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Contract AF 08(635)-1168

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PYE WACKET
Feasibility Test Vehicle Study
(Summary)

(TITLE UNCLASSIFIED)

VOLUME I

Prepared by

Convair/Pomona
General Dynamics Corporation
Pomona, California

for

AIR PROVING GROUND CENTER

EGLIN AIR FORCE BASE, FLORIDA

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15 February 1961

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Convair Division of General Dynamics Corporation
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Prepared for

Detachment 4
HEADQUARTERS, AERONAUTICAL SYSTEMS DIVISION
Air Force Systems Command
United States Air Force
Eglin Air Force Base, Florida

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FOREWORD

This report was prepared under Air Force Contract Number AF 08(635)-1168, Project 3811, (U) "Lenticular Rockets." The work was administered initially under the direction of the Directorate of Development, APGC, and completed under the guidance of Detachment 4, Hq Aeronautical Systems Division at Eglin Air Force Ba e, Florida

This document, except the title, is classified SECRET in accordance with AFR 205-1, paragraph 10b, because of the nature and potential military application of the research work and data described herein, and NOT RELEASABLE TO FOREIGN NATIONALS in accordance with AFDCMI 56.

ABSTRACT

Feasibility studies were conducted of a circular planform, modified lenticular cross section vehicle. The results of these studies form the basis for the ultimate fabrication and flight test of vehicles to prove the omnidirectional launch, stability and control, and maneuver capability of the basic concept.

This summary report includes a short résumé of each major task studied under the auspices of contract AF 08(635)-1168. The basic aerodynamics, the reaction control system and the autopilot are discussed relative to the forward, crosswind and aft launch flight conditions. Also summarized are results of the missile structural analyses. All of the studies were based on the severe sea level flight environment. Brief sections are devoted to prototype weapon considerations and to recommendations for future effort.

The complete task is reported in three volumes: Volume I -- Summary, Volume II -- Aerodynamics, and Volume III -- Configuration and Autopilot/Control.

CONTENTS

Section 1.0
INTRODUCTION

During this contract (AF 08(635)-1168) a detailed aerodynamic evaluation was conducted, a structural design established and an autopilot and control system studied for a Feasibility Test Vehicle of the circular planform, blunted lenticular cross section configuration. It was the intent throughout this work that a missile design be established which could be subsequently fabricated for flight tests from a high-speed rocket-sled to prove the feasibility of this configuration as an advanced airborne weapon.

The circular planform, lenticular cross section concept originated with the Technical Planning Group, formerly of the Directorate of Development of the Air Proving Ground Center, now of Detachment 4 of Wright Air Development Division, Target and Armament Development Directorate, Eglin Air Force Base, Florida. This effort was further advanced by an experimental program conducted in Tunnel E-1 of the Gas Dynamics Facility, Arnold Engineering Development Center. Convair/Pomona continued this developmental work under the auspices of a six-month contract entitled "Lenticular Rocket", AF 08 (635)-542, awarded in June 1959. During this Phase I study contract, a general aerodynamic evaluation and vehicle feasibility design study were conducted with emphasis placed on the determination of those characteristics pertaining to the stability, control and maneuverability of this configuration. A further objective of the latter study was the establishment of the design feasi-

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bility of the configuration as a potential airborne weapon. The unclassified code name "PYE WACKET" was assigned to the study.

As a result of the Phase I study, the symmetrical lenticular cross sectional configuration was modified. The improved version, designated Model III, has the maximum thickness located at the extreme aft end. The major aerodynamic advantage exhibited by Model III over the basic symmetrical lenticular configuration is a rearward shift of the aerodynamic center of pressure. This rearward shift of the center of pressure greatly simplifies the problem of controlling the missile in flight. Another advantage of the unsymmetrical shape is the supersonic drag reduction associated with the blunt trailing edge. This resulted in a higher lift-to-drag ratio and in turn an increased range and maximum velocity. The Phase I work is reported in the "PYE WACKET Feasibility Study, Technical Summary Report".*

The Phase II contract (AF 08(635)-1168) was negotiated for the logical continuation of the feasibility studies initiated during the Phase I contract. These studies comprised wind-tunnel tests, analyses, and designs required for the subsequent fabrication of a flight test or Feasibility Test Vehicle. Per contract specifications, the continued investigations and analyses to establish the feasibility of the Lenticular Rocket were directed toward a test vehicle which could be fabricated in a short time from off-the-shelf items. Wherever possible, these investigations and analyses considered improved techniques and

*PYE WACKET Lenticular Rocket Feasibility Study, APGC-TR-60-25, May 1960, SECRET methods applicable to prototype vehicles of the ultimate performance potential of an advanced airborne weapon.

