphs.

(U) FIGURE 6–3
MACH 12 CANDIDATE BASING AND TEST RANGE PLAN

Test Config.	Staging Base Ⓐ	Airlaunch Site Ⓑ	Ferry Range Ⓐ to Ⓑ n mi (km)	Test Range Ⓑ to Edwards n mi (km)	Test Range Mission Capability Mach No.	Time Min.
Basic and TPS Mod	Edwards AFB	Hidden Hills (Dry Lake)	110 (204)	110 (204	2 4	3.5 0
		Delmar (Dry Lake)	200 (370)	200 (370)	3 4 5	4.5 1.5 0
		Smiths Ranch (Dry Lake)	280 (520)	280 (520)	4 5 6	4.0 1.5 0
		Bonneville (Dry Lake)	410 (760)	410 (760)	5 6 7	4.0 2.0 0
	Holloman AFB	Holloman AFB	10 (18.5)	620 (1150	6 7 8	5.5 3.0 1.75
		Dyess AFB	330 (610)	940 (1740)	6 7 8 9 10	11.0 8.0 5.0 2.5 0.5
		Perrin AFB	490 (910)	1090 (2020)	10 11 12	2.0 0.5 0
	Eglin AFB	Eglin AFB	10 (18.5)	1650 (3060)	9 10 11 12	10.0 7.5 5.0 3.0
		Cecil NAS	250 (465)	1900 (3520)	12	5.0
HTO, VTO and Subsonic Turbojet	Edwards AFB	All missions are single based and flown in the local area				
CSJ Mod	Edwards AFB	Hidden Hills	110 (204)	110 (204)	3 (1)	0
		Delmar	200 (370)	200 (370)	3 (1) 4.7	3.0 0
		Smiths Ranch	280 (520)	280 (520)	4.0 (1) 5.7	3.0 0
		Bonneville	410 (760)	410 (760)	4.5 (1) 6.5 (2)	5.0 0
CSJ and SJ Mods	Edwards AFB	Bonneville	410 (760)	410 (760)	7.0 (1)	0
	Holloman AFB	Holloman AFB	10 (18.5)	620 (1150)	6.0 (1) 8.0 (2)	5.0 0
		Dyess AFB	330 (610)	940 (1740)	8 (1) 11 (1) 8 (2) 9.5 (2)	5.0 0 3.5 0
		Perrin AFB	490 (910)	1090 (2020)	8 (1) 8 (2) 11 (1) 10.5 (2)	7.5 6.0 2.0 0
	Eglin AFB	Eglin AFB	10 (18.5)	1650 (3060)	8 (1) 8 (2) 9.7 (1) 11.5 (2)	13.5 12.0 8.0 1.0
		Cecil NAS	250 (465)	1900 (3520)	8 (1) 8 (2) 9.5 (1) 11.1 (2)	16.5 15.0 11.5 5.0
Arm Mod	Edwards AFB	Smiths Ranch	280 (520)	740 (1370)	Mach 12 Weapon Separation on PMR	
Stg. Mod	Holloman AFB	90 na mi (170 km) East of Dyess AFB	410 (760)	1030 (1910)	Mach 11 Stage on WSMR	

(1) RKT Boost to test Mach No., Cruise on SJ/CSJ
(2) RKT Boost to Mach 3, CSJ to Test Mach No., Cruise on CSJ

6.1.1.1 (U) HTO/VTO Vehicle Options - Flight operations for the HTO/VTO vehicle options can be single based since they function only to demonstrate and compare launch methods. This decision was based on the following considerations:

- o The primary research objectives can be achieved from the take-off and immediate climb out profile (field level to 30,000 feet (9150 m)).
- o Gross weight considerations, particularly for the VTO operation, could prohibit sufficient fuel loadings to accomplish a dual base mission.
- o Missions to maximum or near maximum capabilities would not add significantly to the research objective to warrant the added costs and constraints of dual base operations; these objectives can be obtained with the basic configuration.

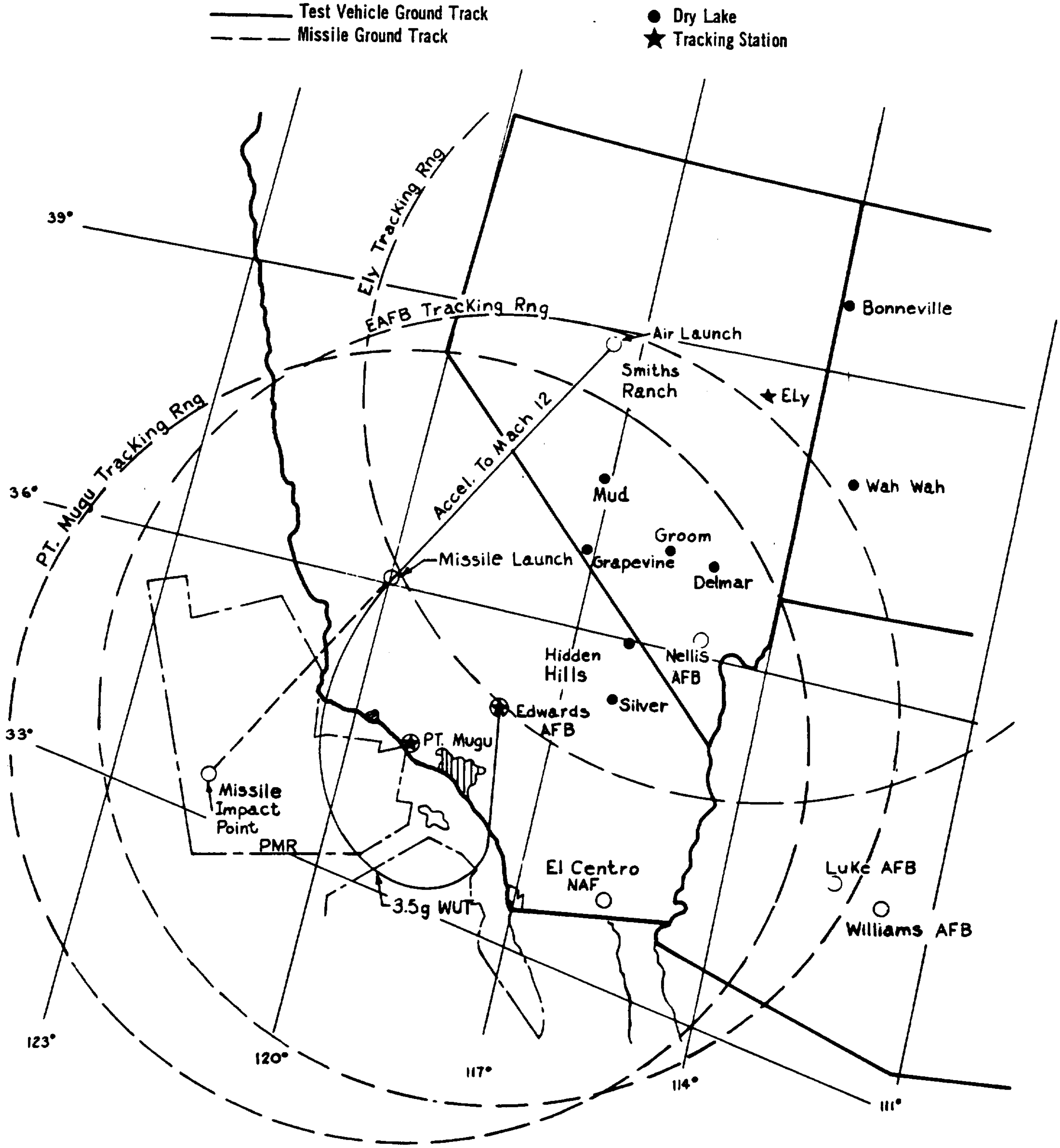
6.1.1.2 (U) SJ and CSJ Vehicle Options - The SJ and CSJ research options can utilize the same test ranges as the basic vehicle. The SJ/CSJ installations reduce the vehicle maximum performance capability somewhat which results in shorter range requirements. Therefore, maximum Mach number missions can be flown on some of the intermediate test ranges from Dyess AFB and Perrin AFB. The longer test ranges (from Eglin AFB and Cecil NAS) will be required on missions flown for extended test time at lower Mach number. The SJ test vehicle would be relegated to dual basing early in the program because Mach numbers above 6 are required to sustain flight with the SJ engine. The CSJ test vehicle is not as constrained since it can be sustained at Mach 3 on the CSJ engine which would permit some single base testing.

6.1.1.3 (U) TPS Vehicle Option - The thermal protection system test section does not effect the basic vehicle mission capabilities. Therefore, basing and test range requirements for this option are the same as for the basic vehicle.

6.1.1.4 (U) ARM Vehicle Option - The armament option, wherein research will be conducted involving the separation of missiles or other stores from the test vehicle, presents a number of additional considerations to the basing and operational plan. These considerations result from the premise that the missile should be separated at conditions, and at a location, such that the missile will impact in a controlled zone suitable for that purpose. It will also be necessary to launch the aircraft near an emergency landing site, have range available to climb and accelerate to the desired launch conditions (Mach number, altitude, and geographic location), and be able to effect a return to Edwards AFB for landing after the separation. It is anticipated that research on missile separations from a hypersonic aircraft would require testing over a range of release conditions encompassing different speeds, altitudes, and aircraft attitudes. Figure 6-4 presents a candidate flight plan which could be employed for a level flight release of a non-powered missile shape at Mach 12 from near equilibrium altitude. The test plan utilizes the existing facilities of the X-15 test range and the nearby Pacific Missile Range (PMR). A turn is required following the missile release in order to land the test vehicle at Edwards AFB. This general test plan could also be used to accommodate launches over a range of Mach numbers and altitudes.

6.1.1.5 (U) STG Vehicle Option - The basing and test range requirements of the staging option are similar to the armament option in that a suitable impact zone for the staged vehicle is required. However, the same test range cannot be used be-

(U)FIGURE 6–4
MACH 12 ARM OPTION CANDIDATE TEST RANGE



cause the high trajectory flight profile of the staging mission, shown in Figure 4-67 puts the vehicle recovery position almost over the stage impact point. This, combined with the fact that all available fuel is consumed during the mission would make it difficult for the vehicle to reach Edwards AFB from a turn initiated over the PMR. Therefore, a straight line type profile was considered more practical for this mission. The only other impact zone considered suitable was the White Sands Missile Range (WSMR) in New Mexico. Figure 6-5 shows a candidate flight plan utilizing the WSMR with Holloman AFB serving as the takeoff base.

6.1.1.6 (U) Subsonic TJ Vehicle Option - This modification can be single based at Edwards AFB for all missions. Flights can be flown in the local area and no specific test ranges will be required.

6.1.2 (U) MACH 6 VEHICLE - The basing and test ranges for all candidate vehicles defined during Phase II were based on a glide range nominally between the maximum L/D glide range and the minimum glide range. A more attractive basing plan that reduced total range requirements was found available for the Mach 6 test vehicle during Phase III. Range reduction was afforded by reconsidering the glide segment of the profile since the accel/climb and cruise ranges are relatively fixed. The Phase III profile utilizes the near minimum glide range for total range requirement as shown in Figure 6-6.

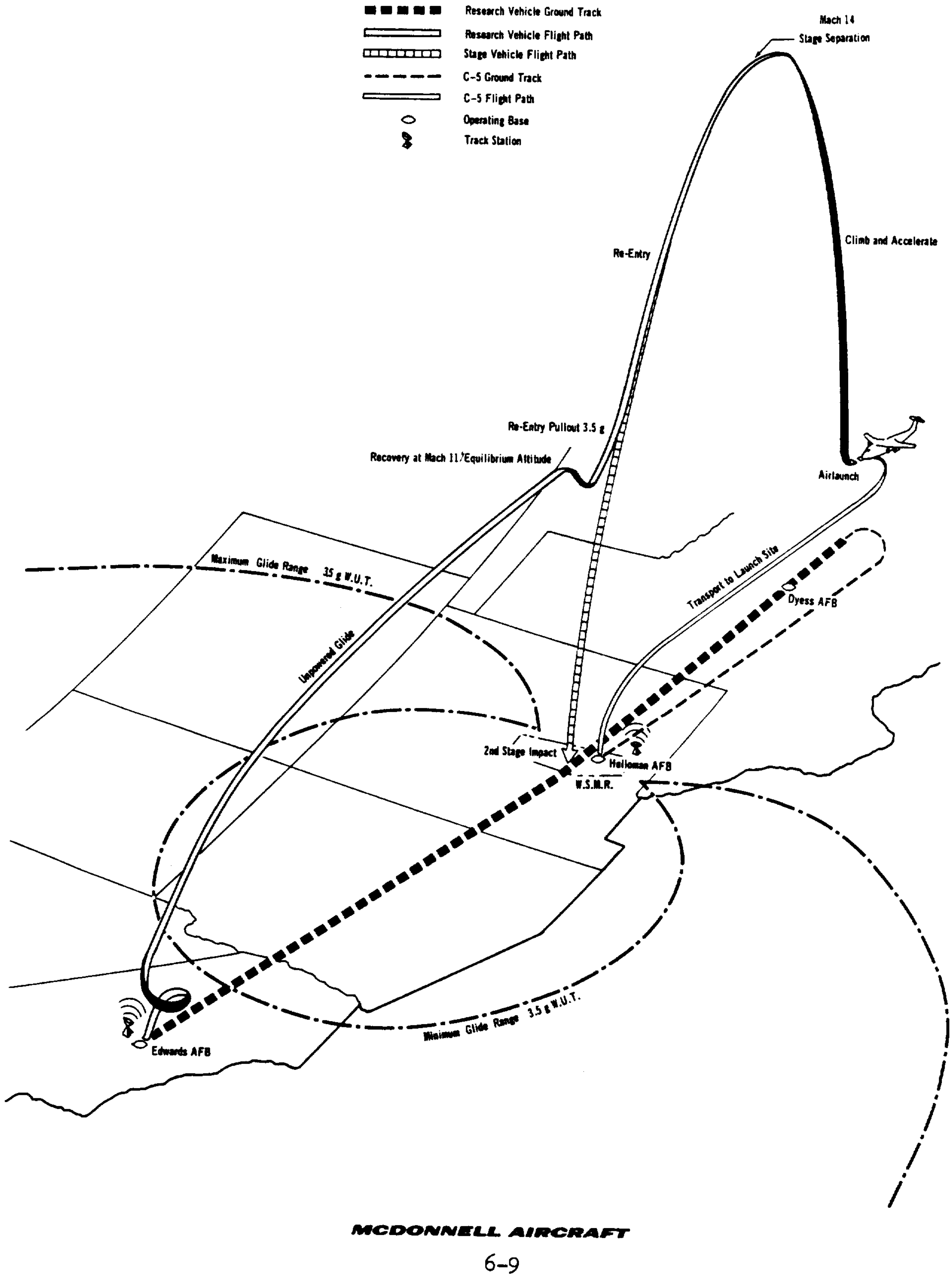
(U) The ability to perform missions over shorter ranges will increase the single base test capability thereby permitting a greater number of the total missions to be single based. Shorter range requirements will also permit the design mission (Mach 6 for 5 minutes) to be flown on the Holloman AFB to Edwards AFB test range which would eliminate the need for a third operating base. Using the near minimum glide range reduces the pilots flexibility in range control; however, the following factors tend to offset this.

- TJ engine is available for landing.
- Pilot has visual reference of the ground throughout the mission.
- Previous Mach 6 operational experience available from the X-15 program.
- Good single base test capability permitting build ups while obtaining incremental performance experience.
- Problem not as critical with Mach 6 as with Mach 12 because of shorter range involved.

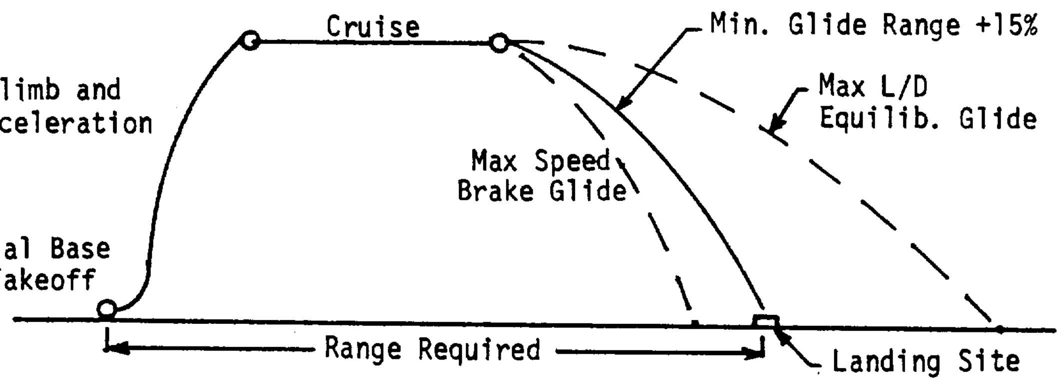
(U) The Mach 6 vehicle has greater single base mission flexibility than the all-rocket vehicles because the TRJ engine provides a more economical subsonic cruise capability. Figure 6-7 illustrates that single base missions using the inbound test profile permit testing to near 50 percent of design Mach number. The limiting factor for single based missions is the amount of JP fuel available. This capability could be increased if additional JP fuel were provided to increase the subsonic outbound cruise range. Two approaches for providing additional JP fuel were considered and are discussed in the following paragraphs.

(U) FIGURE 6–5 CANDIDATE TEST RANGE

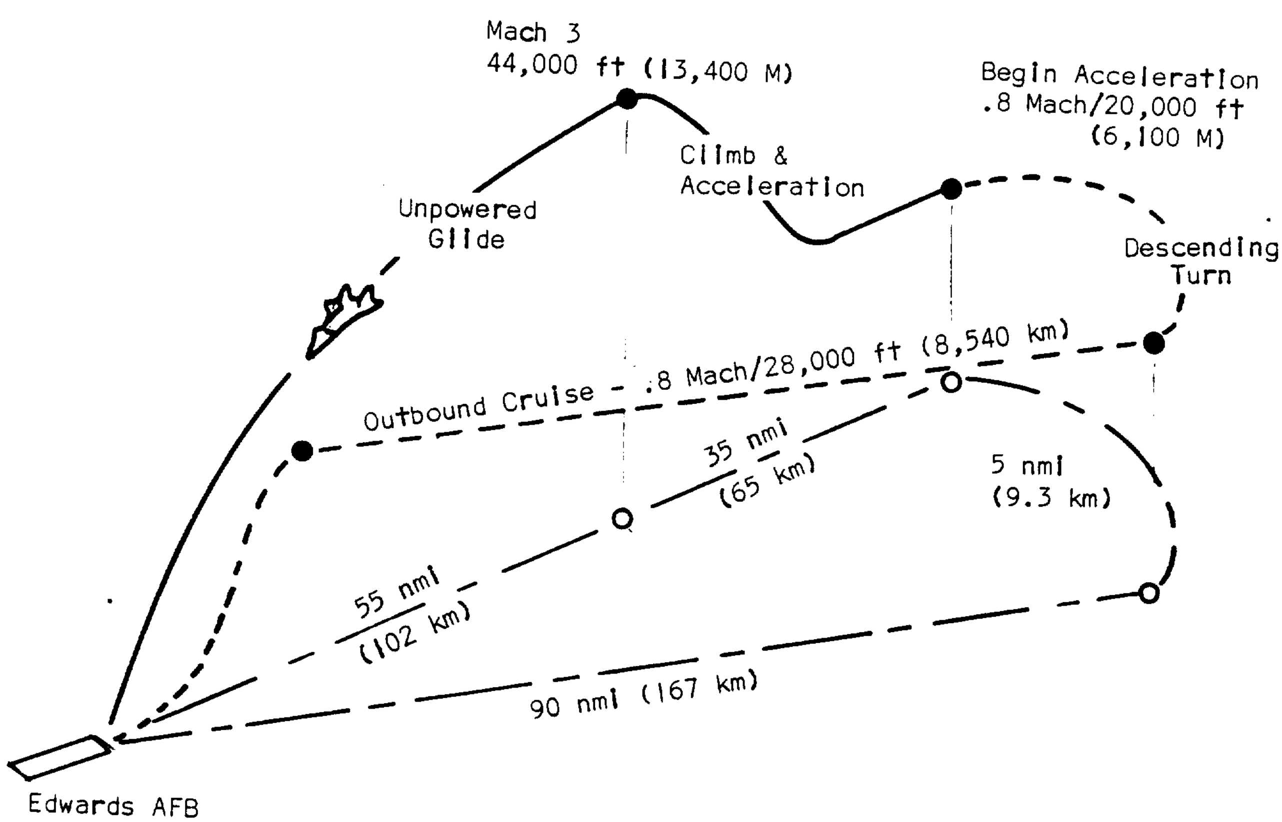
MACH 12 VEHICLE-STAGING OPTION MISSION



(U) FIGURE 6–6
MACH 6 VEHICLE TYPICAL PHASE III FLIGHT PROFILE



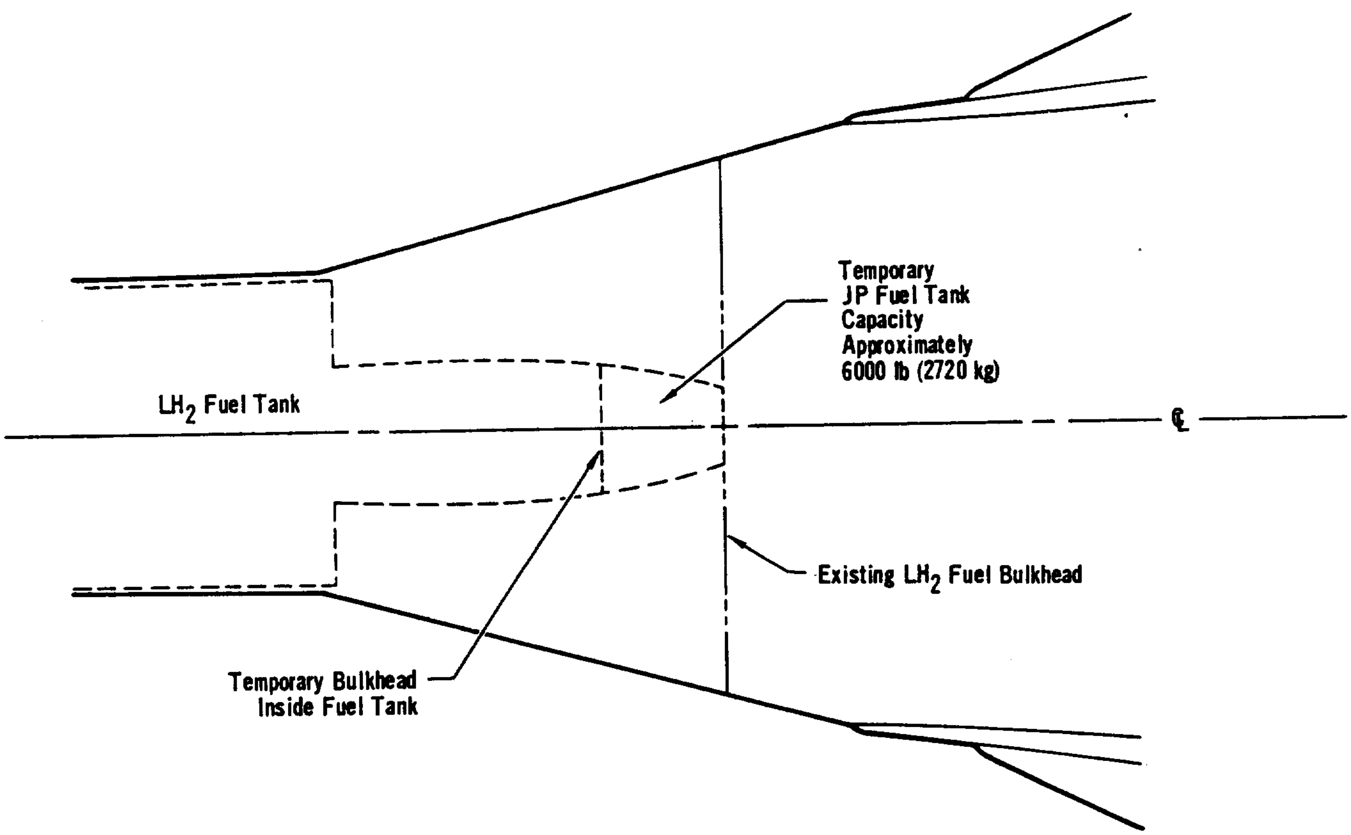
(U) FIGURE 6–7
MACH 6 VEHICLE-SINGLE BASE FLIGHT PROFILE
TAKEOFF FUEL LOAD – 8600 LB JP + 1000 LB LH_2



(a) Increased Internal Tankage - This approach has the advantage of providing the fuel without an increased drag penalty; however, a portion of the LH_2 fuel tankage space must be converted to carry it as illustrated in Figure 6-8. Subsequently when dual base missions are flown, the space would have to be reconverted for LH_2.

(b) External Fuel Drop Tanks - This method would permit mission versatility for a minimum of effort (easy on, easy off). The primary disadvantage of external tanks is the requirement to provide a suitable drop zone. Furthermore, several of these zones would be required in order to accommodate a variety of missions. This presumes that the logical time to drop the tanks is at the end of the outbound cruise just pricr to accelerating for the test.

(U) FIGURE 6-8
MACH 6 ADDITIONAL JP FUEL CONFIGURATION



(U) Figure 6-9 shows the increase in single base mission capability with additional JP fuel. It can be seen that the additional fuel is attractive from the vehicle development standpoint since near design Mach number can be achieved with 6000 lb (2720 kg) of additional JP fuel. This capability enables approximately 80 percent of the total flight program to be conducted with single base operations. Without additional JP fuel, probably only 20 percent of the missions could be single based.

(U) FIGURE 6-9
EFFECT OF ADDITIONAL JP FUEL ON MACH 6 VEHICLE SINGLE BASE MISSION CAPABILITY

FLIGHT PROFILE: (1) T.O AND SUBSONIC CRUISE OUTBOUND ON JP FUEL

(2) ACCELERATE TO TEST MACH NO. INBOUND

a. JP + LH_2 FUEL UP TO MACH 3.5

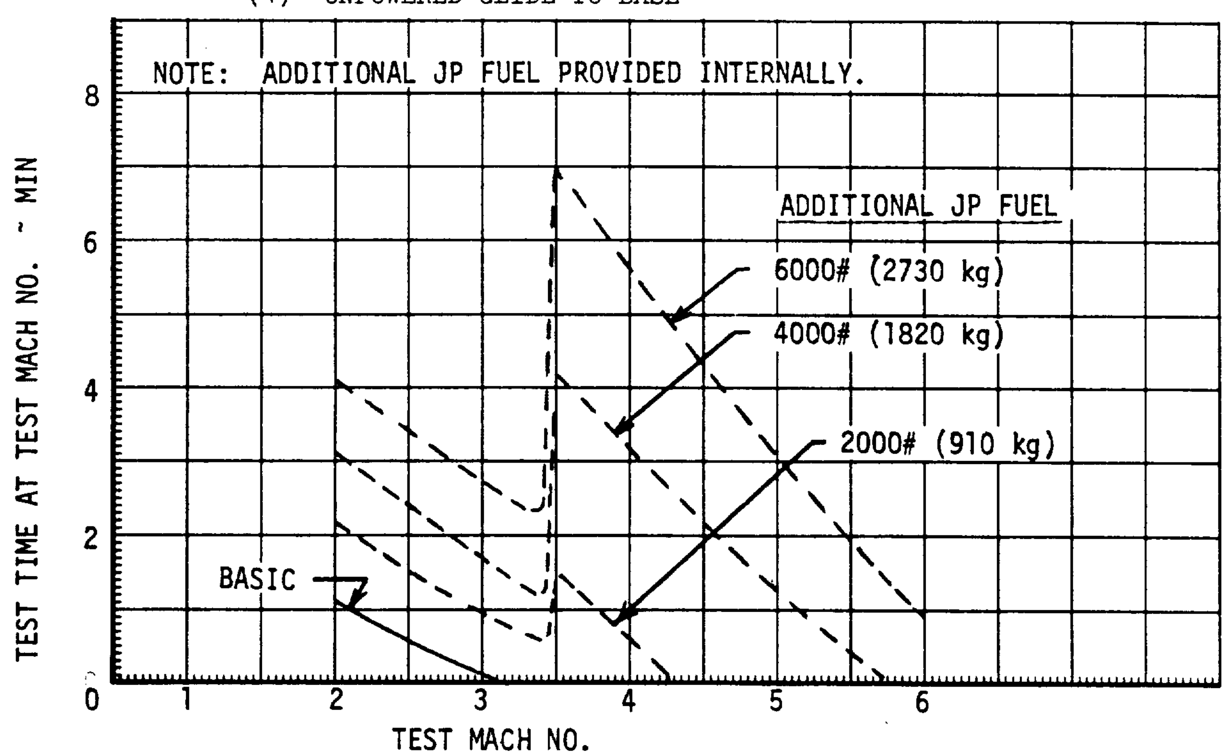
b. LH_2 FUEL ONLY BEYOND MACH 3.5

(3) CRUISE AT TEST MACH NO.

a. ON JP FUEL UP TO MACH 3.5

b. ON LH_2 FUEL BEYOND MACH 3.5

(4) UNPOWERED GLIDE TO BASE



(U) Basing plans for the research vehicle optional configurations discussed in the following paragraphs.

6.1.2.1 (U)ARM Vehicle Option- The Mach 6 armament mission will have the same requirements to use a controlled weapons impact range similar to the Mach 12 vehicle. However, the Mach 6 mission is afforded several options in the basing and test plan considerations. These options were based on the PMR and WSMR being considered as the only acceptable impact ranges. Two candidate plans were identified which utilize the PMR. Figure 6-10 shows the most attractive plan, from an operational standpoint, which involves single basing the mission from Edwards AFB. Weapons separations from subsonic to Mach 6 speeds can be accommodated; however, these missions require that additional JP be carried for the outbound cruise segments. The other plan using the PMR (not illustrated) requires dual basing and employs a straight line flight profile from the takeoff base (Nellis AFB, Nevada) to the launch point. A tight 180 degree turn and glide to landing at Edwards AFB is required. The plan using the WSMR employs the same profile as the dual base PMR mission. Dyess AFB, Texas would be the takeoff base and Holloman AFB, New Mexico the landing site. Weapon separations for dual base missions would be limited to the Mach 5 to 6 envelope in order to have sufficient energy to affect the turn and glide to landing.

6.1.2.2 (U) TPS Option-The thermal protection system test section does not effect the basic vehicle mission capability. Therefore, the basing and test range requirements are the same as for the basic vehicle.

6.2 FLIGHT OPERATIONS

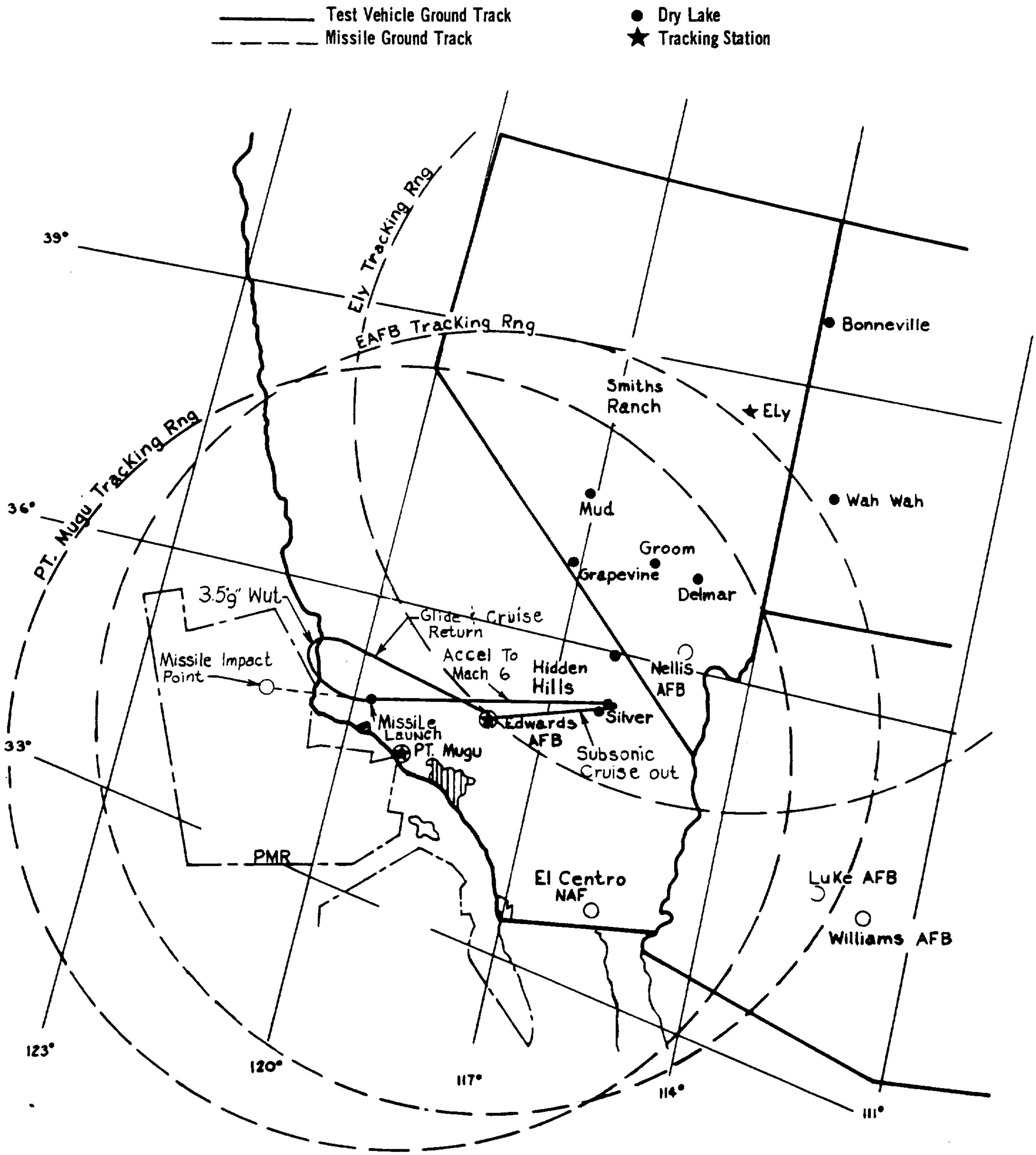
(U) Flight operation studies performed during Phases I and II treated the operational requirements and identified operating constraints for numerous hypersonic research vehicle design concepts. This study describes the operational considerations for the two Phase III basic vehicles and the research option configurations of these vehicles. Some of the options consist of design concepts which when studied as individual concepts during the preliminary phases presented operational problems and constraints. These problems and constraints are reduced by incorporating the concepts as options on the premise that the basic vehicles will have undergone a flight development effort prior to incorporation of the options. Thus, a degree of confidence and reliability as well as operation experience will have been established for the basic vehicle and its systems.

(U) The primary concern in flight development of the Mach 12 vehicle will be the high Mach number/temperature environment of the flight envelope; therefore, the structures and TPS considerations are anticipated to be a dominant and pacing factor in the program. The Mach 6 vehicle development effort will probably be dictated by the airbreathing TRJ propulsion system. The following paragraphs present the results of the flight operation study.

6.2.1 (U) MACH 12 VEHICLE - Flight operations for the Mach 12 airlaunched test vehicle will be similar so that of the X-15 program. Figure 6-11 presents the recommended flight operation plan consisting of contractor predelivery flight tests followed by the government's research phase.

(U) The mission profile for the Mach 12 - airlaunched - all rocket vehicle was discussed in Section 6.1. The aircraft will be released from the C-5A carrier within power off glide range of a suitable landing site in the event the mission has to be aborted shortly after drop. Aborts due to propulsion failures can be minimized by prestarting the engines in idle mode. After release, the aircraft will be accelerated, using a prescribed pitch program for the climb, until near test altitude is reached. At test altitude, the aircraft will be leveled off and the power reduced to that required to sustain the desired cruise velocity. Reduced power on

(U) FIGURE 6-10
MACH 6 ARM OPTION CANDIDATE TEST RANGE



(U) FIGURE 6-11
MACH 12 VEHICLE RECOMMENDED FLIGHT OPERATIONS PLAN

Program Phase	Flight Envelope	Test Method	Test Objectives
Pre-Delivery	Subsonic	Captive	Evaluate compatibility with C-5A carrier aircraft.
		Glide	Evaluate separation, low speed handling, glide, and landing characteristics.
	Supersonic to Low Hypersonic	Accelerations (Low to Max q) Maneuvering Flight (Neg. to Max g)	Evaluate flight and handling characteristics. Verify systems/subsystems operation. Verify structural integrity. Establish operating procedures and piloting techniques.
Research	Hypersonic	Accelerations (Low to Max q) Maneuvering Flight (Neg. to Max g)	Expand envelope to design speed. Verify structural integrity in high temperature environment. Satisfy research objectives obtainable with transient flight.
	Hypersonic	Sustained Flight (Low to High q)	Satisfy research objectives requiring sustained hypersonic flight.

one of the engines will be sufficient for cruise on most missions. At fuel burnout (or intentional shutdown), the speed brakes will be extended (approximately half way) to provide the nominal glide range to reach the intended landing site. Onboard systems will inform the pilot of actual versus desired position so he can modulate the glide range by further extension, or retraction of the speed brakes. Ground tracking stations will also monitor the flight profile to verify the on-board systems, and to provide emergency backup in the event of on-board system failures. Once the aircraft has decelerated below Mach 6, the cockpit can be raised permitting visual acquisition of the landing site.

(U) The most significant operational differences between these vehicles and the X-15 vehicles will result from the long ranges involved. Mach 10 and 12 missions require launches from northern Florida which presents several problems or constraints to the program. These constraints include:

(U) Weather factor - One of the criteria for test missions will be suitable weather conditions, primarily little or no cloud cover. The clear sky conditions will have to exist along the entire test range in order to permit landing at an emergency site if necessary. Missions from Florida will require good weather conditions along 1200 to 1900 na. mi. (2220 - 3520 km) of test range. Based on 1967 survey of weather conditions along the test range, approximately 60% of the days were probably unsuitable for test missions. This could create delays in the research program; however, proper program planning could minimize the problem. During periods of adverse weather in the east, intermediate missions could be flown on the southwestern test ranges where the conditions are suitable most of the time.

(U) Population density factor - Missions flown from eastern launch sites will over-fly more densely populated areas than missions flown in the southwest. Based on 1967 data, the states of Florida, Georgia, Alabama, Mississippi, and Louisiana contain 60% of the population and only 34% of the land area of the states over-flown. The over-flown states would be the probable impact zone if a major failure occurred; however, the potential impact area should also be considered. This area could encompass 30 states plus Mexico for a Mach 12 mission from Cecil NAS, Florida.

(U) Tracking, Communications, and Time Zones - Another problem associated with the long test ranges concerns tracking requirements. It will be necessary to transfer from one tracking station to another several times during the mission. This is also true for the telemetry data and voice communications. The three-hour time differential between eastern launch sites and the western recovery site could reduce the number of daylight hours if they were needed to accomplish the mission.

(U) The impact of testing various research modification packages on flight operations was also studied and the results are reported below.

6.2.1.1 (U) HTO/VTO Vehicle Options - A significant change to the basic vehicle associated with the HTO and VTO options is the installation of a modified configuration of the rocket engine. This change would probably require a preliminary evaluation of the flight characteristics, particularly controllability from vectored thrust at low airspeeds, using the airlaunch mode of operation prior to HTO or VTO flights. Another preliminary test might be required for the VTO launch cart. If it were found that jettison of the cart is a necessary operation, it would be advisable to perform preliminary tests to functionally check the system and evaluate separation characteristics. These tests could be performed during the airlaunched missions flown to evaluate the modified rocket engines. Jettison tests for the HTO wheeled cart cannot be accommodated in this manner because there is not enough ground clearance to install the cart with the vehicle aboard the C-5A carrier aircraft.

(U) Prior to an HTO mission, taxi tests should be made in order to evaluate thrust management, braking, and directional control during ground roll. Accomplishment of these tests on the large lake bed at Edwards AFB would permit taxi to near liftoff speed. Initial flights of the HTO/VTO vehicles should be made at light gross weights. Subsequent flights could be made with incremental increases in fuel loadings. Launch gross weight will be especially critical to VTO operations where the loss of one engine during the first few seconds after lift off could be critical. It is recommended that the gross weight for VTO missions be limited such that adequate T/W ratio is available with one engine out. A single engine failure could also present another problem for the VTO mission. Flight control (pitch, roll, and yaw) is obtained by vectored thrust from the five rocket engines until sufficient airspeed is achieved for the aerodynamic control surfaces to become effective (30 to 40 seconds after lift off). The magnitude of the effects of an engine failure would depend on its location; an outboard engine would be most critical, the center engine the least critical. Controllability problems from an engine failure would probably not be as critical for HTO missions because aerodynamic control would be available at, or shortly after, lift off. Missions which are aborted after lift off will require the dumping of propellant in order to reduce the vehicle weight to an acceptable level for landing. Aborts involving the loss of thrust could be a critical problem for the HTO/VTO mode of operation if the abort were to occur during the climb segment at altitudes which would not provide sufficient time to expel the fuel. It will be necessary to establish a minimum altitude for safe recovery for each mission depending on available fuel dump rates and fuel loading.

6.2.1.2 (U) SJ and CSJ Vehicle Options - Testing an SJ or CSJ propulsion system on a developed vehicle should minimize the risks associated with new propulsion concepts. This occurs primarily because the rocket engines can be used to accelerate the aircraft to the test conditions. The airlaunch mode of operation also affords mission versatility for incremental buildup in the flight envelope which should expedite the research program on the propulsion systems. Incorporation of the SJ or CSJ options will make a significant change to the vehicle configuration. It is anticipated that at least one power-off glide flight would be required to evaluate low speed handling, glide, and landing characteristics prior to research flights. Flights with the SJ engine will require mostly two base operations since Mach 7 or 8 is the minimum operating speed for SJ operation. Little more than startup characteristics could be evaluated on the X-15 test range. The risk of initial flights on the longer two base test ranges would be minimized by the availability of rocket engines if problems with the SJ were encountered. Testing the CSJ option permits some single base missions because it can operate at speeds as low as Mach 3.

(U) The mission profiles and operating procedures for the SJ/CSJ will be similar to that of the basic configuration. Some additional versatility in the mission profile is available because both the rocket and airbreather engines can be used in a variety of combinations during the acceleration to test Mach number as shown in Figure 4-64.

6.2.1.3 (U) TPS Vehicle Option - Incorporation of the thermal protection system test section will not effect the basic vehicle flying qualities or mission capability. Flight profiles and operational missions would be the same as the basic vehicle. The TPS should be evaluated on an incremental envelope expansion basis in order to acquire data over a range of flight conditions. If the design is such that a failure of the TPS test section would not jeopardize the vehicle integrity, the testing can be accelerated by larger incremental increases in Mach number/test time. If a failure cannot be tolerated, then the envelope expansion would have to be commensurate with standard flight safety practices.

6.2.1.4 (U) ARM Vehicle Option - Missions involving the separation of weapons/stores, particularly at high Mach numbers and high altitudes, present an exacting task in space positioning requiring a great deal of pre-mission planning and coordination.

Prior to a weapons separation mission, one or more practice missions (dry runs) should be flown in order to:

- Coordinate the efforts and timing between the pilot and the ground tracking and range control personnel.
- Verify the planned flight profile.
- Give the pilot experience in flying the mission.

6.2.1.5 (U) <u>STG Vehicle Option</u> - The staging mission requirements are similar to those of the armament option with respect to exacting space positioning. The staging mission presents additional complexities because of the very high altitude ballistic profile. Dry runs (without the stage vehicle aboard) would also be required for this mission for the same purpose as the armament mission. Additional pre-mission development test should include captive flight to evaluate the compatibility between the carrier aircraft and the staging cnffiguration.

6.2.1.6 (U) <u>Subsonic TJ Vehicle Option</u> - Incorporation of turbojet engines on the basic vehicle to perform low-speed flight research should not present any particular operating problems. Fly rates for this configuration should be substantially higher than other configurations because it would not be constrained by the factors associated with hypersonic flight, carrier aircraft operations, and a large quantity of instrumentation parameters.

6.2.2 (U) <u>MACH 6 VEHICLE</u> - Flight operations for the Mach 6 test vehicle will be somewhat similar to conventional aircraft because of the turbojet engine and horizontal takeoff capability. Figure 6-12 presents the recommended flight operation plan consisting of contractor predelivery test flights, followed by the government's research phase.