The Phase I aerodynamic evaluation was expanded during this study by an extensive wind-tunnel program with emphasis directed toward determining the omnidirectional launch characteristics of the vehicle. Two wind-tunnel models were employed, one instrumented for force measurements and one instrumented for pressure measurements. The latter model also incorporated provisions for simulating the power-on condition of the four roll-pitch control motors. The pressure readings were used to check the force model data and also to determine the effects of the shock pattern produced by the interaction of the control jet exhaust and the existing aerodynamic boundary layer. Owing to fund and time limitations, it was not possible to obtain force or pressure data on the models with simulation of the main motor power-on conditions, although it was recognized at the time that this information would be eventually required.

The missile structural design was developed to withstand the estimated handling, launch and flight loads. Component layouts were established such that the vehicle center-of-gravity at launch is located at the 43% chord (measured from the leading edge), thus alleviating some of the development problems. Preliminary aeroelastic studies were conducted on the structure to ensure design adequacy.

The reaction control motors and the associated components were studied extensively with special attention directed toward obtaining the high thrust level within the required response time. Several systems have been developed throughout the industry; however further effort is needed to either scale the thrust levels up to the required level and/or reduce the response time to that required for the PYE WACKET Feasibility Test Vehicle. The results of an industry survey and of a company-sponsored program in the reaction control motor field are encouraging.

The aerodynamic test data and analyses, the missile structural characteristics, and the control motor studies were all utilized in a detailed study of the stabilization and control of the missile. The resulting autopilot must not only stabilize and control the vehicle, but it must function to do so in a manner which fully exploits the unique properties of this configuration. The necessity for developing a nonlinear control philosophy to ease the control motor development problem became evident early in the program. At the same time, the aerodynamic moments appeared to be of such magnitude that, in order to minimize the thrust level of each motor, at least four control motors would be required to produce the pitch moments (two positive and two negative) and at least four for the roll moments (two clockwise and two counterclockwise). This immediately dictated the necessity of electronically combining the pitch and roll signals to drive four control motors in a time sharing manner. An optimum method of effectively utilizing the four common control motors was developed and a complete evaluation of the resulting cross coupling terms was conducted. From a study of the omnidirectional launch characteristics of PYE WACKET, it became axiomatic that the assumptions made for a study of a standard ogive-cylinder missile would not be applicable to this vehicle. Because of the system complexities and the nonlinear control philosophy, separate autopilot studies were conducted for each plane of control, i.e., pitch, roll and yaw. The next step combined the pitch and roll control systems. These studies then culminated in simulated missile flights on a time varying, parametrically descriptive, complete three-dimensional simulation. The missile satisfactorily followed each prescribed trajectory.

Section 2.0
AERODYNAMICS

The PYE WACKET Feasibility Test Vehicle is a circular planform, blunted lenticular cross-section configuration (Figure 2.1), 60 inches in diameter with a 21% thickness-to-chord ratio. The maximum thickness of 12.6 inches occurs at the blunted trailing edge.

The aerodynamic data were obtained during 80 hours of wind tunnel testing at Arnold Engineering Development Center, Tullahoma, Tennessee. Because of the large amount of required instrumentation, two 1/3-scale models were utilized during the tests. The two models are geometrically similar; one was instrumented to measure forces and moments and the other to measure the body pressure distributions. The pressure model contained provisions for simulating the pitch and roll reaction control jets with cold gas at chamber pressures ranging from 400 to 1000 psia. The models were tested at Mach numbers ranging from 0.6 to 5.0, at angles of attack from 0 to 15 degrees and at sideslip angles from 0 to 180 degrees.