(U) The primary goal of the initial test phase will be to develop the aircraft to the point where flight can be sustained on RJ operation. Once this goal has been achieved, the vehicle will be ready to enter the research test phase. It is anticipated that most of the test effort during the predelivery and envelope expansion test phases will be directed toward the propulsion system and the TRJ operation in particular. The subsonic flights and initial supersonic flights will be flown with the TJ using JP fuel. RJ testing using LH_2 fuel can be phased in once experience and confidence has been established for TJ operation. Propulsion system testing will encompass the following:

- TJ operation using JP fuel

  Engine/afterburner handling

  Engine/afterburner operation and control

  Airstarts

  Engine - inlet compatibility and performance

  Variable inlet and bypass systems operation

(U) FIGURE 6-12
MACH 6 VEHICLE RECOMMENDED FLIGHT OPERATIONS PLAN

PROGRAM PHASE	FLIGHT ENVELOPE	TEST METHOD	TEST OBJECTIVES
Pre-Delivery	Ground	Taxi (Low and High Speed)	Evaluate Braking, Control and Ground Handling Characteristics
	Subsonic to Supersonic	Accelerations (Low to Max q) Maneuvering Flight (Neg. to Max g) Sustained Cruise	Evaluate flying qualities and T.O./landing performance. Verify systems/subsystems operation. Verify structural integrity. Evaluate propulsion system operation and performance. Establish operating procedures and piloting techniques.
Research	Supersonic to Hypersonic	Accelerations (Low to Max q) Maneuvering Flight (Neg. to Max g)	Expand envelope to design speed. Verify structural integrity at design speeds. Develop propulsion system to design speed. Satisfy research objectives obtainable with transient flight.
	Hypersonic	Sustained Cruise (Low to High g)	Satisfy research objectives requiring sustained hypersonic flight.

- TRJ operation using JP/LH_2 fuel

  RJ starts

  RJ handling and operation

  Effects of RJ operation on TJ operation

  Engine-inlet compatibility and performance

  Variable inlet and bypass systems operation

  Transition from TRJ to RJ only operation

  Transition from RJ to TRJ to TJ operation

  Inlet cooling performance.

(U) Flight operations for the two most feasible research options incorporated on the Mach 6 vehicle, were examined and the results are reported.

6.2.2.1 (U) <u>ARM Vehicle Option</u>-Flight operations with the Mach 6 armament option pose the same considerations as the Mach 12 armament option. The constraints are less severe because of the shorter ranges involved, and as pointed out in the basing study, more mission versatility is available. The single base mission capability would depend on providing additional JP fuel.

6.2.2.2 (U) <u>TPS Vehicle Option</u>-Incorporation of a TPS test section will not effect the basic vehicle flying qualities or mission capability. Therefore, flight operations and mission profiles would be the same as those employed for the basic vehicle. The TPS will be evaluated using the philosophy of incremental envelope expansion testing in order to establish reliability in the system by extrapolating the test results. The rates of envelope expansion with Mach 12 vehicle TPS option was predicted to be a function of survivability if the TPS failed. This is also true for the Mach 6 vehicle; however, another aspect must be considered which concerns the airbreathing engines and inlets. The vehicle could be designed to survive a TPS test section failure from the structural standpoint, but ingestion of debris into the inlet would present an additional hazard. Therefore, the location, as well as the design philosophy could have an effect on the operational test plan.

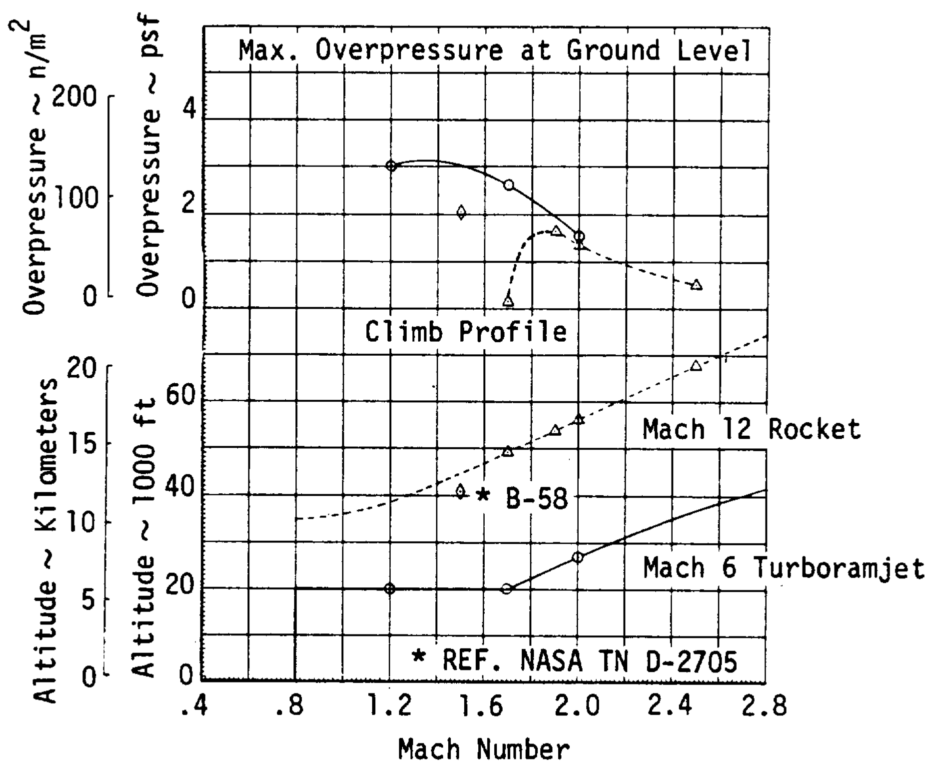
6.3 SONIC BOOM

(U) The sonic boom far field characteristics of the two basic configurations are determined for the acceleration-climb and cruise phase of flight. The computer program of Reference (24) is employed to determine the sonic boom overpressure that will be experienced at ground level. The computer program includes both climb and acceleration effects. The results of the analysis for the acceleration-climb are presented in terms of sonic boom ground level overpressures in Figure 6-13. The maximum sonic boom ground level overpressure produced during the climb is 3.0 lb/ft^2 (144 N/M^2) for the Mach 6 aircraft and 1.6 lb/ft^2 (76.5 N/M^2) for the Mach 12 vehicle. The peak overpressure for the Mach 12 aircraft occurs at a higher Mach number due to the higher flight path angle. The Mach 12 vehicle will therefore produce ground level overpressures comparable to those associated with current operational supersonic aircraft. The Mach 6 aircraft can be expected to cause some damage to large plate glass windows if the peak pressure is generated over a populated area. The sonic boom overpressures obtained during the cruise portion of flight are 0.20 lb/ft^2 (9.6 N/M^2) and 0.08 lb/ft^2 (3.8 N/M^2) for the Mach 6 and Mach 12 aircraft, respectively.

6.4 EMERGENCY DESCENT AND LANDING

(U) A major consideration in the selection of test ranges was the emergency landing requirements. Candidate emergency landing sites were identified during the Phase I and II studies. These studies were continued in Phase III in order to accommodate changes in the recommended test ranges and to re-evaluate the adequacy and suitability of the candidate sites. The Phase III study was expanded to define the requirements and identify the problems associated with the execution of emergency landings. The following paragraphs present the results of the study.

(U) FIGURE 6-13
SONIC BOOM EFFECTS
Far Field



6.4.1 (U) EMERGENCY LANDING SITES - One of the criteria for selecting the launch sites identified in the basing and operating plan for the Mach 12 vehicle was the availability of emergency landing sites along the flight path. The study was primarily concerned with the dual base missions. Single base missions on the X-15 test range will utilize dry lake beds which were found suitable as emergency landing sites for the X-15 program.

(U) One objective of the study was to show that a suitable emergency landing site could be reached if an emergency occurred at any point along the flight path. The following criteria were used to achieve this objective:

(1) The site must be an existing government-owned military airfield.

(2) The test vehicle should be able to reach an intended site from an unpowered glide utilizing 80 percent or less of the maximum-minimum glide range. If fuel is available a reduced power cruise may be used to increase range capability.

(3) The heading change from the initial flight path to line up on the emergency site should not exceed 30 degrees.

(4) The site location should be remote enough from nearby communities to avoid flying over the community during final approach.

(5) The site should have at least one hard surface runway of 10,000 feet (3050 m) or longer.

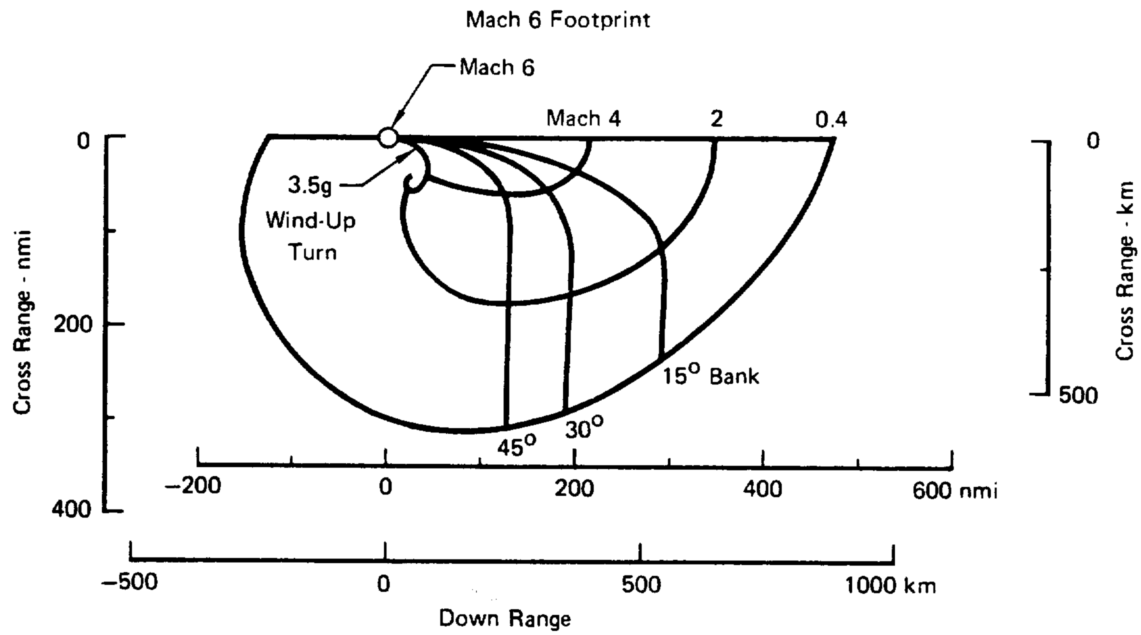
(6) The site facilities should have some standard navigation aids such as TACAN, VOR, UHF/DF, or Radar.

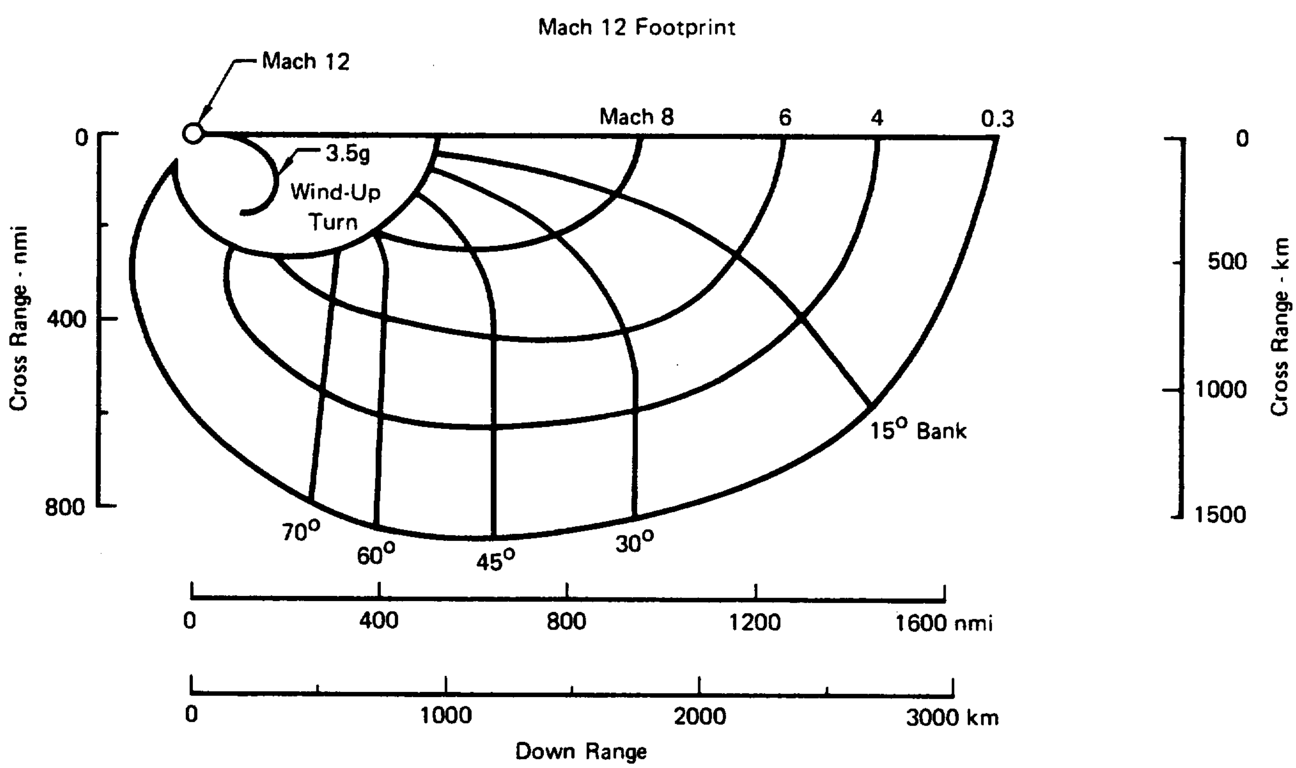
Sites meeting criterion (1) were identified for the selected test ranges. The reduced power glide consideration of criterion (2) differs from the Phase II study where only unpowered glide capabilities were considered. Subsequent studies have shown that the limited glide ranges from lower Mach numbers during the early acceleration segment of the flight were insufficient to reach the nearest site in the event of complete loss of thrust. However, the probability of such a failure is low. Investigation shows that one RL10-A-3-9 engine operating at 35 percent thrust is sufficient to maintain level flight. Data showed that the probability of three of the five engines firing is .99995 or 20,000 flights between a failure to get three engines operating. Thus, the probability that at least one of the five engines would not ignite is miniscule. In addition, a redundant (dual) propellant control and feed system has been incorporated in the vehicle design thus assuring a very low probability of loss of engine thrust due to propellant system failure.

(U) Criterion (3) was established to minimize the turn requirements in the event of an emergency. It applies primarily to emergencies occurring at higher Mach numbers (above 1.5 for Mach 12 vehicle and above Mach 3 for the Mach 6 vehicle). For emergencies at lower Mach numbers a 180 degree turn is required for return to the take-off/launch base. The available landing footprints for the Mach 12 and Mach 6 vehicles are shown in Figure 6-14. These footprints were developed from various trajectories utilizing maneuvers which would provide high emergency landing site capability.

(U) The study also shows that not all candidate sites met the criteria items (4) and (5) above. Any base which did not meet criterion (4) was rejected because it would present an additional safety hazard. All bases with runways less than 10,000 ft. (3080 m) (criterion 5) were rejected as primary sites. Some bases with less than 10,000 foot (3050 m) runways were retained as alternate sites because they offered the nearest site attainable from a particular point in the mission. These bases are equipped with arresting gear on the runway over-runs which could make them acceptable if an arresting hook were incorporated on the test vehicle. Figure 6-15 presents the emergency landing sites available for the Mach 12 vehicle on a Mach 12, 5 minute mission from Cecil NAS, Florida, to Edwards AFB, California. The bases identified as primary are those which are the nearest attainable base from a particular Mach number, or the most attractive landing sites when several sites can be reached. The alternate sites are those available if the primary site is unavailable for any reason. All emergency site acquisitions are from an unpowered glide unless otherwise noted. In some cases, a site is considered primary until the next attractive site can be reached and it then becomes an alternate site. Edwards AFB, which is the intended landing site for the Mach 12, 5 minute mission, is also shown as an emergency site since it can be reached after 1.5 minutes of cruise at Mach 12.

(U) FIGURE 6–14
HYFAC VEHICLE LANDING FOOTPRINTS



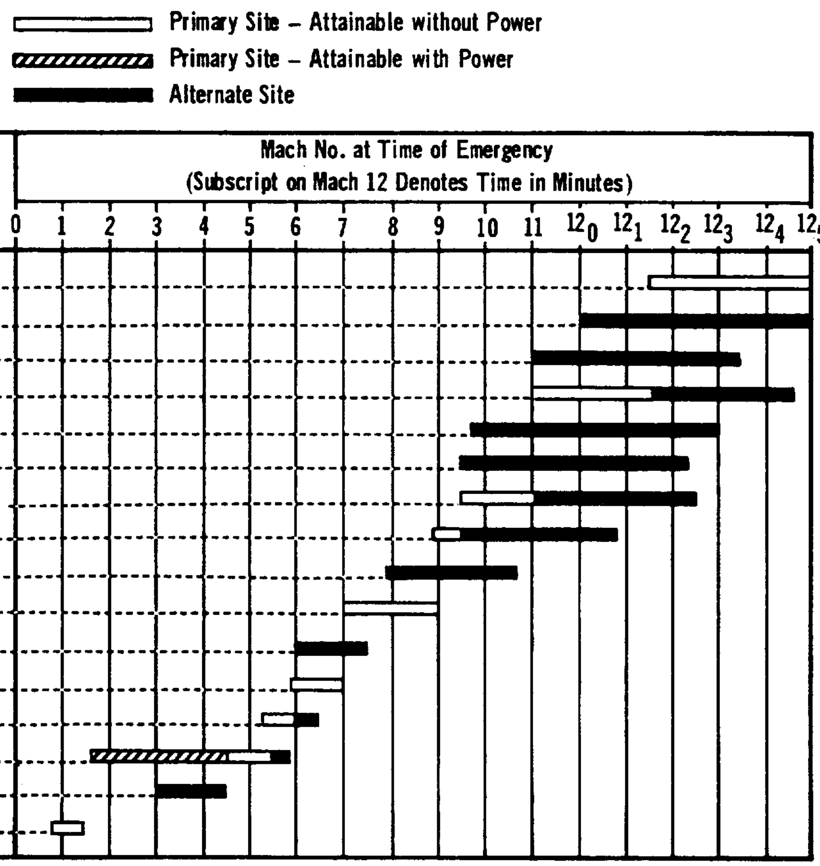


(U) FIGURE 6-15

MACH 12 VEHICLE EMERGENCY LANDING CAPABILITY

Test Range: Cecil NAS to Edwards AFB

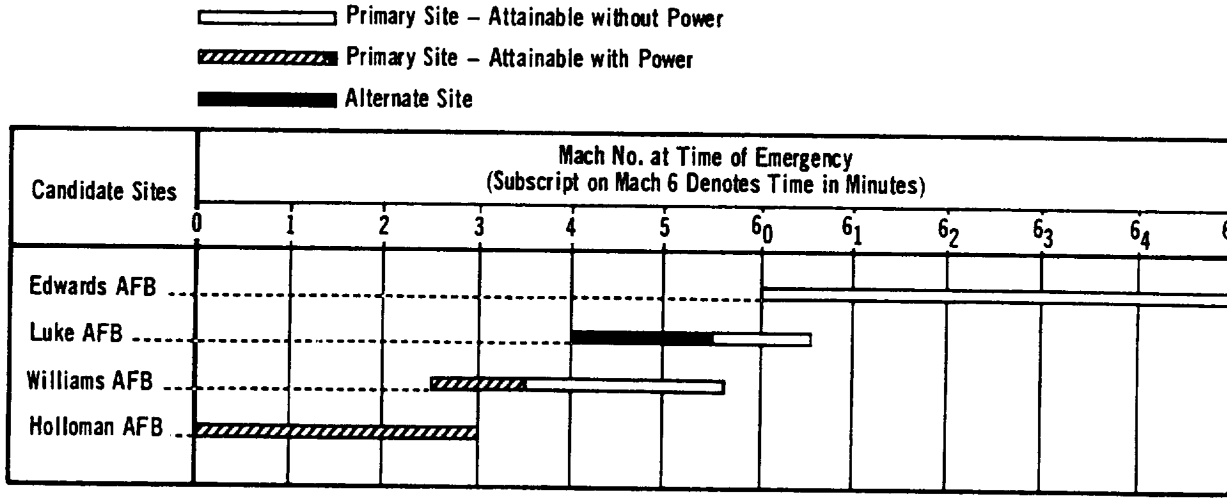
Mission: Mach 12/5 Min



(U) The emergency landing site requirements for the Mach 6 test vehicle are considerably less than the Mach 12 vehicle. Using the recommended two-base test range between Holloman and Edwards AFB, there are only two candidate emergency sites, Luke and Williams AFB. These bases are located 40 nm (74 km) apart approximately midway between Holloman and Edwards AFB. Figure 6-16 presents the emergency landing capability during a Holloman to Edwards mission. The figure shows that emergency landing capability is more dependent on having thrust available than for the Mach 12 vehicle. The single engine Mach 6 vehicle may not have the high probability of thrust available for emergency landing like the five engined Mach 12 vehicle. The emergency itself could be created by a propulsion system malfunction.

(U) FIGURE 6-16
MACH 6 VEHICLE EMERGENCY LANDING CAPABILITY

Test Range: Holloman AFB to Edwards AFB
Mission: Mach 6/5 Minutes



(U) Edwards AFB would be the primary landing site for emergency landing situations occurring during single based missions on the X-15 test range. The ability to reach the site, however, is dependent on the use of the TJ engine if the emergency occurs farther out than the maximum glide capability of the vehicle at the time of the emergency. Since single based missions will be utilized extensively during the early stages of the program, the reliability and confidence level in the engine will be considerably lower than that for dual base mission downstream in the program. Therefore, it will also be necessary to rely on the dry lakes as alternate emergency sites.

(U) Reliance on the TJ for return on aborted missions has an impact on the test missions. It would not be advisable to perform a mission such that the vehicle would not have sufficient JP fuel available to return to base from any point in the flight. This does not effect the basic vehicle which is limited to Mach 3 on single based missions; however, if the JP fuel capacity is increased in order to extend the single base capability, it would be possible to fly missions wherein aborted returns could not be accomplished.

6.4.2 (U) EMERGENCY LANDING OPERATIONS - The operational aspects of emergency landing were studied in order to determine the requirements necessary for the successful execution of the landing; and to define any problem areas. The following paragraphs describe the major elements of concern for an emergency landing.

(U) Recognition - The initiation of an emergency landing would necessarily begin with the recognition that the requirement exists. This recognition could come from two sources for the research vehicle. The pilot would be the primary source from either physical or sensible anomilies, or onboard instrument indications.

Ground station flight monitors would be the other source from the telemetry data. Once a problem dictating an emergency landing has been identified, it will be necessary to assess the urgency of the situation. The assessments may be classed as follows:

- o Urgent Action - problem necessitates an emergency landing at the nearest attainable site
- o Deferrable Action - problem requires an emergency landing but the situation permits the pilot to be more selective in the site.

(U) Site Selection - Once the decision to abort the mission and make an emergency landing has been made, the primary consideration concerns the selection of the landing site. The site selection will depend on the vehicle capability and the ability to assess the capability. Factors effecting vehicle capability are:

- o Range Capability - A function of speed, altitude, thrust availability and maneuvering limitation
- o Vehicle Position - distance from emergency sites
- o Nature of the Emergency - urgent or deferrible.

(U) Heading and Descent - Initiation of the landing would commence with a heading change (if required) toward the landing site. Candidate sites were selected along the test ranges to minimize turning requirements which would permit the pilot to concentrate on other tasks. Proper management of the descent profile would be critical to reaching the intended site. This would involve control of angle of attack, speed brake modulation, and thrust management (if required). The capability to control the critical parameters would be available to the pilot from on-board systems. Ground station monitoring of control and position data will be available to back-up and verify on-board systems.

(U) Approach and Landing - The basic problem in accomplishing the final approach and landing will be to arrive at the desired high key conditions over the landing site. This is particularly critical where the site is an airfield runway and thrust is not available to permit range or control changes. The use of navigation aids would be essential to assuring the success of the landing. Another consideration for landing assist would be standby chase aircraft employed in a manner similar to the X-15 flights.

6.5 PILOT ESCAPE SYSTEM

(U) A prerequisite to any manned aeronautics system is crew safety and survivability in the event of unpredictable system failures. Obviously, crew survival is the primary consideration. Secondarily, the information received from a surviving crew member provides invaluable data which cannot normally be attained by other means. In a research program in which high supersonic and hypersonic flight regimes, as well as state-of-the-art technologies, are being explored, crew recovery is essential.

6.5.1 (U) ESCAPE SYSTEM SURVEY - To achieve the desired level of crew survivability necessary for this research program over a wide range of operational environment, a study was conducted of candidate escape systems. Consideration was given to the relatively small number of research vehicles and the requirement to restrict escape system development costs to a level compatible with the overall program. Several escape systems from similar studies and operational systems shown in Figure 6-17 were examined and evaluated to select a reasonable escape system concept for the research vehicles proposed in this study. The applicability of each of these systems is illustrated in the Preferred Escape System column of Figures 6-18 and 6-19 for the existing operational condition.

6.5.2 (U) ESCAPE SYSTEMS EVALUATION - Five escape system concepts were examined and evaluated as to their relevant merit in this research program.

6.5.2.1 (U) Cockpit Module (CM) - The cockpit crew module escape system, Figure 6-20, severs and ejects the entire cockpit from the disabled aircraft. The pressurization, life support systems, and crew station comforts are retained throughout the escape sequence. The module protects the pilot from the high altitude, windblast, and thermal hazards of the environment. Automatic recovery parachute deployment lowers the module to the ground at a prescribed rate of descent, and the landing shock attenuation mechanism absorbs the final landing energy. For water landings, self-righting inflation bags and flotation bags maintain the proper module orientation. The cockpit serves as a foul weather shelter on land and a life raft on water until rescue is completed. Pressure suits and survival suits are not normally required for the crew because of the protection afforded by the module and available survival equipment aboard. The cockpit can be designed more efficiently without hampering the pilot's visibility or mobility and maintain a higher standard of crew comfort than any other escape system. A typical example of a production crew module escape system is that used in the F-111 aircraft. Reliability and service experience with this system has been excellent.

6.5.2.2 (U) Separable Nose (SN) - The separable nose module escape system, Figure 6-21, is similar in concept to the previously discussed cockpit crew module. In this system the entire nose section of the aircraft is separated behind the crew compartment and recovered. When compared to the cockpit crew module, this system presents a considerable weight penalty in the vehicle design because of the increased size and weight of the recoverable section. An extensive design study for the F-8A aircraft with this type escape system has been completed and development sled testing has been performed on a nose module of the F-104, References (25) and (26).

(U) FIGURE 6-17
SUMMARY OF TYPICAL ESCAPE SYSTEMS

System Reference Number	Escape System Type / Aircraft Usage	Speed Range	Altitude Range	Pressure Suit
1	Cockpit Module (CM)			
	F-111	(Aircraft capability in speed and altitude)		No
2	Separable Nose (SN)			
	F-104 (Study)	(Aircraft capability in speed and altitude) (0-700 kt (0-1297 km/hr) sled tests)		No
3	Encapsulated Seat (ES)			
	Goodyear (Study)	150 kt (278 km/hr) to Mach 1.2 Mach 4.0	0-55,000 ft (0-16,760 m) 55,000-100,000 ft (16,760-30,480 m)	No
	B-58	100-600 kt (185-1112 km/hr)	0-45,000 ft (0-13,720 m) 45,000-80,000 ft (13,720-24,380 m)	No
4	Separable Nose With Open Ejection Seat (SN/OES)			
	None	(Combination of systems 2 and 5)		Yes
5	Open Ejection Seat (OES)			
	Escapac IC (McDonnell Douglas)	0-600 kt (0-1112 km/hr)	0-50,000 ft (0-15,240 m)	Yes
	X-15	90-700 kt (167-1297 km/hr) Mach 4	0-60,000 ft (0-18,290 m) 60,000-120,000 ft (18,290-36,580 m)	Yes Yes
	SR-71	Mach 3	0-50,000 ft* (0-15,240 m)	Yes
	McDonnell Douglas Advanced Concept Ejection Seat (ACES) (Study)	0-600 kt* (0-1112 km/hr)	0-50,000 ft* (0-15,240 m)	Yes

*Estimated Value

(U) FIGURE 6-18
MACH 12 ESCAPE SYSTEM UTILIZATION

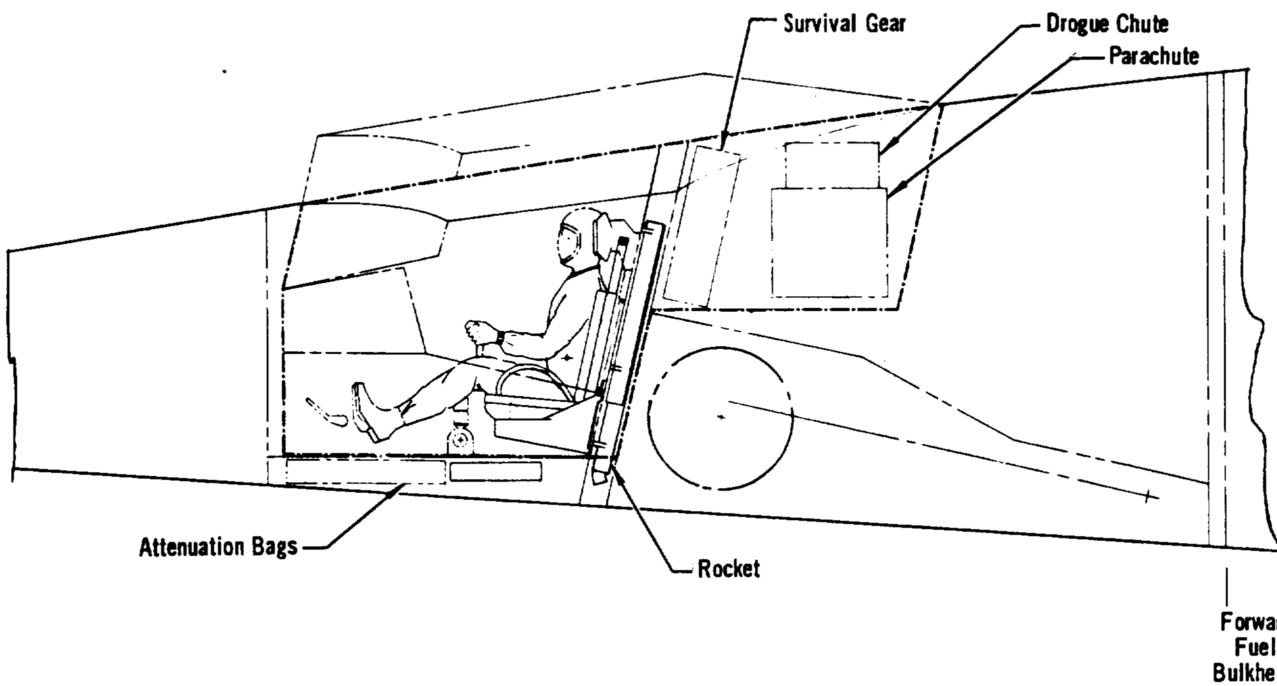
Situation	Vehicle Speed	Vehicle Altitude	Preferred* Escape System
Pilot Aboard Research Vehicle on C-5A on Ground	0	0	OES
C-5A Takeoff	Less Than Mach .8	Less Than 35,000 ft (10,670 m)	OES
Research Vehicle Pre-launched	Mach .8	35,000 ft (10,670 m)	CM, SN, or SN/OES
Research Vehicle Launched from C-5A	Accelerating to Cruise Mach Number	Climbing to Cruise Altitude	
Research Vehicle in Cruise Flight	Cruise Mach Number	Cruise Altitude	CM, SN, or SN/OES
Power Off, Maximum L/D Glide Descent to Base	Decelerating from Cruise Mach Number	Descending from Cruise Altitude	CM, SN, or SN/OES
Approach and Landing at Base	Mach 3 or Below	60,000 ft (18,290 m) or Below	OES

(U) FIGURE 6-19
MACH 6 ESCAPE SYSTEM UTILIZATION

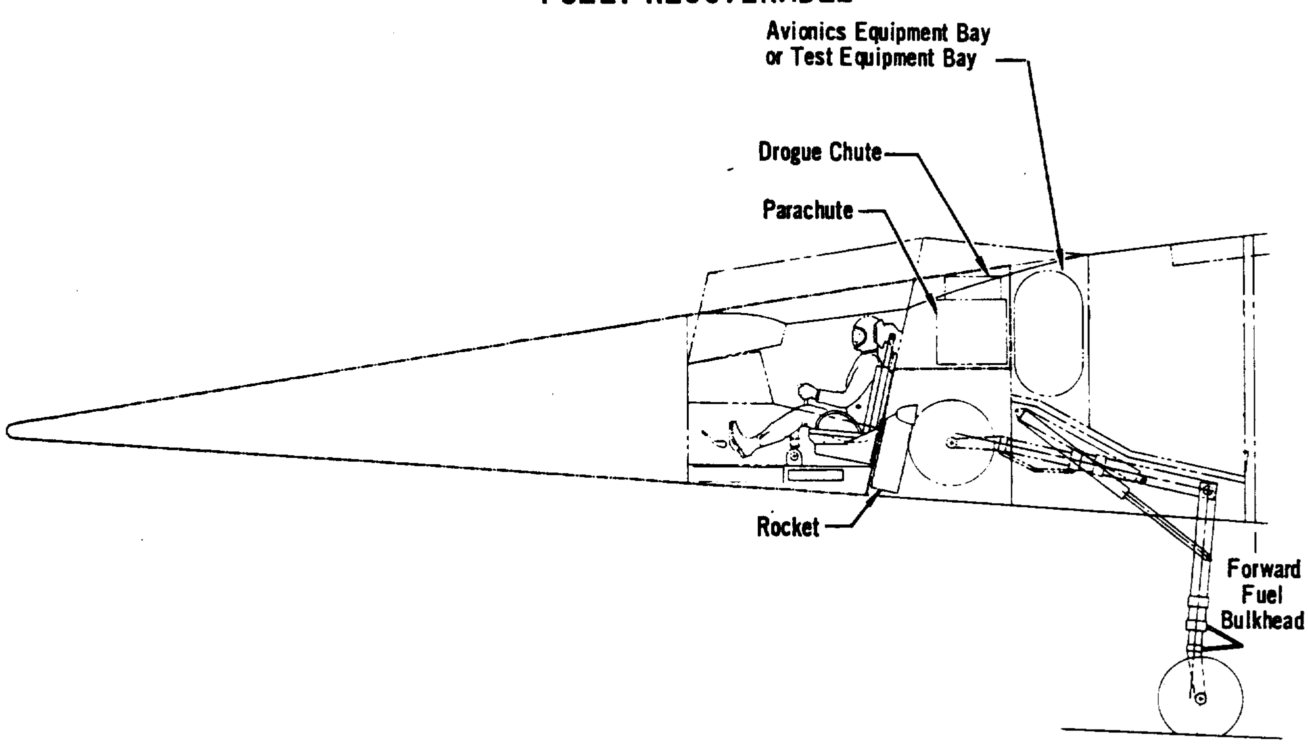
Situation	Vehicle Speed	Vehicle Altitude	Preferred Escape System
Engine Start and Taxi	0 - 40 kt (0 - 74 km/hr)	0 ft	OES
Takeoff and Climb Out	Less Than 600 kt (1112 km/hr)	Less Than 60,000 ft (18,290 m)	OES
Climb to Test Speed and Altitude	Accelerate to Mach 6	Climb to Cruise Altitude	CM, SN, or SN/OES
Research Vehicle in Cruise Flight	Cruise Mach Number	Cruise Altitude	CM, SN, or SN/OES
Power-Off Maximum L/D Glide Descent to Base	Decelerating from Cruise Mach Number	Descending from Cruise Altitude	CM, SN, or SN/OES
Approach and Landing at Base	Mach 3 or Below	60,000 ft (18,290 m) or Below	OES

*See Figure 6-17.

(U)FIGURE 6-20
SYSTEM 1 COCKPIT CREW MODULE – FULLY RECOVERABLE

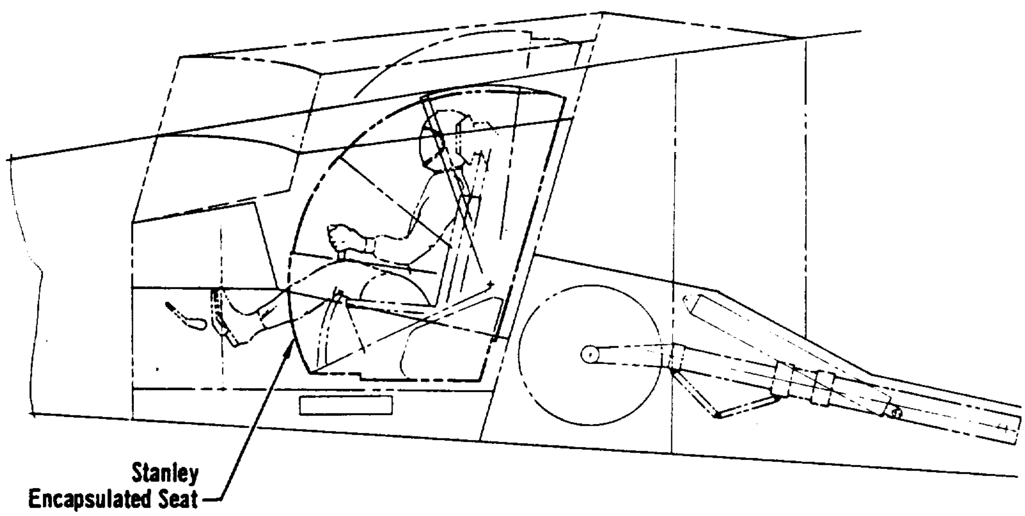


(U)FIGURE 6-21
SYSTEM 2 SEPARABLE NOSE CONE
– FULLY RECOVERABLE



6.5.2.3 (U) Encapsulated Seat (ES) - The encapsulated ejection seat, Figure 6-22, like the two previously discussed systems, provides full protection from high altitude, thermal and windblast hazards. It also provides the added survival equipment, shelter from adverse weather conditions, and flotation during water landings.

(U) FIGURE 6-22
SYSTEM 3 ENCAPSULATED SEAT

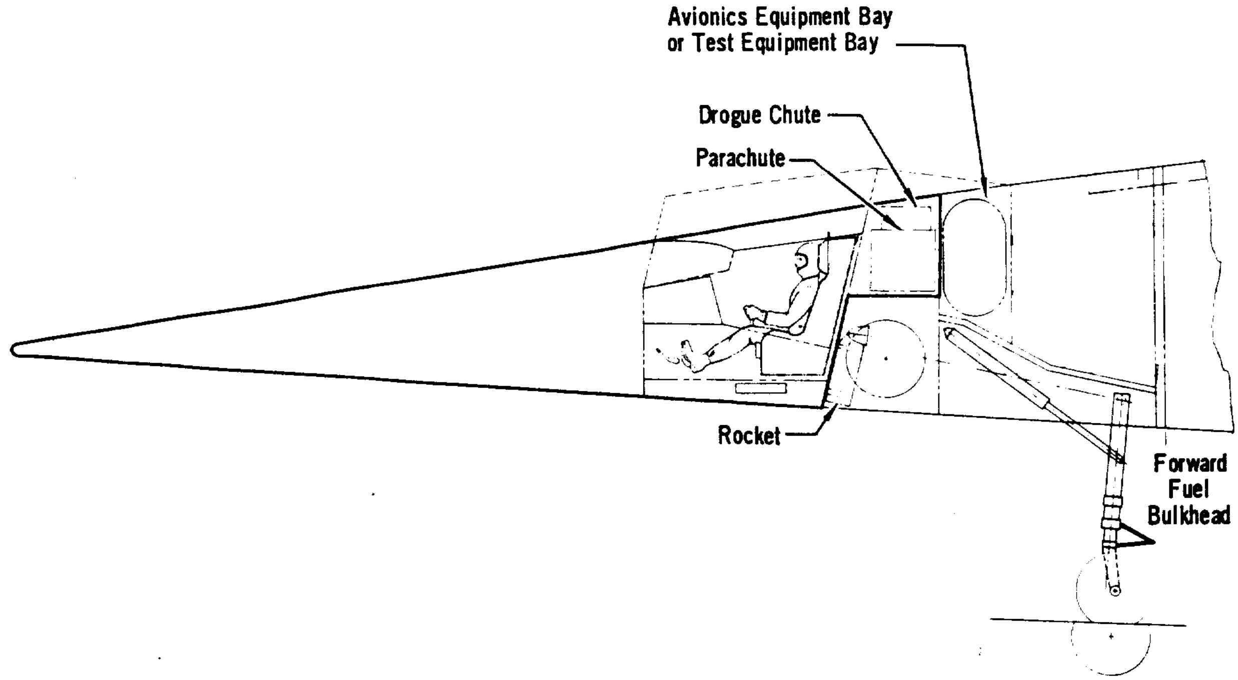


(U) The encapsulated seat systems presently developed require additional cockpit volume. The crew escape system of the B-70 required a considerable increase in crew station volume to provide good crew visibility and mobility. The encapsulating device was behind the crew seat and the seat moved aft and tilted into the encapsulating device before ejection. The Stanley seat used in the B-58 aircraft does not require any lateral or rotational motion of the crewman before encapsulation, but the crew visibility and mobility are reduced. The physical size of the encapsulated seat requires a larger cockpit and tends to limit the cockpit arrangement possibilities. Weight of the existing systems average approximately twice that of the open ejection seat, Reference (26).

6.5.2.4 (U) Separable Nose with Open Ejection Seat (SN/OES) - A fourth escape system consists of a separable nose module equipped with a low speed ejection seat, Figure 6-23, for the recovery phase of the escape. This system could provide maximum crew safety for minimum weight penalty.

(U) Pyrotechnic severance of the nose module would be followed by ignition of two solid motor rockets to propel the module from the damaged aircraft. Module stabilization devices would be deployed at separation. Drogue parachutes and stabilization devices would slow the module to a safe speed and altitude for the low speed open seat ejection. The seat would be about 100 lb (45 kg) heavier than a non-ejectable seat and would be ejected automatically at a preset altitude or speed. The pilot is parachuted to earth in a conventional manner. At low altitude and low speeds, the escape sequencing time is reduced by providing immediate seat ejection without nose module separation.

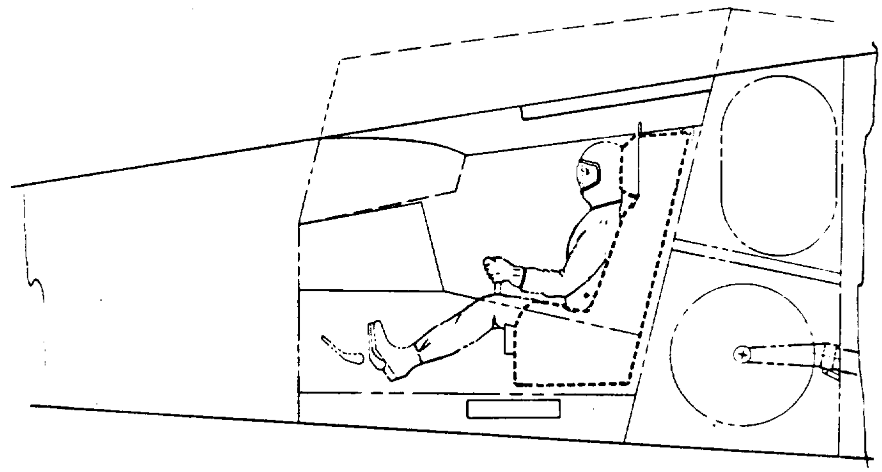
(U) FIGURE 6-23
SYSTEM 4 SEPARABLE NOSE CONE – OPEN SEAT



6.5.2.5 (U) Open Ejection Seat (OES) - The final system investigated was the open ejection seat, Figure 6-24. The ejection seat designed for the X-15 experimental aircraft was to be used from a low speed (90 kt, 167 km/hr), low altitude condition to Mach 4.0 at 120,000 ft (36,580 m). A study of the X-15 accident potential revealed that 98% of the X-15 accidents would occur within the ejection seat safe escape envelope and that the 2% beyond the envelope could be countered by staying with the aircraft until it descends and/or decelerates to the ejection envelope. Although the HYFAC research vehicle mission profile exceeds the X-15 performance envelope, the similarity of missions would indicate an accident potential comparable to the X-15. This comparison could justify the use of an ejection seat for the research vehicle.

(U) Various open ejection seats have been considered, but the two most likely candidates for the research vehicle would be the North American X-15 seat modified to provide zero altitude, zero speed escape capability, or the Lockheed SR-71 ejection seat. The McDonnell Douglas Advanced Concept Ejection Seat (ACES) system could possibly be made adaptable to the flight vehicle requirements.

(U) FIGURE 6-24
SYSTEM 5 OPEN SEAT



6.5.2.6 (U) Pressure Suit Requirements - In operational aircraft utilizing encapsulated seats or separable cockpit module crew escape systems, pressure suits are not required because the normal mission altitudes do not exceed the altitude at which decompression would present a crew hazard. However, the cruise altitude of the Mach 12 and Mach 6 research vehicles demands instant protection from depressurization, regardless of the escape system selected.