The normal force coefficients per degree angle of attack as a function of Mach number are presented for the forward, crosswind and aft launch positions. The normal force derivative (Figure 2.2) for the forward ($\beta = 0^{\circ}$) and crosswind ($\beta = 90^{\circ}$) launch positions are applicable up to a 4-degree angle of attack. However, the normal force derivative for the aft launch, $\beta = 180^{\circ}$, is extremely nonlinear with respect to angle of attack at subsonic Mach numbers. Therefore, the normal force

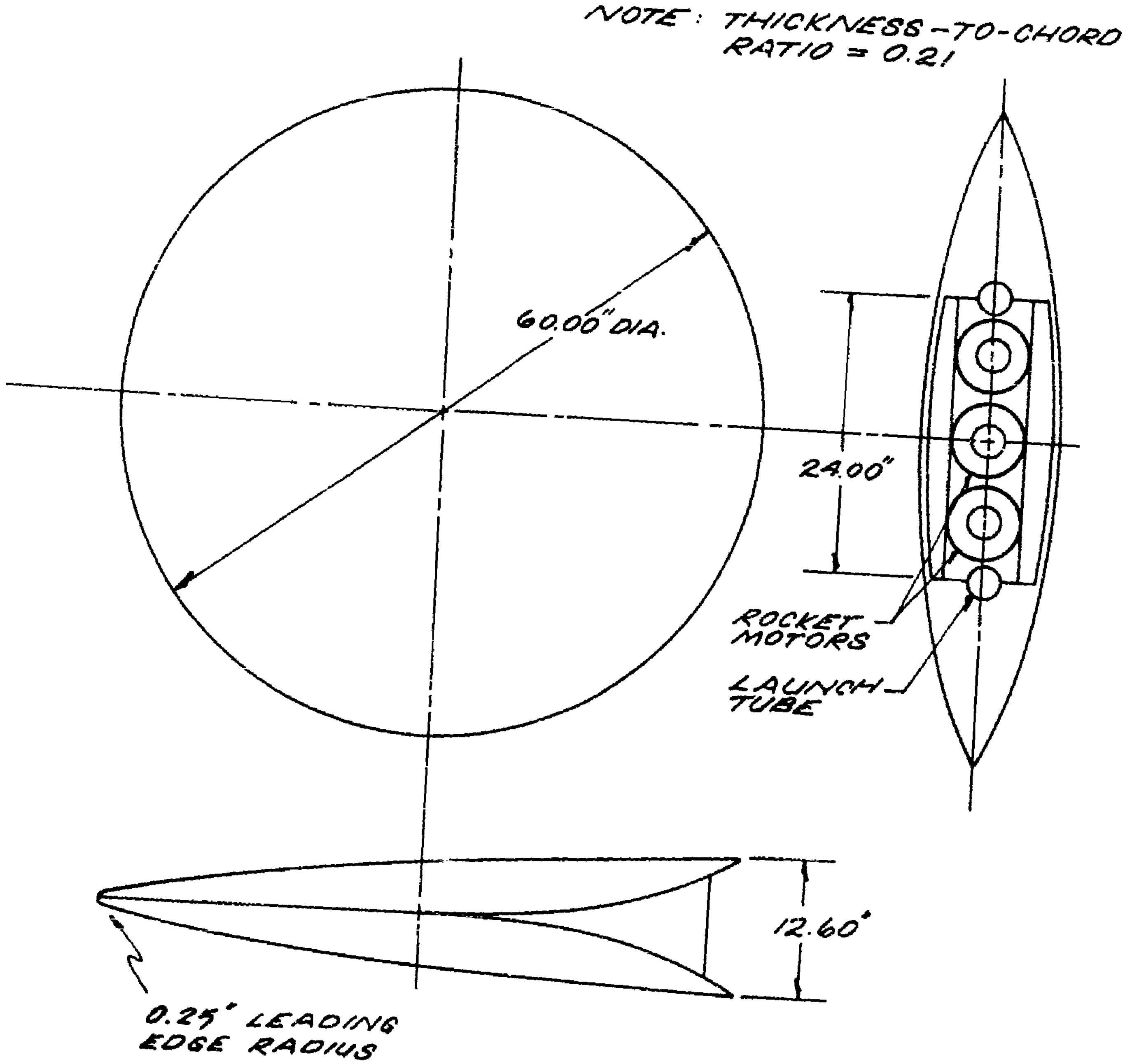


Figure 2.1 Feasibility Test Vehicle Configuration

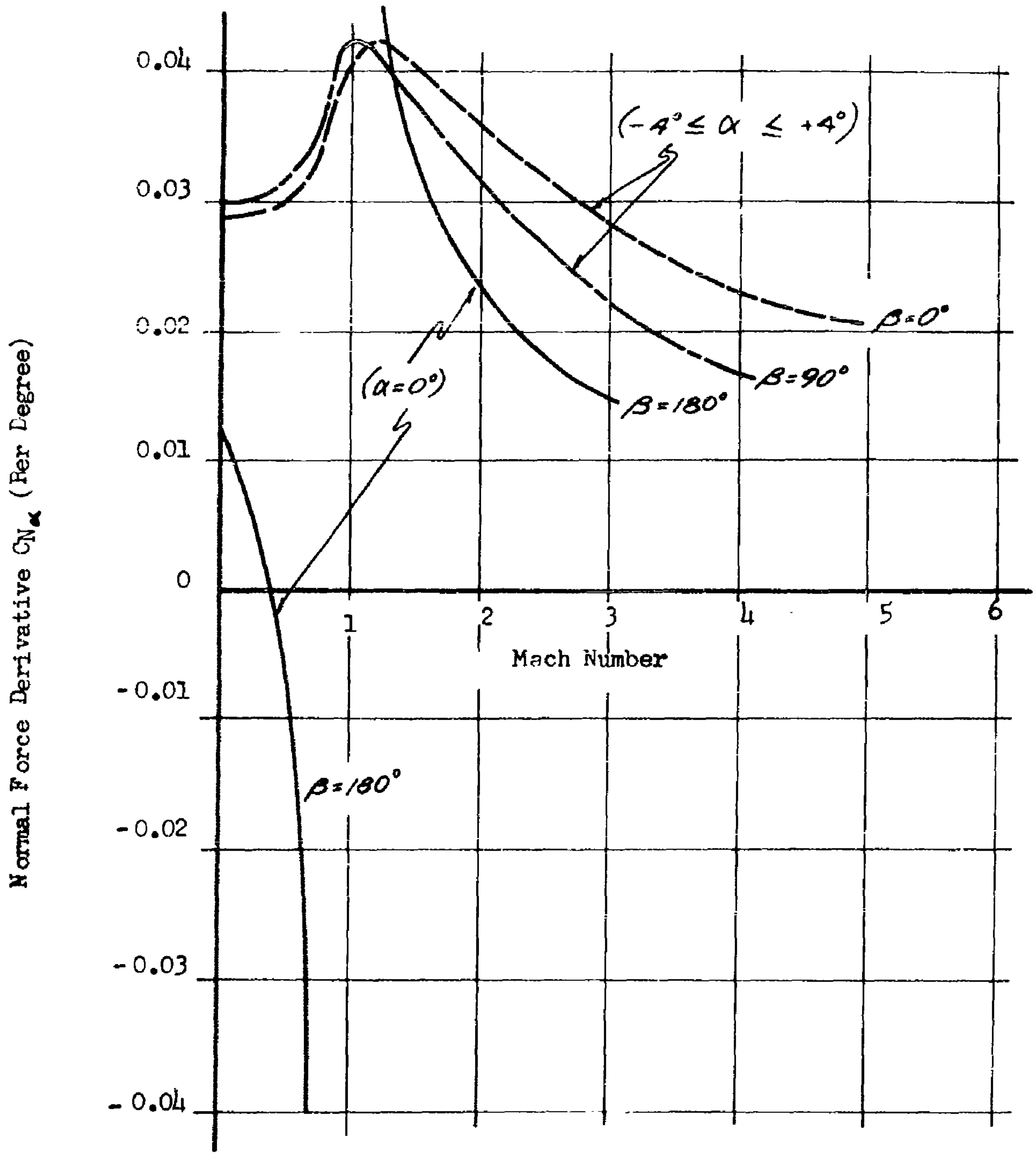


Figure 2.2 Variation of Normal Force Derivative with Mach Number

$t/c = 0.21$ power-off

derivative for the aft launch case, is applicable only at zero degrees angle of attack. For the aft launch condition, the flow separation from the blunt base resulted in the generation of a negative normal force for positive angles of attack. However, it must be emphasized that the aft launch data were obtained for the main booster-motor power-off conditions. It is expected that power-on data will show significant differences from that data presented here for aft launch.

The pitching moment derivatives as a function of Mach number are presented for the forward, crosswind and aft launch positions. The preceding discussion regarding normal force derivatives is equally applicable here. The pitching moment derivatives (Figure 2.3) for the forward ($\beta = 0^{\circ}$) and crosswind ($\beta = 90^{\circ}$) launch positions are applicable to a 4-degree angle-of-attack. The aft launch ($\beta = 180^{\circ}$) pitching moment derivative is extremely nonlinear with respect to angle of attack and hence is applicable only at a zero degree angle of attack. The flow separation causes negative pitching moments at high subsonic velocities. The aft launch data is again applicable only to the main booster-motor power-off condition.