6.5.3 (U) ESCAPE METHOD RATIONALE - During the test flight, there are several time periods that present unique conditions for escape, as indicated in Figures 6-18 and 6-19. These times correspond to the periods of acceleration, cruise, or initial unpowered glide when the speed conditions exceed the limits of existing operational open ejection seat systems. Except for cockpit fires, the cockpit module protects the crewman from all hazards if cockpit pressure integrity is maintained. Conventional standards of fabrication used on modern aircraft and spacecraft to limit cockpit fires would be used to minimize this problem. The cockpit module would also provide safe descent and landing without the hazards of a personnel parachute landing. Additional survival equipment is carried and inherent flotation is achieved for water landings. The cockpit module also acts as a foul weather shelter. However, the nature of the test operations would tend to reduce the required ground survival time before rescue. The major portion of the Mach 12 test flight is routed over the southern states of the continental United States and no major bodies of water are crossed. A complete radar tracking network will be in operation along the entire flight path with continuous ground communications. In event of an in-flight emergency necessitating ejection at any point along the test flight path, rescue teams would be dispatched instantaneously to the predicted impact area. With a concerted search effort, radio beacons and brightly colored parachutes, an early rescue could be anticipated. This reduces the need for a sheltering structure and additional survival gear other than that carried in an ejection seat survival kit.

(U) The reasons for a module escape system, except for high speed escapes, do not appear to be worth the severe weight, complexity, and cost penalties to the basic vehicle when compared with other systems. The module system testing and development costs would be prohibitively high for the limited number of research vehicles.

(U) Statistics reveal that most accidents occur at the start or end of the flight with the majority of ejections occurring below 10,000 ft (3040 m) and 250 knots (463 km/hr). A similar projection statistic can be made for the research vehicle concepts developed during this study on the basis of the X-15 accident study in which it was indicated that the ejection seat design envelope was capable of countering 98% of the projected accidents. On this assumption, an open ejection seat would operate most efficiently in the high risk area of the flight profile at a reduction in cost, weight, and complexity.

(U) The encapsulated seat extends the open seat ejection envelope only slightly and at an increase in cost, weight, and complexity, which is not considered to be justified.

(U) A cost evaluation of four escape concepts investigated in this study were previously compiled in Reference (27). A quantitative economic evaluation was made of the crew cockpit module 1, separable nose module 2, the encapsulated seat 3, and the open ejection seat 5. The results show a relative cost of the separable module systems 1, 2, and 4 to be approximately 7 times greater than for the open ejection seat system 5. Also indicated is a weight increase of approximately 5 to 1 over that of the open seat ejection system. Although the encapsulated ejection seat system penalty was less, it was considered significant. Another study reported in Reference (28), indicates a large savings in weight and cost for a single place open ejection seat system when compared to a cockpit module type escape system. The cockpit module weight was 837 pounds or 2.7 times heavier than the selected open ejection seat. The cockpit module was also rated as approximately ten times more costly than the open ejection seat.

(U) All systems considered have a zero altitude and zero speed survival capability which is most easily incorporated into the open ejection seat system. The modular and encapsulated escape systems with this capability require large rocket motors with reasonable acceleration limits to eject the large mass of the module safely. The additional complications of a pyrotechnic severance system and self righting flotation and shock attenuation equipment create severe cost, weight, and structural penalties for the aircraft when compared to the open ejection seat and pressure suit system.

(U) One alternate condition presenting a problem to an open ejection seat survival, other than high speed, is a major fuel tank rupture while in the vertical takeoff (VTO) position. If all tanks rupture and the fuel and oxidizer mix immediately, the resulting fire ball could be as large as 450 ft (137 m) in diameter and could last for as long as 38 seconds, Reference (29). A crew module escape system could be lofted high enough and far enough away from the fireball to permit a safe recovery for the pilot. With the pilot only 40 ft (12 m) above the ground and 65 ft (20 m) above the fireball center, a horizontal seat ejection while on the VTO pad would not permit pilot survival. The seat recovery system would not have time to operate before impacting the ground.

(U) A possible alternate system for a single vehicle which would be configured for VTO, is a rocket powered escape tower, a separable nose module, and an open ejection seat. The tower would be used only during VTO operation and would normally be ignited and released after a successful VTO was accomplished. The separable nose module would be accomplished by a pyrotechnic severance of a splice plate at the forward fuel bulkhead and forward fuselage aft ring. The escape would be made by severing the splice plate and igniting the escape tower rocket to lift the module up to altitude sufficient for a safe seat ejection with adequate recovery time.

(U) Another alternate system would be to design the cockpit and seat to provide for a lofted seat trajectory from the VTO launch position. The seat rails would be canted forward to provide the proper lofted trajectory and would result in a longer cockpit, larger ejection clearance envelope requirement, and an inefficient use of aircraft volume. The special seat, rails, and seat attachment fittings would entail additional weight and cost penalties over the basic cockpit design.

(U) The probability of the catastrophic failure occurring using normal launch precautions and inspections is remote and does not warrant the required weight and cost penalties associated with the crew module for the basic configuration. Providing a separation tower and a nose separation capability for one vehicle which would be designated for VTO operations could prove feasible, but the development costs to demonstrate the flight article would be excessive.

(U) Another critical condition for research vehicle crew escape occurs during an airlaunch mission after C-5A takeoff and during climb out to launch altitude and speed. All ejection systems would be designed to clear the C-5A (primarily the wing leading edge), Figure 6-25, if actuated while the research vehicle is still mounted on the C-5A wing pylon.

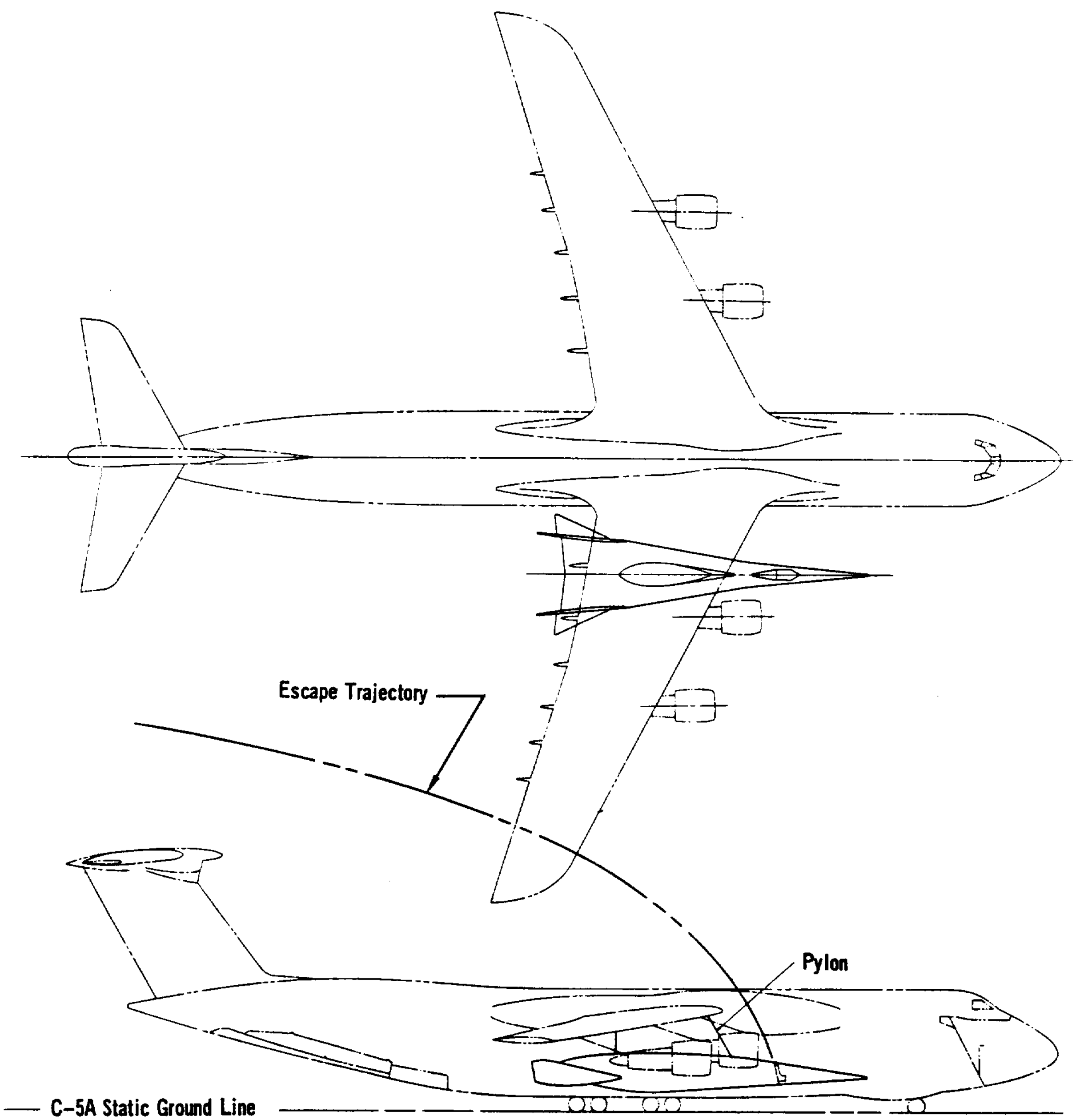
6.5.4 (U) MACH 6 RESEARCH VEHICLE CREW SAFETY - The rationale presented is basic to the Mach 12 research vehicle, but is also applicable to the Mach 6 vehicle. There is no VTO requirement for the Mach 6 vehicle and the test flight path is shorter. The time required to slow the aircraft to a speed and altitude compatible with an existing ejection seat system envelope is greatly reduced.

6.5.5 (U) RECOMMENDATIONS - An open ejection seat similar to that used in the X-15 research vehicle or the SR-71 aircraft with the pilot dressed in a full pressure suit is the preferred escape system for this program. The majority of the accidents which will cause the pilot to abandon the aircraft occur at the lower end of the flight spectrum. The seat zero altitude and zero speed to 600 kts (1112 km/hr) ejection capability will provide pilot escape in this phase of flight and cover most accidents. The small percentage of accidents occurring outside the ejection envelope can be countered by remaining with the vehicle until the ejection envelope is attained. Although the crew module would provide a slightly improved escape capability, the cost and weight penalties are prohibitive for the limited number of vehicles required.

6.6 UNUSUAL SUPPORT OR MAINTENANCE REQUIREMENTS

(U) The hypersonic research flight facilities including the research option configurations will require a normal amount of support and maintenance to operate. Most of the equipment and facilities required for the support and maintenance are

(U) FIGURE 6-25
MACH 12 AIRCRAFT INSTALLATION ON C-5A



either currently available or are not unique to test flight programs. This section identifies those few significant items considered to be unusual.

(U) The VTO research option of the Mach 12 vehicle will require a launch complex consisting of support mechanism and launch pad. The launch pad will require a flame deflector and exhaust disposal system to channel the hot exhaust gases away from the launch pad area.

(U) The fly-by-wire flight control system concept, if employed in either vehicle, would require the use of a test set employing an analog computer to check out the closed-loop performance of the control system. The analog computer is required in order to represent the airframe dynamics and sensor outputs in the closed-loop performance test.

(U) All vehicle configurations employ cryogenic propellants which can be trucked in, and the aircraft fueled directly from the trucks. Provisions may have to be made to handle off-loaded fuel in the advent of aborted missions, or following ground runs.

7. SAFETY STUDIES

(U) Safety studies were performed to evaluate the safety aspects of each flight research concept. The studies encompassed the two basic vehicles, i.e., Mach 12 and Mach 6 concepts, and the configuration options studied in adapting these vehicles to accommodate various research packages. The purpose of these studies was to identify hazardous conditions which must be eliminated from the vehicle concept or adequately controlled. The scope of the safety studies consisted of the following.

- Identification of possible safety interface problems.
- High-lighting special areas for safety consideration, such as system limitations, risks, and man-rating requirements.
- Defining areas requiring further safety investigation during advanced development activities.

(U) The safety studies were limited to those considerations which are unique to the design and operation of the candidate hypersonic flight research facilities. Safety aspects which are universal or common to flight vehicles in general were not considered unless they produced more drastic effects because of the hypersonic flight regime. Safety studies were performed for each of the major vehicle systems, i.e., structure, thermal protection, propulsion, and propellants. The following paragraphs describe the study methods.

(U) Hazard Identification - Potentially hazardous conditions were identified that could cause injury or death to flight/ground personnel, or damage to or loss of equipment/property.

(U) Hazard Effects - The effects of the potential hazards on system operation, interfacing systems, vehicle, and flight/ground operations were studied to determine their relative severity.

(U) Hazard Causes - The most probable causes or conditions which could create the potential hazards were identified for each hazard.

(U) Action Required - Action(s) required to eliminate or reduce the hazard were identified. These actions, in order of preference, were as follows:

- Design for minimum hazard - Select appropriate design features; e.g., fail safe, redundancy, etc., throughout the design phases.
- Safety devices - Reduce known hazards which cannot be eliminated to an acceptable level through the use of appropriate design safety devices; e.g., fire suppression, self sealing tanks, etc.
- Warning devices - Employ warning devices for the timely detection of the condition and the generation of an adequate warning signal where it is not possible to preclude the existence or occurrence of a potential hazard.

- o Special procedures - Develop special procedures, e.g., operating/maintenance instructions, etc., where it is not possible to reduce the magnitude of an existing or potential hazard through design, or the use of safety and warning devices.

(U) The results of this study indicate that all of the identified hazards and their effects can be either eliminated or reduced by the actions described above. These actions should be implemented to provide the optimum degree of safety within the constraints of operational effectiveness, time, and cost. A systems safety program should be integrated into all phases of the system design, development, production, and operations in order to provide a disciplined approach to control safety aspects.

(U) The identified hazards associated with the various vehicle systems were analyzed to determine their probable effects, should they occur during a mission. The results of these evaluations are summarized in Figures 7-1 and 7-2 for the Mach 12 and Mach 6 vehicles, respectively. The hazards produce a variety of effects which are identified in the second column. These effects in turn require emergency actions which then lead to the final results shown in the last column. The analysis was performed for the vehicles during self sustained flight, i.e., after airlaunch of the Mach 12 vehicle and after takeoff of the Mach 6 vehicle. Malfunctions occurring prior to that time would abort the mission.

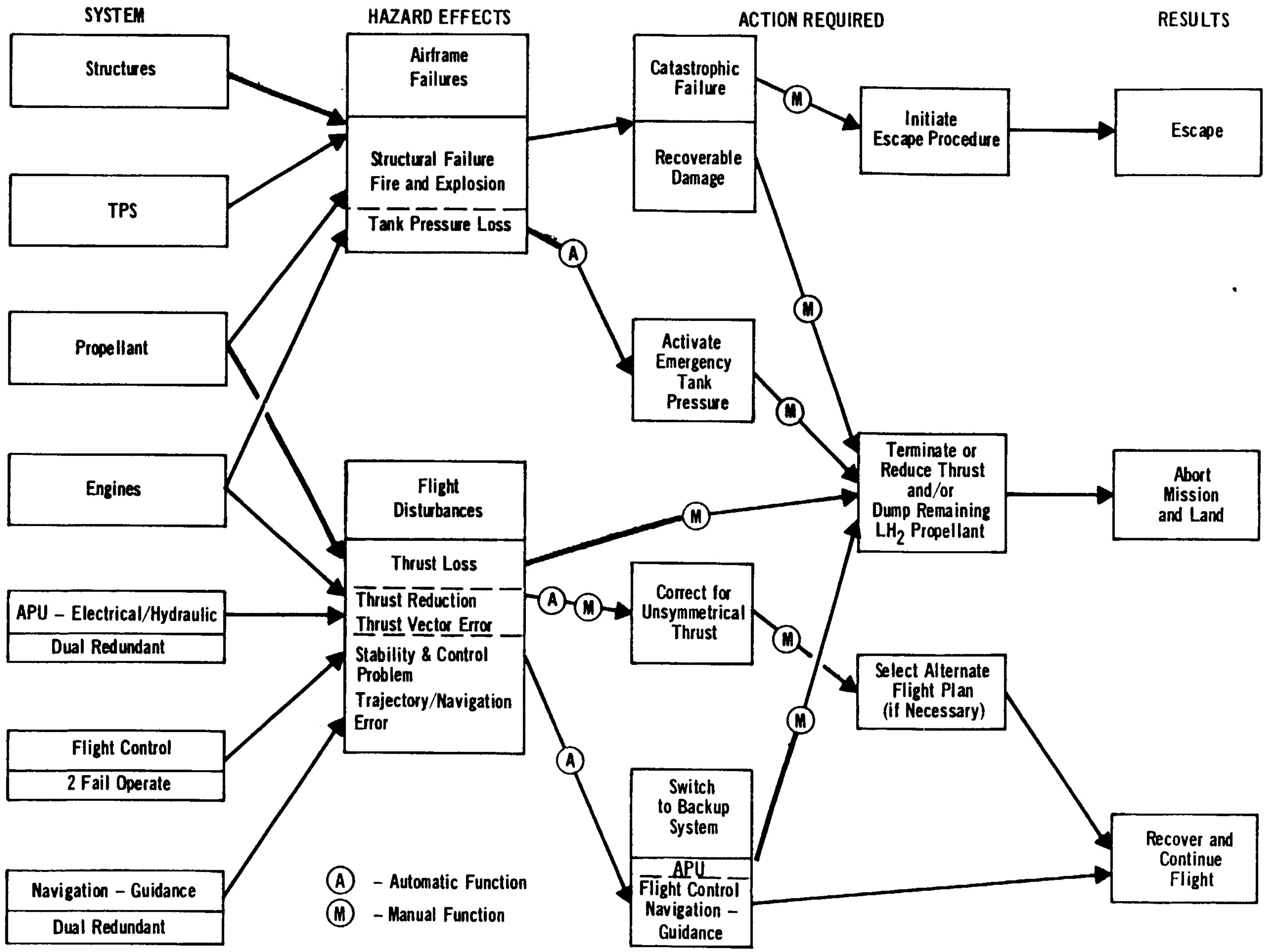
7.1 FIRE AND EXPLOSIONS

(U) Fire and explosive hazards have been investigated for both the Mach 6 and Mach 12 basic flight vehicles. The result of this investigation is a high level of confidence in respect to minimizing the hazards associated with cryogenic propellants.

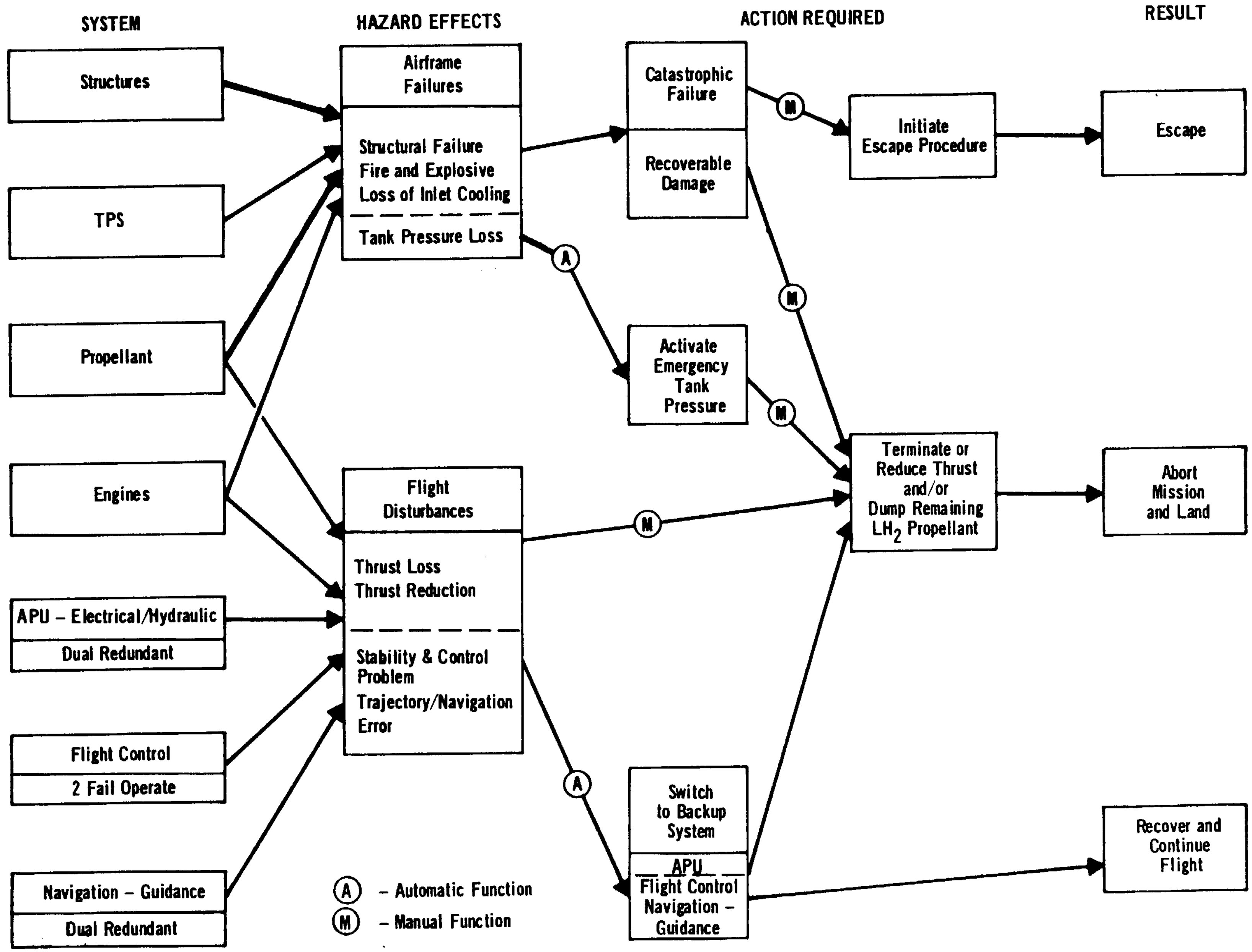
(U) It is anticipated that, given close attention to fire and explosion hazards from the initiation of vehicle design, a total system concept can be established which will provide the required level of flight system and personnel safety. Specific guidelines which should be considered are:

- o Eliminate closed compartments where vapors can collect; provide for air circulation to dilute small leaks (7.1.2).
- o Incorporate vapor and fire detection and suppression systems in tankage, flow, and propulsion system areas (7.1.2).
- o Provide redundancy where practical for those components whose failure would be catastrophic; design to "fail-operational" mode where redundant approaches are not satisfactory (7.1.2).
- o Minimize potential sources of leakage through techniques such as minimum flanged joints, tank passthroughs (7.1.2).
- o Provide for remote servicing (7.1.1).
- o Eliminate handling of propellants in the C-5A (applicable to Mach 12 vehicle only) except during emergency procedures (7.1.2).

(U) FIGURE 7-1
MISSION HAZARD ANALYSIS
Mach 12 Vehicle



(U) FIGURE 7-2
MISSION HAZARD ANALYSIS
Mach 6 Vehicle



(U) The cause and effect of potential fires and explosions are discussed in the following section together with required safety considerations. The hazards are categorized in the following major areas:

- Ground Hazards

  LH_2 Spills
  LO_2 Spills
  Simultaneous LH_2/LO_2 Spills
  Tank Rupture
  Vapor Venting

- Flight Hazards

  Engine Fires/Explosions
  Feed System Failures/Leakage
  Propellant Dumping.

7.1.1 (U) GROUND HAZARDS - Hazards associated with handling of LH_2 and LO_2 impact both the research vehicle and the associated ground support facilities and personnel.

(U) LH_2 spills pose hazards due to their wide flammability range (4% to 74% by volume in air) and low required ignition energy. Ignition energy for a LH_2/air mixture is approximately 1/10 that required for a gasoline/air mixture at one atmosphere. If the hydrogen/air mixture is not confined, detonation is not likely, but should the spill occur in a four sided enclosure (for example in a U shaped bay on the ground), detonation is probable. Normal procedure for handling a hydrogen fire is to let it burn under control until the flow of hydrogen can be stopped. If the fire is extinguished without eliminating the flow of hydrogen, a hazardous combustible mixture will form which very probably will re-ignite and cause additional damage. Standard fire fighting procedures established in support of hydrogen fueled vertical launch vehicles must be employed.

(U) An LO_2 spill is very dangerous in the presence of any combustible material, although LO_2 is not a flammable liquid. The use of asphalt runways or servicing areas can become extremely shock sensitive if sufficient oxygen is absorbed, and many other normally "safe" materials become flammable in the presence of high oxygen concentrations.

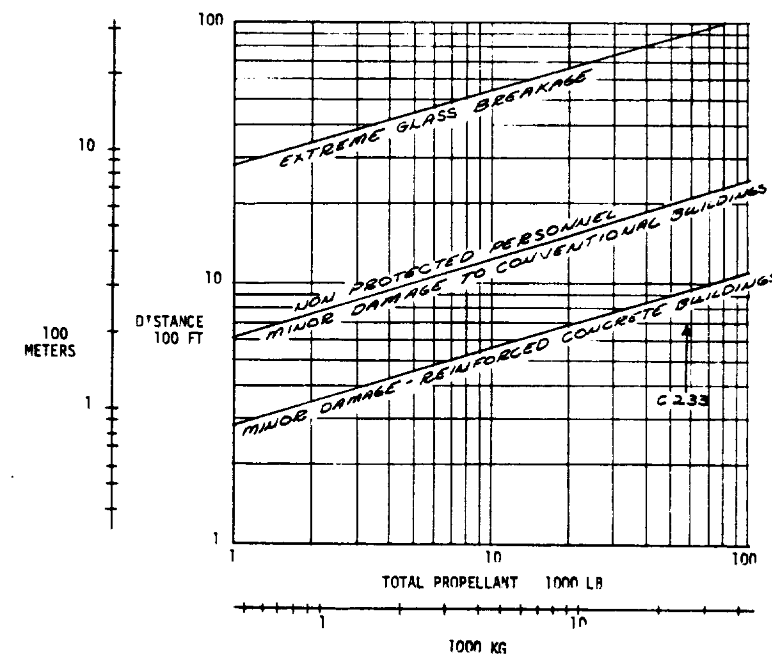
(U) For either spill, LO_2 or LH_2, the required safety procedures are essentially the same. Vapor and fire detection equipment is essential for determination of leaks at critical areas and rapid detection of fires. Attention must also be given to servicing hardware and procedures to prevent accidental spills during connect and disconnect of servicing lines and associated hardware in addition to eliminating combustible materials in or near the servicing area.

(U) In order to eliminate the potential of simultaneous spills, the propellants should be loaded sequentially (LO_2 first) to eliminate simultaneous handling of the propellants. Applicability of a dual or LO_2 spill is limited to the Mach 12 vehicle.

(U) The simultaneous release of LH_2 and LO_2 resulting from tankage rupture or similar large failures discussed in Section 7.3, present the greatest danger to both the vehicle and personnel during ground operation. Such a failure would, in all probability, result in fire and explosion. Severity of the hazard is a function of quantity and nature of the spill. For a spill of 48,270 lb (21,950 kg) of LO_2 and 8,040 lb (3,650 kg) of LH_2, which is the full propellant load for the Mach 12 vehicle, the potential yield is equivalent to 17,400 lb (7,900 kg) of TNT (References (29 and (30)). For that yield, the minimum safe operating distance for non-protected personnel and conventionally constructed buildings is approximately 2100 ft (640 m). By providing explosion resistant structures and personnel protection, the minimum safe distance can be greatly reduced dependent on the degree of protection and surrounding terrain. Design criteria, such as presented in Figure 7-3, must be used to establish required distances between a loaded vehicle and surrounding buildings and personnel.

(U) Similar fire/explosion hazards exist if vapor venting is permitted near the aircraft. This potential danger can be alleviated by using remote vent lines exhausting to the atmosphere through flare stacks or into controlled burn ponds. Quick disconnect couplings would be provided to allow rapid vent line removal.

(U) FIGURE 7-3
SAFE OPERATING DISTANCE OVER PRESSURE EFFECTS



7.1.2 (U) FLIGHT HAZARDS - Airborne fire and explosion hazards are mainly concerned with propellant feed, propulsion, pressurization, and tankage system leakage. The location and extent of leakage determines the relative danger to aircraft and personnel.

(U) Two of the more serious hazards are engine explosions or fires in the engine area. These hazards are applicable to all of the vehicles. Causes of an engine explosion or fire include failure of a flow system component or leakage. Response to a critical-level explosion would be to abort the mission and land as soon as possible, to preclude further or secondary damage. Response to a catastrophic-level explosion requires that the crew abandon the vehicle. Response to fires should include a crew warning system and countermeasures such as automatic fire extinguishments. Further response might require engine shutdown, for which the direct effect is loss of thrust with the resulting effects discussed in Section 7.2, Vehicle Control.

(U) Feed system failures and leakage hazards can be minimized by designing with the ground rule of providing for self containment of leakage; e.g., jacketing components or providing purge capabilities to disperse/dilute vapor concentrations. Sensing equipment in closed equipment bays and near flow system components would include temperature monitoring and vapor detection sensors in conjunction with provisions for automatic fire suppression. Actual quantity, type, and location of sensors would require a detailed experimental investigation as to vapor dispersal rate and sensor response rate. Critical components, such as feed pumps, control valves, and pressure regulators, will require sufficient redundancy to permit system operation with a single failure. The extent of redundancy is also dependent on demonstrated component/subsystem reliability which is in a continual state of improvement. O_2 monitors will be located in each LH_2 tank to permit detection of hazardous vapor concentrations.

(U) Propellant dumping presents fire and explosion hazards to both the aircraft and ground installations, depending on dump rate, vehicle speed, and altitude. The greatest hazard to ground personnel will occur if the propellants must be rapidly dumped before the flight research vehicle or the vehicle/C-5A combination has gained sufficient altitude to permit vaporization and dispersal of the propellants before they contact the ground. It is anticipated that the potential of danger to ground personnel/facilities is greater for a LO_2 dump than for a LH_2 dump due to the high vaporization rate of LH_2 and its buoyancy effect in contrast to the cold O_2 vapor (heavier than air). It is also likely that LH_2 will ignite when dumped from the aircraft. Because of the very low radiative energy of an LH_2 flame, the hazards associated with the flame front will be minimal if the LH_2 is dumped near the aft end of the aircraft (flight research vehicle or C-5A if in the unlaunched condition). For the Mach 12 vehicle, LO_2 would be dumped first to reduce the weight within acceptable limits for landing. Dumping LH_2 is not required from a weight standpoint for the Mach 6 vehicle, but will greatly reduce hazards. For the Mach 12 vehicle only 50% of the LH_2 need be dumped prior to landing, but total dumping would be desirable. Accurate determination of the hazards requires actual flight testing to establish allowable dump rates and operational procedures as a function of altitude and flight speed.

7.2 LOSS OF VEHICLE CONTROL

(U) The hazard created by loss of vehicle control, i.e., the ability to maintain a desired flight path without exceeding structural limitations, was an essential consideration in the safety study. The many potential malfunctions which could cause a control problem require an extensive analysis of design details, which is beyond the scope of this study. The results presented herein were limited to control problems caused by basic malfunctions or anomalies in the vehicle's major interfacing systems. Two general types of control problems were considered. First, those caused by the inability to control the vehicle due to failures such as loss of rocket gimballing, engine thrust, or aerodynamic surface control. Second, those which are more transient in nature caused by some initial disturbance such as control system hardovers, inlet unstarts, or natural phenomena (gusts, wind shear, etc.). The primary consideration in the analysis was to determine whether or not the control problems could end in catastrophic results such as structural failure from excessive load factor or non-correctible out-of-control situations. The results indicate that recovery is probable from control problems (assuming single failures). The causes of control problems were grouped into three areas for discussion as follows:

- Inlet unstarts
- Engine thrust loss
- Flight control system malfunctions.

7.2.1 (U) INLET UNSTARTS - Effects of the moments produced by inlet unstart of the airbreathing propulsion systems were studied. Inlet unstart is a phenomena of external airflow spillage, and occurs when the inlet airflow supply exceeds the sum of the engine demand plus maximum bypass capability. Some of the major causes are pitch excursions to large positive angles such that captured airflow increases beyond demand, or bypass system failure, or engine transients which decrease demand. Design techniques to preclude this event might include bypass systems able to handle very large airflows, and propulsion system controls able to rapidly increase engine airflow demand (such as by opening the engine nozzle throat) in response to an imminent inlet unstart.

(U) The turboramjet of the Mach 6 vehicle and the CSJ and SJ modifications to both the Mach 12 and Mach 6 vehicles are the only candidate propulsion systems that have potential for causing significant moments due to unstart. An inlet unstart of the turboramjet inlet of the Mach 6 vehicle would cause an increase in drag on the unstarted side of the bifurcated inlet and a decrease in gross thrust, while maintaining nominal drag on the started side of the inlet. The primary effect of this event would be a yaw moment pulling the nose toward the unstarted inlet. A simplified analysis indicated that approximately 10° of rudder deflection would counteract this moment at any supersonic speed. An inlet unstart of any module on the CSJ and SJ engines was estimated to propagate to the other modules and thus unstart the complete engine. The effect of this action would be a nose down pitching moment. Analysis indicated that this moment could be counteracted by very small elevator deflections at all hypersonic speeds.

7.2.2 (U) ENGINE THRUST LOSS - Effects of the moments produced by loss of engine thrust were studied. This phenomena could occur on any of the various vehicles, as a result of fuel system malfunction which would terminate propellant flow, or of engine component failure which would terminate propellant flow, or ignition failures.

(U) The basic Mach 12 vehicle uses five rocket engines installed in a spanwise row at the vehicle trailing edge for propulsive power. The most severe moment would result from either of two conditions: (1) both engines on one side inoperative, or (2) both engines on one side plus the center engine inoperative. A combination of gimballing the operative engines ± 4 degrees side to side (square pattern), and deflecting the rudders approximately 80% could handle condition 2, while approximately 55% of the available rudder could handle condition 1. For the J2S powered alternate of the basic Mach 12 vehicle, with a single rocket engine mounted on the vehicle centerline, thrust loss would not cause any moments.

(U) The basic Mach 6 vehicle uses one turboramjet for propulsive power, installed on the vehicle centerline. Thrust loss of this engine would thus not cause any noticeable moments.

(U) Modifications to the basic vehicles involve a variety of propulsion systems. Effects of the moments produced by thrust loss in these systems is summarized in Figure 7-4. The effects indicated in that figure are for the unique controllability aspects of the modified vehicle.

(U) FIGURE 7-4 THRUST LOSS ON MODIFIED VEHICLES

Basic Concept	Modification	Propulsion System Description	Cause of Severest Moment	Vehicle Controllability
Mach 12	HTO, VTO	Same as basic vehicle except shorter nozzles	Same as basic vehicle	Requirements less than basic vehicle, since thrust with short nozzles is less
	CSJ, SJ	Same as basic vehicle plus CSJ or SJ added	CSJ or SJ inoperative	Small stabilator deflection needed ($\approx 3^\circ$)
	TPS, Armament, Staging	Same as basic vehicle	Same as basic vehicle	Same as basic vehicle
	Subsonic TJ	Same as basic vehicle but rockets inoperative, with 2 x JT4 TJ added	One TJ inoperative	Very small rudder deflections needed ($\approx 2^\circ$)
	RJ	Same as basic vehicle plus test bed RJ added	Same as basic vehicle	Same as basic vehicle
Mach 6	Armament, TPS	Same as basic vehicle	Same as basic vehicle	Same as basic vehicle
	JZ6C	Basic TRJ replaced with 1 x JZ6C TRJ (all H_2)	Same as basic vehicle	Same as basic vehicle
	CSJ	Same as basic vehicle plus CSJ or SJ added	CSJ or SJ inoperative	Small stabilator deflection needed ($\approx 10^\circ$)
	RJ	Same as basic vehicle plus test bed RJ added	Same as basic vehicle	Same as basic vehicle

7.2.3 (U) FLIGHT CONTROL SYSTEM MALFUNCTION - The control of the vehicle is critical throughout all phases of the flight. Positive control is required during separation from the launch vehicle to prevent collision and possible loss of both vehicles. Precise control is also required during landing. Between these times, control must be adequate to modulate the flight path as desired and to prevent the occurrence of excessive loads upon the vehicle. The potential malfunctions that may occur in the aircraft flight control system are generalized in Figure 7-5. The flight control system is designed to remain operational with these failures. These design features are described further in the following paragraphs.

(U) The principle controls of both the Mach 6 vehicle and the Mach 12 vehicle are the aerodynamic surfaces. Both concepts employ a fly-by-wire (FBW) control system. The Mach 12 vehicle, in addition to the aerodynamic surfaces, can also be controlled in attitude by thrust vector (gimbal the rocket engines during boost). The thrust vectoring is governed by the FWB system in order to have an integrated flight control scheme. The aerodynamic surfaces have greater authority than the thrust vector, which is used as a delta to obtain larger attitude changes, if necessary, during boost.

(U) FIGURE 7-5
POTENTIAL FLIGHT CONTROL MALFUNCTIONS

System	Potential Malfunction	Design Features Inhibiting Failure Effects
Hydraulic	a. Pump failure b. Ruptured lines or components	Separate primary and secondary systems
Electrical	a. Generator failure b. Broken wire or circuit breaker	Separate generators and primary and secondary busses
Avionic	a. Failed component causing circuit malfunction	Triple redundancy and fail operational circuits

(U) The components of the FBW system, e.g., sensors, transducers, hydraulic and electrical sources, actuators, and avionics, must be designed to achieve the utmost in reliability with redundancy employed to the degree necessary to provide fail operational performance. Built-in-test (BIT) should be provided to detect and isolate malfunctioning components prior to and during flight. In-flight continuous performance monitoring advises the pilot on system status. Redundancy is required such that component malfunctions cause the affected channels to "fail-to-zero", thereby precluding system "hardovers."

7.3 LOSS OF STRUCTURAL INTEGRITY

(U) Flight safety is one of the major influences in selecting design criteria, design philosophy, structural concepts, and redundancies in the design of aircraft structure. Weight and cost elements presented herein are based on primary structural elements incorporating present day operational aircraft flight safety features. Several discrete elements of hypersonic aircraft structure that are distinctly different than any of the present day aircraft and will require unique design solutions to obtain required structural integrity. Specific elements peculiar to hypersonic aircraft that fit into this category are: cryogenic propellant tanks, thermal protection system heat shield, and heat exchangers (inlet and engine).

7.3.1 (U) BASIC AIRFRAME AND CRYOGENIC TANKAGE - Individual structural elements that must be designed with "fail safe" requirements are those that could cause vehicle loss if structural failure were to occur. Possible causes of cryogenic tank structure failure include foreign object penetration and initiation and propagation of cracks originating at manufacturing defects or stress concentrations. Incorporation of longitudinal stiffeners and circumferential rings assures a fail-safe design by preventing growth of cracks to catastrophic proportions. The possibility of fire and explosion due to extensive fuel leakage is minimized by providing a double seal against fuel leakage. A bladder is used to contain the fuel and, in addition, the tank structure is sealed as additional protection against leakage.

Potential problems arising due to fuel leakage are discussed in greater depth in Section 7.1.2. Leakage in the common bulkhead between the LO_2 and LH_2 tanks would result in a catastrophic failure, and therefore would require a redundant leak protection system.

7.3.2 (U) TPS COMPONENT FAILURES - A study of the TPS has shown that if the heat shield is lost from the aircraft at high speed, a catastrophic failure could occur. For this reason, the heat shields must be of fail safe design. The attachment pattern is designed to provide adequate strength with 20-25% of the attachments broken or removed. Also, the panels are designed to stay in place, even though cracks or small foreign object damage should be introduced. The probability of significant TPS component failures during flight is considered remote, assuming proper preflight and postflight inspection procedures are conducted. Since the only TPS concept incorporated in the Mach 12 aircraft is the active system, potential in-flight failures are limited to the water supply, hot side insulation blanket, and/or external shingle. Investigating the effects of variations in water blanket saturation level on structural temperatures has been a major Phase III effort and is discussed in detail in Section 4.6.1. In summary, it was concluded that the probability of depleting the water supply is remote and any such occurrences would be local effects with resultant structural damage in terms of loss of strength, but not of catastrophic proportion. However, incorporation of a water sensing system is still deemed desirable. Damage to the hot side insulation blanket, resulting from either loss of material or severe degradation of material properties, would result in locally higher internal heat transfer rates. This would cause the premature depletion of water resulting in similar structural damage. Unless the area subjected to insulation damage is extensive, it is probable that heat transfer through the internal structure from the prematurely dry area to adjacent, still wetted, areas would maintain structural temperatures to levels where little permanent damage would result. Minor damage to the shingles (dents, etc.) could be tolerated as such damage would merely impose slightly higher heating rates to the interior and lead to effects such as those discussed above. However, if the shingle completely failed to the point of becoming detached, a serious condition would exist. It is likely that loss of a shingle would cause immediate loss of insulation and, as no conceivable warning system could furnish adequate reaction time, the aluminum structure would attain temperatures sufficient to yield the material. Consequently, the shingle design precludes the possibility of shingle detachment during high speed flight. Through the normal process of systematic flight envelope expansion, a high confidence level in shingle design should be established.

(U) Mach 6 aircraft external moldline areas incorporate passive TPS concepts to protect primary structure. Insulation and minor shingle damage on this aircraft, as on the Mach 12 aircraft, would not prove catastrophic, but could significantly reduce the design structural safety factors. In cryogenic fuel areas, fuel tank wall temperatures in excess of the design value would degrade the cryogenic foam insulation, boiling off significant quantities of fuel. JP fuel tank liners would suffer permanent damage from excessive wall temperatures created by a TPS component failure. Shingle detachment must be avoided on the Mach 6 aircraft as well as the Mach 12, although, at the lower speed, it would take the internal structure at least a few seconds of total exposure to aerodynamic heating to reach the yield point of the metal.

7.3.3 (U) INLET DUCT REGENERATIVE COOLING PANEL FAILURES - Structural failures involving regenerative heat exchangers are applicable to research vehicles equipped with airbreathing propulsion systems, e.g., the Mach 6 TRJ and the Mach 12 SJ or CSJ configurations. The potential causes of failure in the regeneratively cooled inlet structure of these vehicles is from the loss of coolant flow, either by system malfunctions or local failures.

(U) The loss of LH_2 coolant flow throughout the heat exchanger (at speeds of Mach 6 or greater) would result in heat rise causing structural damage which, if left unchecked, could spread causing complete loss of the inlet structural continuity. The loss of coolant flow could be caused by a flow stoppage, or by a local failure resulting in LH_2 spillage into the inlet. Spillage should not damage the inlet by reason of hydrogen burning, since the flow is limited; however, the downstream coolant passages could overheat because of the lack of a heat sink. Potential causes of local failures include inlet overpressure or excessive stress levels due to excursions outside the operating limits or inlet unstarts. The necessity to operate within specified limits, and to implement design features which provide the ability to do so, are applicable to all aircraft and are not unique to hypersonic research vehicles. Inlet overpressure from unstarts can occur within the safe operating envelope.

(U) As long as design coolant (LH_2 fuel) flow rates pass through the regenerative cooling panel, temperatures are maintained below 1550°F (1118°K). However, if an adverse condition severely inhibits fuel flow through one of these panels, temperatures would increase sharply and in the extreme case, result in burn through of the panel, creating a severe hydrogen leak and potential fire hazard. Such an overheat condition could be caused by a flow regulator valve failing in a closed or partially closed position, pump failure, blocked passage, or supply line/component rupture. Thermal and pressure sensors can be used to monitor cooling circuit performance which could automatically alert the pilot and activate a fire suppressant system if overheating occurs. With rapid deceleration to low supersonic or subsonic conditions and a fire suppressant system to prevent or minimize fire damage, it is felt that this type of failure can be controlled without unduly jeopardizing the safety of the pilot and aircraft. During the emergency deceleration, hydrogen coolant flow rates would be maintained to prevent overheating of undamaged areas.

(U) Inlet failures are potentially more serious for the Mach 6 TRJ vehicle because the inlet structure is integrated more directly into the airframe structure. Vehicles with inlets provided as an "add-on" modification, e.g., the SJ or CSJ options, could probably sustain failures which are catastrophic to the SJ/CSJ propulsion system without causing the loss of the aircraft.

(U) Design provisions required to eliminate or reduce the hazards associated with regenerative heat exchangers would include a failure detection system which would permit engine shutdown or fuel cutoff before the failure can propagate into catastrophic proportions. It is unlikely that a completely redundant coolant system could be installed because of the weight and complexity involved. Inlet overpressure problems might be alleviated by incorporating pressure relief devices such as blow-out doors.

7.3.4 (U) ENGINE COOLING - Potential structural damage to the research vehicle's various engines due to inadequate cooling was studied. Inadequate cooling could result from a wide variety of conditions discussed for each of the research vehicles below. Primary engines of the Mach 12 and Mach 6 basic vehicles are discussed first. This is followed by discussion of the engines of the CSJ and SJ modifications to both of the basic vehicles, for which the safety aspects are identical. None of the other engines have potential cooling failures that are unique to the candidate vehicles.

(U) Mach 12 Vehicle - This vehicle uses liquid propellant rocket engines of conventional design, cooling the nozzle and combustion chamber by circulating all of the hydrogen fuel flow through these components. Inadequate local cooling could result from reduced coolant flow due to blockage in the coolant passages or from leaks. Insufficient cooling could be tolerated for short periods of time, however, if not checked by shutting down that engine, serious structural damage would result to the engine. Overheat detectors could be used to warn of impending failure and allow time for corrective action. On the other hand, loss of all hydrogen flow to the engine would not cause overheating, since the heat load (due to combustion) would be removed at the same time as the coolant flow.

(U) Mach 6 Vehicle - The primary propulsion system of the Mach 6 research vehicle consists of a turboramjet engine and a mixed-compression inlet. The turbojet engine is aircooled during operation. At flight speeds above the TJ shut down speed (Mach 3.5) this component is shielded from the environment by closure doors, and windmilled by a small amount of air taken from the inlet and cooled with ramjet hydrogen fuel to approximately 1000°F. All elements of the inlet and ramjet engine are cooled with ramjet hydrogen fuel, except the foremost inlet ramp and wall surfaces which are radiation cooled.

(U) At flight speeds above Mach 3.5, an increase in the temperature of the windmilling air could damage the TJ components. The increased temperatures could result from improper operation of the air cooling heat exchanger, from leaks in the RJ inner wall heat protection. Extent of this damage would depend on the temperature reached.

(U) Two types of ramjet engine structural damage are related to heat protection. The first concerns the primary engine cooling system. This uses hydrogen heat exchanger panels similar to those of the inlet, and thus would be subject to the same type of panel failure discussed with the inlet thermal protection system, Section 7.3.3. Another type of RJ damage could arise as secondary damage due to a heat exchanger panel detaching in the inlet or RJ air duct. Such a panel could damage any of the RJ components in its path--fuel injectors, flame holders, nozzle--while it moves to the exit. It is estimated that duct overpressures would not occur from such a panel plugging the air passages, since the RJ nozzle control system would respond to any pressure rise by increasing the nozzle throat area.

(U) CSJ and SJ Modifications - Two types of CSJ or SJ engine structural damage are related to heat protection. The first concerns the primary engine cooling system. This uses hydrogen heat exchanger panels similar to those of the inlet, and thus would be subject to the same type of panel failure discussed with the inlet thermal protection system, Section 7.3.3. Another type of damage could arise as secondary damage due to a heat exchanger panel detaching from the inlet. Such a panel could damage any of the CSJ or SJ components in its path--fuel injectors, walls, nozzle--while it moves to the exit. It is estimated that duct overpressure could only occur from very large pieces (on the order of one foot (.3 m) across) plugging the air passages. Such large pieces are not likely.

7.4 GROUND HANDLING HAZARDS

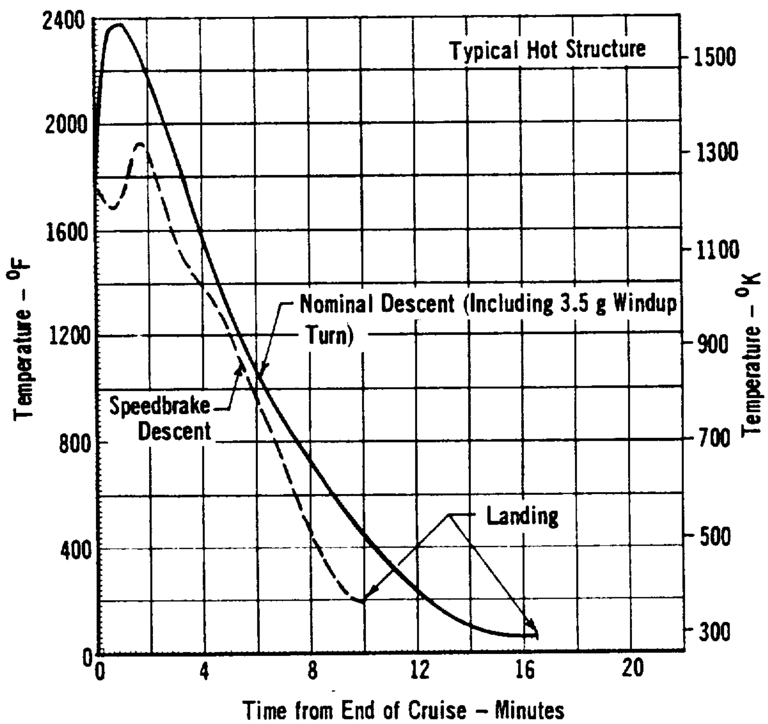
(U) Postflight ground handling of these aircraft will not require extensive safety precautions particularly since rapid turnaround times are not a prime consideration in research aircraft. Structural temperatures at landing have been determined to define areas on the research vehicles that would require special handling while conducting postflight inspection and maintenance procedures. Certain safety precautions presently used when servicing high speed aircraft such as avoiding landing gear areas and wearing gloves while reaching into compartments inside the aircraft's moldline immediately after landing will still be necessary.

(U) Mach 12 aircraft external surface temperatures at landing are primarily a function of the surface structural mass. Lightweight shingles cover the majority of Mach 12 surface areas where the TPS is used to protect the internally located load-carrying structure. These shingles respond rapidly to aerodynamic cooling effects encountered during descent. Upon landing, these shingles are near ambient temperature and rise only 5-10°F (3-6°K) above ambient while dissipating internally stored heat. Hot structure areas such as the delta tip controls and vertical tails, because of their greater mass, do not respond as rapidly to external cooling effects and are quite dependent on the descent profile as reflected in Figure 7-6. This figure shows the temperature response of a typical hot structure area (delta tip control) subsequent to Mach 12 cruise for two descent profiles. The nominal descent shown includes a 3.5 g windup turn at the conclusion of cruise, which significantly increases the temperature above its cruise level. However, the descent is sufficiently long to cool the surface down to essentially ambient temperatures at landing. If a minimum range (speedbrake) descent is conducted, structural temperatures at landing would be approximately 200°F (367°K). It can be deduced that hot structure areas will be hot enough to require handling with gloves following a landing when the descent has been conducted in less than 12-13 minutes.

(U) Internal areas of the Mach 12 aircraft protected by the active TPS will not present a problem following flight. The only significant internally stored heat is located in the thin insulation blanket behind the shingles which cools down considerably during descent.

(U) External surface temperatures at landing on the Mach 6 aircraft will be less severe than those experienced on the Mach 12 aircraft. Again, hot structure areas should be handled cautiously following a rapid descent profile. Mach 6 internal temperatures will produce a handling problem, since the aircraft uses a passive TPS concept. That is, internal structural temperatures will be near their maximum design value at landing with fuel tank walls near 250°F (394°K) and structure in non-fuel areas approaching 300°F (422°K). These areas will require handling precautions for extended periods following a landing. Engine inlet duct and engine cavity areas would be in excess of 300°F (422°K) after landing and would cool down even slower than moldline areas. Since rapid turnaround times are not a strong consideration in research aircraft, it would appear that waiting for the Mach 6

(U) FIGURE 7-6
MAXIMUM POST FLIGHT SURFACE TEMPERATURES
(Mach 12 Rocket Aircraft)



aircraft internal structure to cool down following a flight until it can be handled easily would be a logical safety precaution that would not impose an unrealistic restriction. An operational Mach 6 aircraft utilizing a passive TPS design could contain provisions for ram air cooling of the structure during descent to improve turnaround time requirements.

8. RESEARCH AREAS REQUIRING SPECIAL EMPHASIS

(U) This study has provided the characteristics of a number of attractive facilities capable of conducting the research required for future operational airbreathing hypersonic cruise aircraft. Evaluation of the research potential of each of these facilities is presented in Volume IV, Part 3.

(U) Attractive Mach 12 and Mach 6 research vehicles have been defined which will provide substantial verification of design and analytical techniques necessary for high confidence design of future operational hypersonic systems. A carefully planned flight research aircraft test program will enable demonstration of integrated configuration, structural, and subsystems concepts in a realistic aero/thermodynamic and operational environment. Such a program offers the potential to obtain a high confidence level in all flight systems and their component technologies. In addition, the operation of a test vehicle enables exploration of non-quantified factors such as maintenance, refurbishment, and system flexibility - all intangible without the benefit of vehicle operational experience.

(U) However, in such a program, part of the aircraft development would include a comprehensive program of ground testing. This research and development effort would be directed toward the specific system selected and would be conducted to provide substantiation of the design concepts selected for the system. This part of the flight research aircraft program is a significant element of the research contribution of the total program.

(U) Prior to receipt of the Authority to Proceed with such a program and selection of a final vehicle configuration there is much work that can even now be initiated in order to achieve a usable flight research aircraft in a timely manner. The various technological areas that should receive special emphasis for the two vehicles refined in Phase III are given in the following sections. The basis for this effort is, of course, the selected final configurations previously presented. However, much of the recommended research is generalized in nature and would contribute to a wide variety of research aircraft configurations as well as to operational systems.

8.1 MACH 12 RESEARCH AIRCRAFT

(U) Early acquisition capability of this aircraft is enhanced by incorporation of "off-the-shelf" rocket engines. Thus this is a particularly important area to initiate research and development. Areas which are especially important to the basic Mach 12 flight vehicle are:

- Configuration Development

  Performance, stability and control configuration refinement
  Separation studies
  Surface heating anomalies

- Structural/Thermal Protection Systems

  Radiative shingle
  Insulation systems
  Materials characterization

- Component and Subsystems Demonstration

  Cryogenic propellant systems
  Flight data acquisition systems

(U) Establishment of concept feasibility does not require specific solutions in these areas, since the basic technologies are relatively well understood. The major area of effort lies in substantiation of specific design/operational details in conjunction with subsystem/vehicle integration for the Mach 12 design. Substantiation and high confidence verification will come with actual vehicle experience during the logical speed buildup of the research aircraft, and minor modifications may be incorporated where a need is indicated by initial flight data.

(U) In addition to the basic Mach 12 airlaunched rocket vehicle, several research options are provided by vehicle modifications as discussed in detail in Section 4.2. For these, only the SJ/CSJ systems require special development emphasis. Candidate development items include inlet and nozzle integration, basic combustion technology (currently underway), regeneratively cooled component and fuel control system development. Specific requirements for SJ research are discussed in Volume IV, Part 3. Integration of the research options into the basic Mach 12 vehicle will of course require additional effort in each of the areas outlined below.

8.1.1 (U) CONFIGURATION DEVELOPMENT - The selected basic flight vehicle concept for Mach 12 research is an air launched, rocket powered, all-body configuration. The wide excursions in Mach number and altitude provide an inherent versatility for hypersonic research. To realize the full potential of the research vehicle, configuration refinement will be required during the engineering design and development phase to provide for the substantiation of performance and stability and control characteristics. Specific investigative areas are noted and their relationship to each research task is identified in Volume IV, Part 3.

Performance, Stability and Control Configuration Refinement - The aerodynamic data base used for the study incorporates empirical data from industry and Government high-speed wind tunnel facilities, coupled with vehicle design requirements from past studies. The result is an accepted high confidence level technique for prediction of aircraft drag and flight performance, as well as a representative evaluation of stability and control characteristics. As with any new aircraft design, a certain amount of wind tunnel verification and developmental effort is required to enable more exact determination of vehicle shape for the substantiation of performance and stability and control characteristics. Specific items which must be addressed include configuration shaping studies to increase L/D, reduce neutral point travel, and reduce trim drag; determination of component lift, drag, and stability contributions to the boost, cruise, descent, and landing phases; and definition of flow field interaction areas to allow better definition of pressure distribition, aerodynamic loads, and thermal profiles.

Separation Studies - Air drop of a rocket powered vehicle from a large subsonic aircraft has been a conventional operation with research aircraft for twenty years. Although no technological breakthrough is required, concentrated definition of vehicle flow field and flow field interactions is a necessity for the specific vehicle shapes involved. Past program results can provide high confidence design and operational guidance, and when added to a thorough wind tunnel separation test program, will provide the necessary analytical definition.

Aeroheating Anomalies - Current analytical methods used for prediction of vehicle temperature profiles will be complemented by thermal mapping tests. Both will be integrated with the configuration development work outlined earlier. Existing prediction techniques in interference regions (such as control surface intersections where shock/boundary layer interactions occur) are not fully definitive and require concentrated flow testing. Further complications are introduced by radiation interference between intersecting surfaces and control surface gaps. This interference does not permit the surfaces to cool by radiating freely to space. Wind tunnel testing is required to properly indicate the locations and magnitude of these high temperature regions as well as to provide a thermal map for the specific research vehicle configuration. Test objectives include an iterative evaluation of changes in vehicle shape with regard to configuration performance to obtain an optimum compromise between aerodynamic and thermodynamic considerations.

8.1.2 (U) STRUCTURAL/THERMAL PROTECTION SYSTEMS - The structural/thermal protection systems chosen for the Mach 12 aircraft represent high confidence, near state-of-the-art concepts. Because flight into the thermal environment associated with Mach 12 can result in some local equilibrium skin temperatures near stagnation values (as high as 3000°F, 1923°K) it is necessary to utilize advanced concepts for radiative shingles and active cooling of the primary structure. Cool primary structure is maintained by means of a thermal protection system which consists of coated external shingles, insulation, air gap, and an active water wick cooling system. The feasibility of this concept has been established by laboratory and flight tests of various system components and a structural element representative of the complete system.

(U) Subsequent improvements in composite materials technology could be incorportated into the research vehicle, and performance of the advanced composite materials verified as these materials attain hardware status, i.e., satisfactory demonstration of manufacturing, fabrication, and inspection techniques. If advanced composite structures were incorporated in all or a portion of the primary structure, they would require special emphasis in development.

Radiative Shingles - Protected primary structure was selected for the Mach 12 vehicle concept because of the pronounced effect on overall structural weight efficiency. The radiation shingles are designed to provide a smooth external moldline with adequate strength and stiffness at all flight temperatures to prevent panel flutter or excessive deflection under design air loads. A major consideration in shingle design is the large temperature gradient imposed during boost, which has the potential to warp the shingle and open gaps allowing hot boundary layer air to leak into the interior. Local burnthroughs could be catastrophic. As a result, focused development activity should be concentrated on thermal stress relief within the panel and assessment of heat shorts through attachments. Initial steps to verify the shingle concept in a simulated operational environment may be conducted on a laboratory scale (two or three panels). Evaluations should include investigation of panel strength and deformation characteristics and the effects of thermal expansion on panel interaction and surface gaps. The intent is to verify the structural concept when exposed to representative airloads and heating cycles. Additional effort should be directed toward an assessment of fabrication and inspection techniques including an evaluation of lifetime and maintainability requirements. Definition of techniques for fabrication, joining, and mounting to primary structure requires relatively large scale systems to evaluate interface parameters; but this can be accomplished most expediently by fabrication of a typical fuselage section.

Insulation System - The Mach 12 vehicle concept utilizes insulation and a saturated water wick blanket as basic insulating elements of the thermal protection concept. This enables protection of the substructure to temperatures corresponding to the boiling point of water at the local ambient pressure. Concept feasibility has been demonstrated through numerous laboratory experiments at MCAIR and in flight on advanced reentry vehicles. These tests have allowed evaluation of wicking materials, methods of wick containment, operating efficiencies, and water retention under both static and acceleration conditions. It is recognized that continued laboratory development activity is desirable to fully explore the operational effects and potential recycle/refurbishment requirements associated with numerous thermal cycles. Additional techniques may be investigated to integrate the concept with the entire thermal protection system, support system heat shorts, and to further reduce wicking thickness, hence, total system weight.

Materials Characterization - The selected Mach 12 concept is designed on the basis of existing structural/material concepts representative of a near term development status. An integral part of the structural/thermal protection system is the use of coated refractory alloys necessary to provide oxidation resistance when approaching 3000°F (1923°K). Current coating investigations have proven the feasibility of the concept on a laboratory scale, but have been rather limited in scope as to configuration, duty cycles, and complete environmental simulation. Design allowables and performance characteristics must be established by test before efficient design of reliable, reusable coated columbium alloy structures can be assured for the concept. The performance of a protective coating on a columbium or tantalum alloy structure is strongly influenced by the design of that structure and by the conditions of its service environment. An orderly test program designed to simulate the anticipated hypersonic environment will provide the necessary design confidence level to permit the incorporation of reusable coated radiative shingles into the Mach 12 flight research facility. In conjunction with an evaluation of coatings, the bare metal must also be characterized at temperatures approaching 3000°F (1923°K). The testing can be accomplished on a relatively small (laboratory) scale leading to material selection and comparative evaluation with coated specimens.

(U) Although not specifically required to provide structural integrity for the Mach 12 vehicle, advanced composite materials, both resin matrix and metal matrix, can provide significant performance improvements through increased strength/weight ratios. A similar development effort can be applied as highlighted above for the coated refractory shingles. Verification may be attained later in the flight program by replacement of conventional structural components on the Mach 12 research vehicle with components fabricated from advanced composite materials.

(U) A number of recent studies have been conducted of other attractive thermal protection system concepts. Work as described above as well as alternate systems would be extremely valuable, particularly in consideration of the high research value placed on development of thermal protection systems for operational aircraft.

8.1.3 (U) COMPONENT AND SUBSYSTEMS DEMONSTRATION - Major subsystems and components incorporated into the research vehicle design represent demonstrated hardware concepts (particularly in the case of the cryogenic propellant systems), although demonstration may have been performed in a different operating mode and thermal environment. The initial effort during the engineering design and development phase should be directed toward demonstration of relatively long life, reusable concepts in an operational environment representative of sustained Mach 12 flight.

Cryogenic Propellant Systems - Current analytical modeling for cryogenic tank pressurization is based on vertically positioned tankage with continuous high outflow rates and applied thrust loads. Experimental coefficients must be determined for analysis of heat and mass transfer with horizontal tank orientation, relatively low flow rates and aircraft dynamic environment. An integral part of the aircraft cryogenic requirements includes demonstration of reusable integral tank concepts in concert with demonstration of the structural/thermal protection concepts discussed earlier.

(U) One of the principal elements in the tankage system which requires concentrated emphasis is the reusable internal insulation/vapor barrier system. Loss of insulation integrity can result through leaks in the vapor barrier. If leaks develop cold gas will be trapped between the vapor barrier and tank wall. After the tank is empty, these cold gases will heat up and expand, potentially separating the vapor barrier from the insulation. Additionally, leaks in the vapor barrier can degrade the insulation performance to a point where ambient air will condense on the exterior of the tank. These problems are being actively investigated by numerous investigators.

Data Acquisition - Data acquisition plays a major role in any research/development program. For the Mach 12 vehicle, sensors must be capable of operating at or near temperatures of 3000°F (1923°K). Current capability under these conditions is limited to short duration boost glide vehicles or ground applications. Specific tasks which must be performed include investigations of methods to obtain boundary layer heat transfer and transition data as well as methods to provide dynamic pressure, angle-of-attack, and yaw data to both the pilot and a telemetry/on-board flight recorder.

8.2 MACH 6 RESEARCH AIRCRAFT

(U) Areas which are especially important to the basic flight vehicle are:

o Configuration Development

Performance, stability and control configuration refinement
Surface heating anomalies

- o Structural/Thermal Protection Systems

  Radiative shingle
  Regeneratively cooled panels

- o Propulsion integration

  Inlet integration
  Ramjet engine development

- o Component and subsystems demonstration

  Cryogenic propellant systems
  Flight data acquisition systems.

(U) Establishment of concept feasibility does not require specific solutions in these areas, since the basic technologies are relatively well understood. The major area of effort lies in substantiation of specific design and operational details in conjunction with subsystem/vehicle integration. Substantiation and high confidence verification will come with actual vehicle experience during the logical speed buildup of the research aircraft, and minor modifications may be incorporated where a need is indicated by initial flight data.

8.2.1 (U) CONFIGURATION DEVELOPMENT - The selected basic flight vehicle concept for Mach 6 research is a ground takeoff, turboramjet powered, wing/body configuration. Excursions in Mach number and altitude provide an inherent versatility for research in the high supersonic and low hypersonic flight regime. The full potential of the research vehicle can only be achieved by configuration refinement and performance substantiation during the engineering design and development phase. Specific investigative areas are noted and their relationship to the research tasks is identified in Volume IV, Part 3.

Performance, Stability and Control Configuration Refinement - The aerodynamic data base used for the study incorporates empirical data from industry and Government high speed wind tunnel facilities, coupled with vehicle design requirements from past studies. The result is an accepted high confidence level technique for prediction of aircraft drag and flight performance, as well as a representative evaluation of stability and control. As with any new aircraft design, a certain amount of wind tunnel verification and developmental effort is required to enable more exact determination of vehicle shape for performance substantiation and identification of stability and control characteristics. Specific items which must be addressed include configuration shaping studies to reduce both neutral point travel and zero lift pitching moments and thereby reduce trim drag at high speed; determination of component lift, drag, and stability contributions in all flight phases; and definition of flow field interaction areas to allow better definition of pressure distribution, aerodynamic loads, and thermal profiles.

Aeroheating Anomalies - Current analytical methods predict vehicle temperature with varying degrees of accuracy and complement the thermal mapping studies related to the configuration development work outlined earlier. However, existing prediction techniques in interference regions (such as wing/body intersections or in the engine inlet where shock/boundary layer interactions occur) are not fully definitive and require concentrated flow testing. Further complications are introduced by radiation interference between intersecting surfaces and control surface gaps. This interference does not permit the surfaces to cool by radiating freely to space. Wind tunnel testing is required to properly indicate the locations and magnitude of these high temperature regions as well as to provide a thermal map for the specific research vehicle configuration. Test objectives include an interactive evaluation of changes in vehicle shape with regard to configuration performance to obtain an optimum compromise between aerodynamic and thermodynamic considerations.

8.2.2 (U) STRUCTURAL/THERMAL PROTECTION SYSTEMS - The structural/thermal protection systems chosen for the basic vehicle represent minimum risk, near state-of-the-art concepts. Because flight into the thermal environment associated with Mach 6 can result in some local equilibrium skin temperatures near stagnation (2500°F, 1645°K) values, incorporation of advanced concepts for radiative shingles and active cooling of the engine inlet structure may offer a high degree of flexibility for the vehicle concept. Primary structure is maintained at design operating temperature by means of a thermal protection system which consists of external shingles and passive insulation. The feasibility of this concept has been established by laboratory and flight tests of various system components and a structural element representative of the complete system. The potential exists to restructure sections of the flight vehicle with advanced composite structure to verify the level of flight performance as confidence increases in the utilization of composite materials.

Radiative Shingles - Protected primary structure was selected for the basic vehicle concept because of the pronounced effect on overall structural weight efficiency. The radiation shingles, primarily Rene' 41, are designed to provide a smooth external moldline with adequate strength and stiffness at all flight temperatures to prevent panel flutter or excessive deflection under design air loads. Focused development activity should be concentrated on thermal stress relief within the panel and assessment of heat shorts through attachments. Initial steps to verify the shingle concept in a simulated operational environment may be conducted on a laboratory scale (two or three panels). Evaluations should include investigation of panel strength and deformation characteristics and the effects of thermal expansion on panel interaction and surface gaps. The intent is to verify the structural concept when exposed to representative airloads and heating cycles. Additional effort should be directed toward an assessment of fabrication and inspection techniques for the superalloy, including an evaluation of lifetime and maintainability requirements. Definition of techniques for fabrication, joining, and mounting to primary structure requires relatively large scale systems to evaluate interface parameters; but this can be accomplished most expediently by fabrication of a typical fuselage section.

Regeneratively Cooled Panels - Regenerative panels are used for internal inlet surface cooling (in areas where radiation cooling is not practical) during operation of the ramjet. These panels must possess a high degree of reliability to protect the primary structure from temperatures which approach the stagnation environment. Recently, two governmental contracts have been performed by industry with the goals of fabrication and test of LH_2 regeneratively cooled panels. The contracted efforts, in conjunction with related MCAIR studies, have provided for probable solutions. It is apparent from regenerative panel modes of failure (loss of bonding between vertical webs and face sheets) that fabrication techniques require additional investigation. Two promising methods, which should provide superior bonding over conventional furnace brazing, are salt bath brazing and diffusion bonding. Salt bath brazing provides very accurate and uniform temperature control, thereby eliminating the cold spot and warpage problems which are encountered in furnace brazing processes. The present limit on salt bath brazing is one of available salt bath size, but larger baths can economically be achieved. Extensive diffusion bonding investigations illustrate that structural integrity of the bond is excellent, since there is a diffusion of the parent material which alleviates the need for a braze or welding alloy. High temperature heat exchanger panels will utilize superalloys, such as Rene' 41, Inconel, TD Nickel, or Hastelloy. As a result, substantial fabrication and process development work is required to assure the structural integrity of the superalloy joint. In conjunction with the fabrication development effort, flow testing must be performed under typical flight heat loads and panel flow rates. Factors requiring investigation include flow distribution and control, warpage effects, and useful operating life. Graphite radiant heaters have recently been developed which operate in an inert atmosphere and are capable of providing heat fluxes in excess of 300 Btu/ft^2-sec (340 W/cm^2) for extended periods of time. These heaters will permit close duplication of environmental heat loads.

8.2.3 (U) <u>PROPULSION INTEGRATION</u> - Two aspects of the Mach 6 airbreathing engine development which require emphasis are the mixed compression inlet and integration of the wraparound ramburner with the J-58 gas generator core.

<u>Inlet</u> - There has been only modest operational use of mixed compression inlets on supersonic cruise aircraft. For example, the XB-70 incorporates a two-dimensional, variable internal ramp geometry designed for nominal Mach 3 operation, while other aircraft have successfully employed a translating spike. The inlet operational and control problems encountered have been identified and, in many cases, solutions effected. Inlet unstarts were common in early flight test and inlet control requirements have been identified as a primary element in the successful usage of mixed compression inlets. Operation of the vehicle with a mixed compression inlet in an environment 2000°F (1100°K) hotter than current operational weapon systems will require solutions of these same problems in addition to satisfaction of high temperature seal and actuator requirements. The pressure recovery characteristics used for the inlet were based on a compilation of experimental data. Demonstration of a mixed compression inlet designed to operate at flight speeds from takeoff to Mach 6.0 to verify the performance levels characterized by these data is a valid research objective.

<u>Ramjet</u> - The ramjet portion of the turboramjet engine for the basic Mach 6 vehicle is a new system, designated the STRJ11A-27. The engine, considered specifically for this vehicle program, is suitable for flight test to Mach 7 above 110,000 ft (33.5 km) and utilizes the JT11D (J-58) as the gas generator core. Development of the ramjet mode will be accomplished in existing ground facilities and is based on demonstrated technology. In order to maintain the high confidence approach recommended for the basic program, Pratt and Whitney Aircraft has recommended a 40 month engine development program resulting in an engine configuration substantiated for

initial flight test operation to Mach 4, including demonstration of transition to the ramjet mode. Structural integrity and durability of the ramjet will be verified throughout the flight regime. Vitiated air is considered for structural tests into the speed regime above Mach 4. Engine performance (thrust and SFC) can be demonstrated in ground tests with PFRT to Mach 4, followed by flight test verification for speeds above Mach 4. This flight test can be accomplished with relatively high confidence as a result of the demonstration engine structure through a simulated aerothermal environment.

8.2.4 (U) COMPONENT AND SUBSYSTEMS DEMONSTRATION - Major subsystems and components incorporated into the vehicle design represent demonstrated hardware concepts (particularly in the case of the cryogenic propellant systems) although demonstration may have been performed in a different operating mode and thermal environment. The initial effort during the engineering design and development phase should be directed toward demonstration of relatively long life, reusable concepts in an operational environment representative of sustained Mach 6 flight.

Cryogenic Propellant Systems - Current analytical modeling for cryogenic tank pressurization is based on vertically positioned tankage with continuous high outflow rates and applied thrust loads. Experimental coefficients must be determined for analysis of heat and mass transfer with horizontal tank orientation, relatively low flow rates and aircraft dynamic environment. An integral part of the aircraft cryogenic requirements includes demonstration of reusable integral tank concepts in concert with demonstration of the structural/thermal protection concepts discussed earlier, and reusable internal cryogenic insulation systems.

Data Acquisition - Data acquisition plays a major role in any research/development program. For the Mach 6 vehicle, sensors must be capable of operating at or near temperatures of 2500°F (1645°K). Current capability under these conditions is limited to short duration boost glide vehicles or ground applications. Specific tasks which must be performed include investigations of methods to obtain boundary layer heat transfer and transition data as well as methods to provide dynamic pressure, angle-of-attack, and yaw data to both the pilot and a telemetry/on-board flight recorder.

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9. OBSERVATIONS AND CONCLUSIONS

(U) A significant amount of research must be accomplished in order to proceed confidently with the development of future (1980-2000) operational high speed aircraft systems. This is particularly true in the areas of propulsion, propulsion system-airframe integration, structural materials, thermal protection, structural design concepts, and fabrication and refurbishment techniques. Much of the knowledge needed can only be acquired through flight experience.

(U) Historically, two alternate avenues of approach have been pursued in accomplishing flight research. One approach is to accomplish only that research necessary to develop a particular operational system through prototype testing within the development program. This is a direct approach. The test results obtained have specific and immediate applicability. However, they are often so narrowly defined that the data base cannot be extrapolated with confidence to other potential systems. Hence, in practice, the development of each system requires a prototype and the associated research testing. There are overlapping areas of interest and, in general, this approach, though direct, is inefficient from an overall point of view. It is also very expensive.

(U) A second approach is to expand the technology base as a whole by undertaking a broad flight research program well in advance of initiating the development of potential operational systems. This route is not so direct. Such a program must be geared to the development needs of all foreseeable future systems. It requires careful planning and timely execution. The research results obtained will be only as valuable as the anticipation of requirements and the forethought employed in formulating the program. But when properly carried to completion, this approach will yield results that are applicable to the development of not just one, but several advanced systems. Overall, it represents the most fiscally responsible means of providing the research needed.

(U) The initial planning required to successfully undertake the latter approach is needed now. The HYFAC Study results provide the necessary foundation.

(U) The flight research facilities studied during Phase III and described in the foregoing sections of this report have wide applicability in meeting the development needs foreseen for future operational aircraft. It has been shown that a broad spectrum of research objectives can be satisfied by utilizing the concept of two basic flight research vehicles, each designed with sufficient flexibility to accommodate a variety of research options. This is illustrated in Figure 9-1 in comparison with the nine previously defined potential operational aircraft systems and is summarized in detail in Volume IV, Part 3 of this report.

(U) It has also been shown that it is feasible to include provisions within the basic vehicle design to accommodate the future testing of these several advance research options. This serves to further enhance the inherent research capability of each vehicle. These designs can be initiated with confidence using today's technology base.

(U) FIGURE 9–1
FLIGHT RESEARCH FACILITIES SATISFY FUTURE AIRCRAFT DEVELOPMENT NEEDS

- High Capability
- Moderate Capability
- Low Capability
- NA Not Applicable

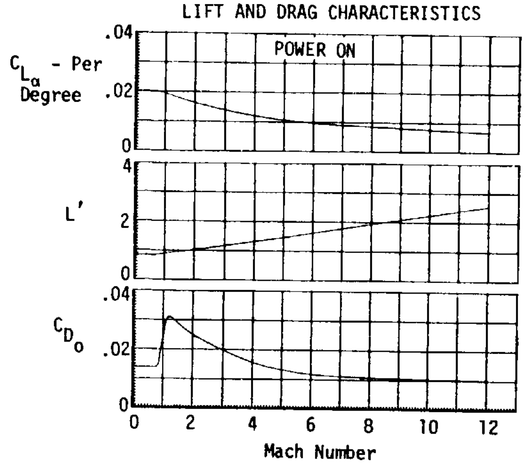
Potential Operational Hypersonic Aircraft Systems			HYFAC Phase III Flight Research Facilities												
			Mach 6 Research Aircraft				Mach 12 Research Aircraft								
Description	Prop. Sys.	CODE	Basic	Arm Option	TPS Option	Adv. TRJ	Basic	HTO Option	VTO Option	CSJ Option	SJ Option	Arm Option	TPS Option	STG Option	TJ Option
Reusable Launch Vehicle	TRJ	L1	High	NA	High	High	Low	Low	Low	Moderate	Moderate	NA	High	High	Moderate
Reusable Launch Vehicle	TJ + CSJ	L2	Moderate	NA	Moderate	Moderate	Moderate	Moderate	Moderate	High	High	NA	High	High	Moderate
Reusable Launch Vehicle	Rkt	L3	Low	NA	Moderate	Low	High	High	High	NA	NA	NA	High	High	High
Reusable Launch Vehicle	Rkt + SJ	L4	Moderate	NA	Moderate	Moderate	High	High	High	High	High	NA	High	High	High
Hypersonic Transport	TRJ	C1	High	NA	High	High	Low	Low	NA	Moderate	Low	NA	High	NA	Moderate
Hypersonic Transport	TJ + SJ	C2	Moderate	NA	Moderate	Moderate	Moderate	Moderate	NA	High	High	NA	High	NA	High
Military – AMI	TRJ	M1	High	High	High	High	Low	Low	NA	NA	NA	High	High	NA	Moderate
Military – Strike	Rkt + SJ	M2	Moderate	Moderate	Moderate	Moderate	High	High	NA	High	High	High	High	NA	High
Military – Interceptor	Rkt + SJ	M3	Moderate	Moderate	Moderate	Moderate	High	High	NA	High	High	High	High	NA	High

(U) Summaries of the aerodynamic and propulsion characteristics, weights, costs, and performance for each basic vehicle are presented in Figures 9-2 and 9-3. The basic power plants suggested are modifications of presently existing engines. The performance they provide is significantly greater than any known proposed near-term aircraft system. The costs are moderate in comparison with the scope of the undertaking. The eventual savings in development costs will be substantial.

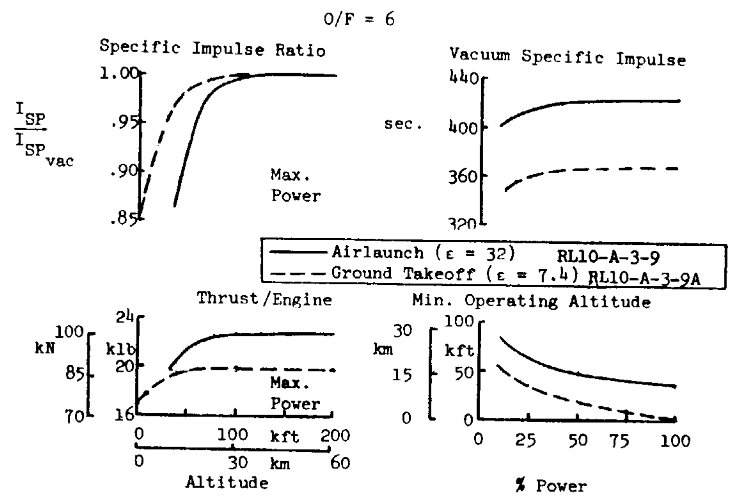
(U) The key factors in realizing these benefits in the development of future (1980-2000) operational hypersonic aircraft are expediency in initiating such a research program and flexibility in the design of the research aircraft. The latter must provide not only for a broad research capability, but for versatility in adapting to currently unforeseen research requirements, as well.

(U) FIGURE 9-2
MACH 12 ROCKET VEHICLE SYSTEM SUMMARY

LIFT AND DRAG CHARACTERISTICS



ROCKET ENGINE PERFORMANCE



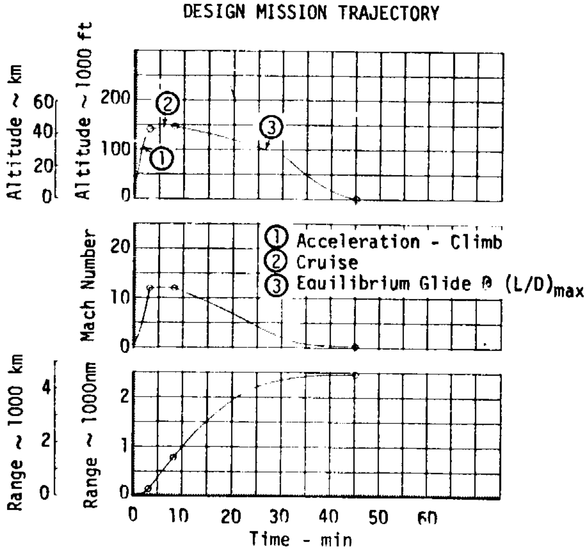
WEIGHT SUMMARY

	lbm	kg
Structure	13,958	6,331
Equipment	6,242	2,829
Propulsion	3,140	1,425
Propellant	56,310	25,544
OWE	23,340	10,585
TOGW	79,650	36,129

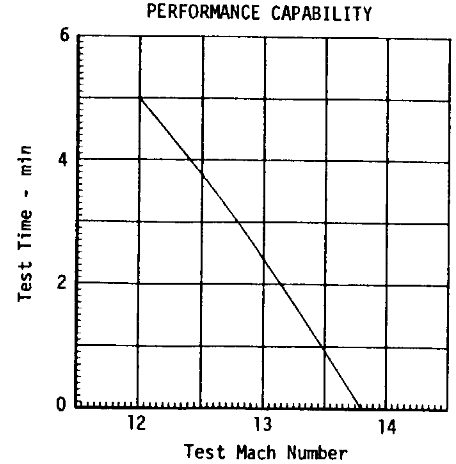
COST SUMMARY
(3 VEHICLES)
MILLIONS OF 1970 DOLLARS

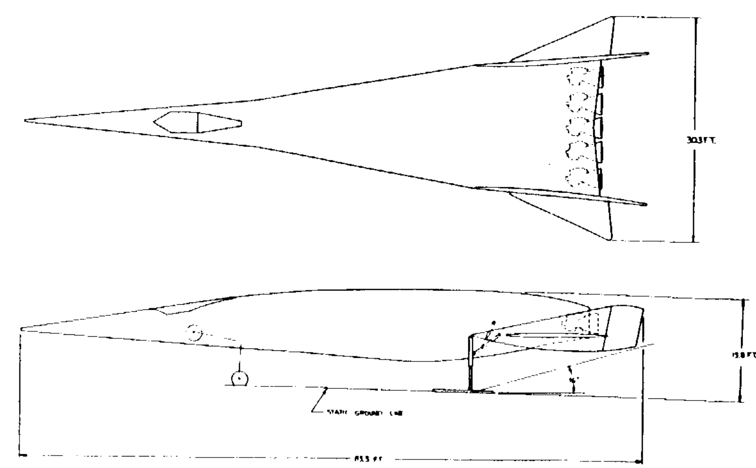
RDT&E	126
Investment	137
Operating	88
Total	351

DESIGN MISSION TRAJECTORY

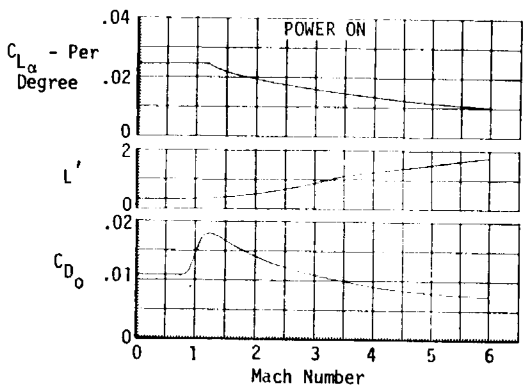


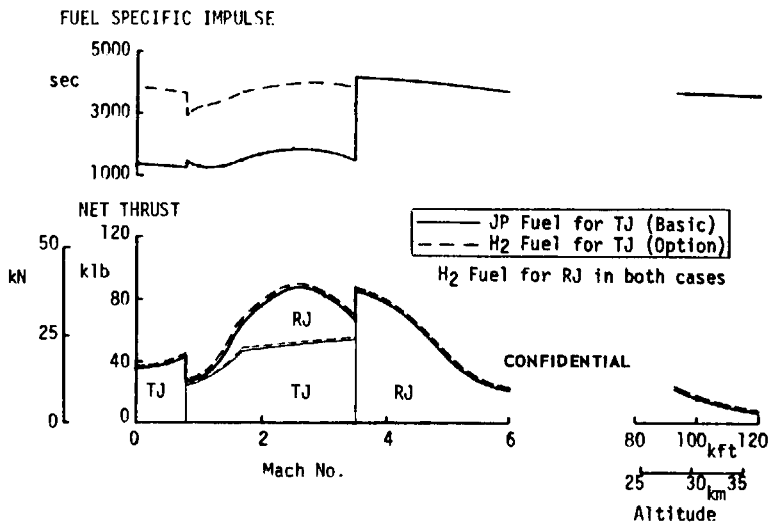
PERFORMANCE CAPABILITY





(C) FIGURE 9-3
MACH 6 TURBORAMJET VEHICLE SYSTEM SUMMARY





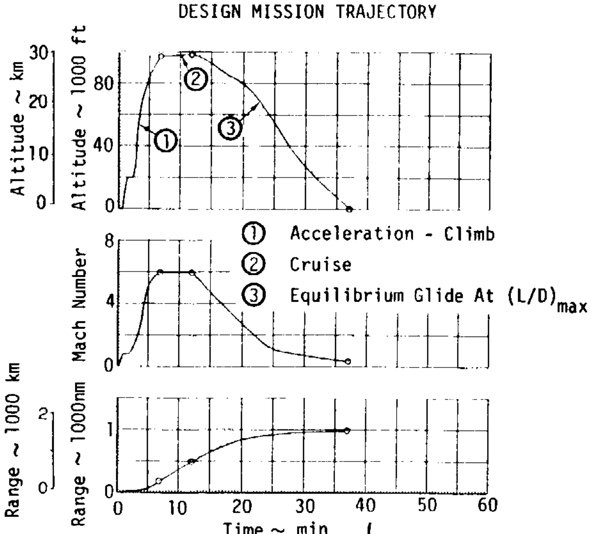
CONFIDENTIAL

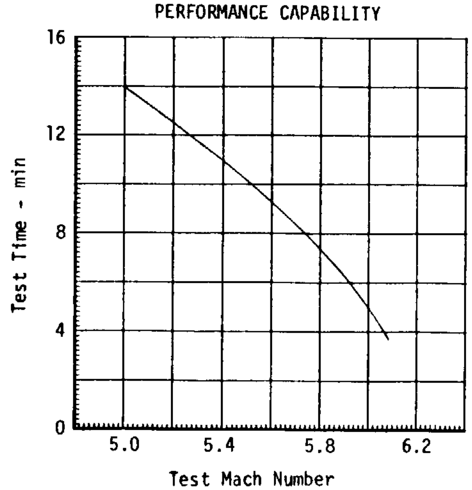
WEIGHT SUMMARY

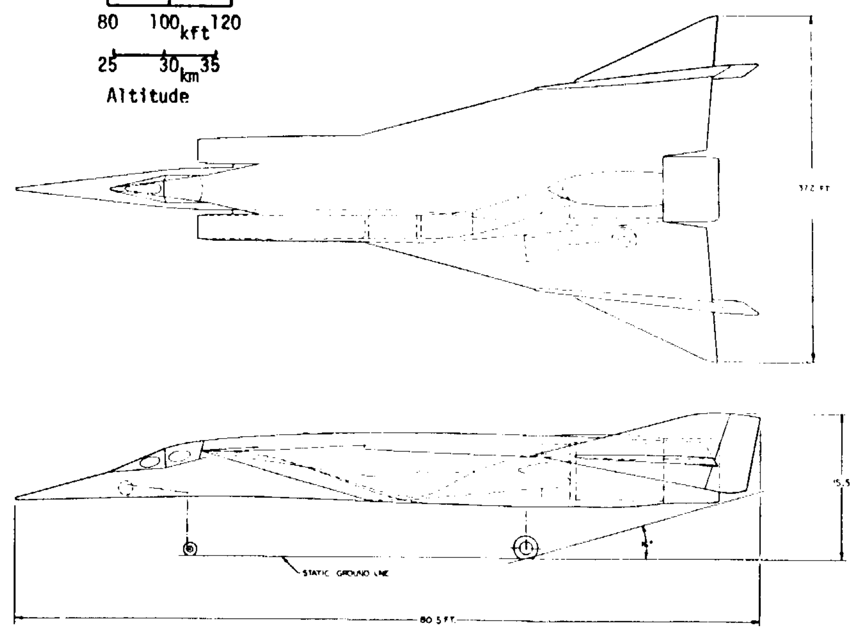
	lbm	kg
Structure	21,033	9,539
Equipment	5,802	2,630
Propulsion	21,621	9,807
Propellant	12,970	5,883
OWE	48,456	21,976
TOGW	61,426	27,859

COST SUMMARY
(3 VEHICLES)
MILLIONS OF 1970 DOLLARS

RDT&E	212
Investment	186
Operating	92
Total	490







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