The variation of the aerodynamic center of pressure as a function of sideslip angle is illustrated in Figure 2.4 for transonic and low supersonic velocities. The center of pressure moves off of the missile longitudinal axis as the sideslip angle increases from 0 to 90 degrees. This outward shift of the center of pressure is partially caused by the effective increase in the bluntness of the leading edge presented to the velocity vector as the sideslip angle increases, i.e., increased

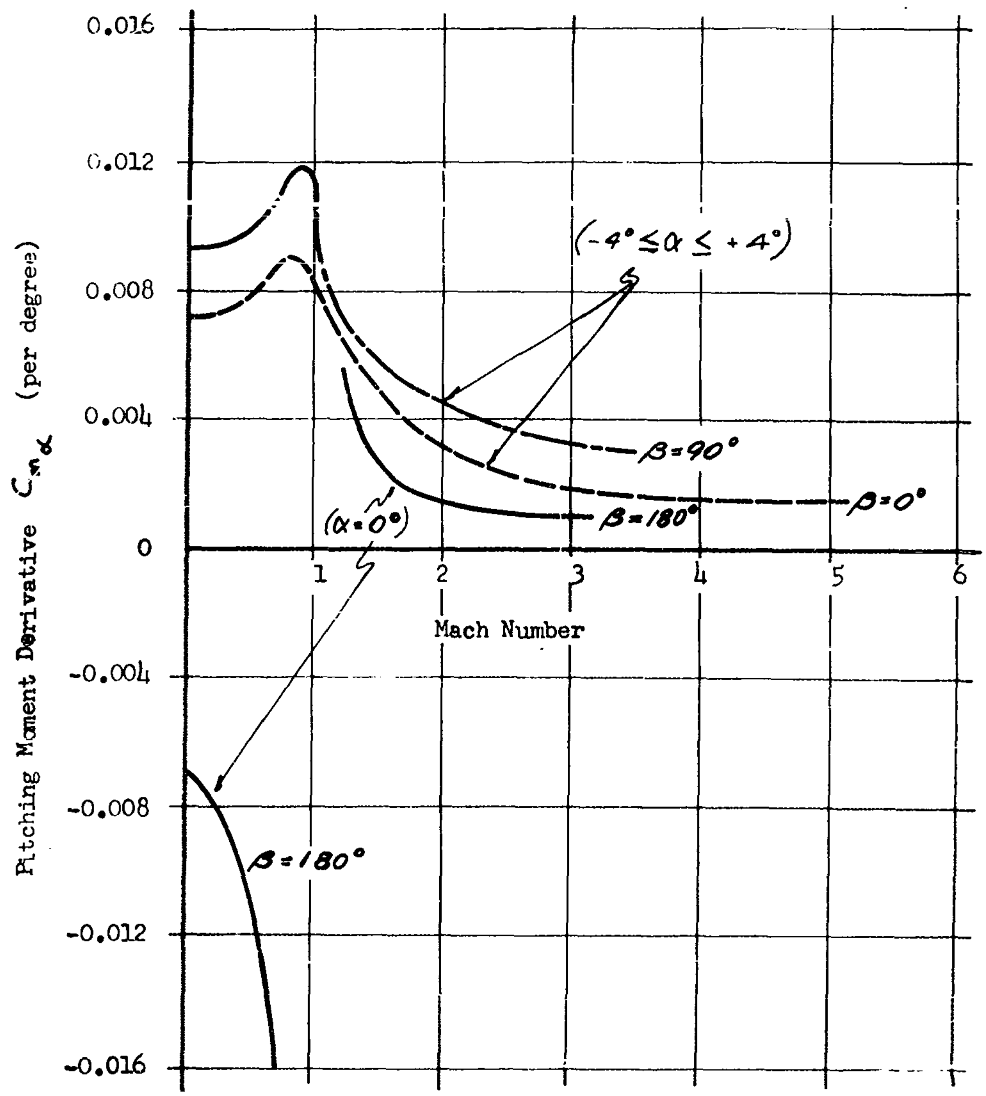


Figure 2.3 Variation of Pitching Moment Derivative with Mach Number.

$t/_c$ = 0.21 power off.

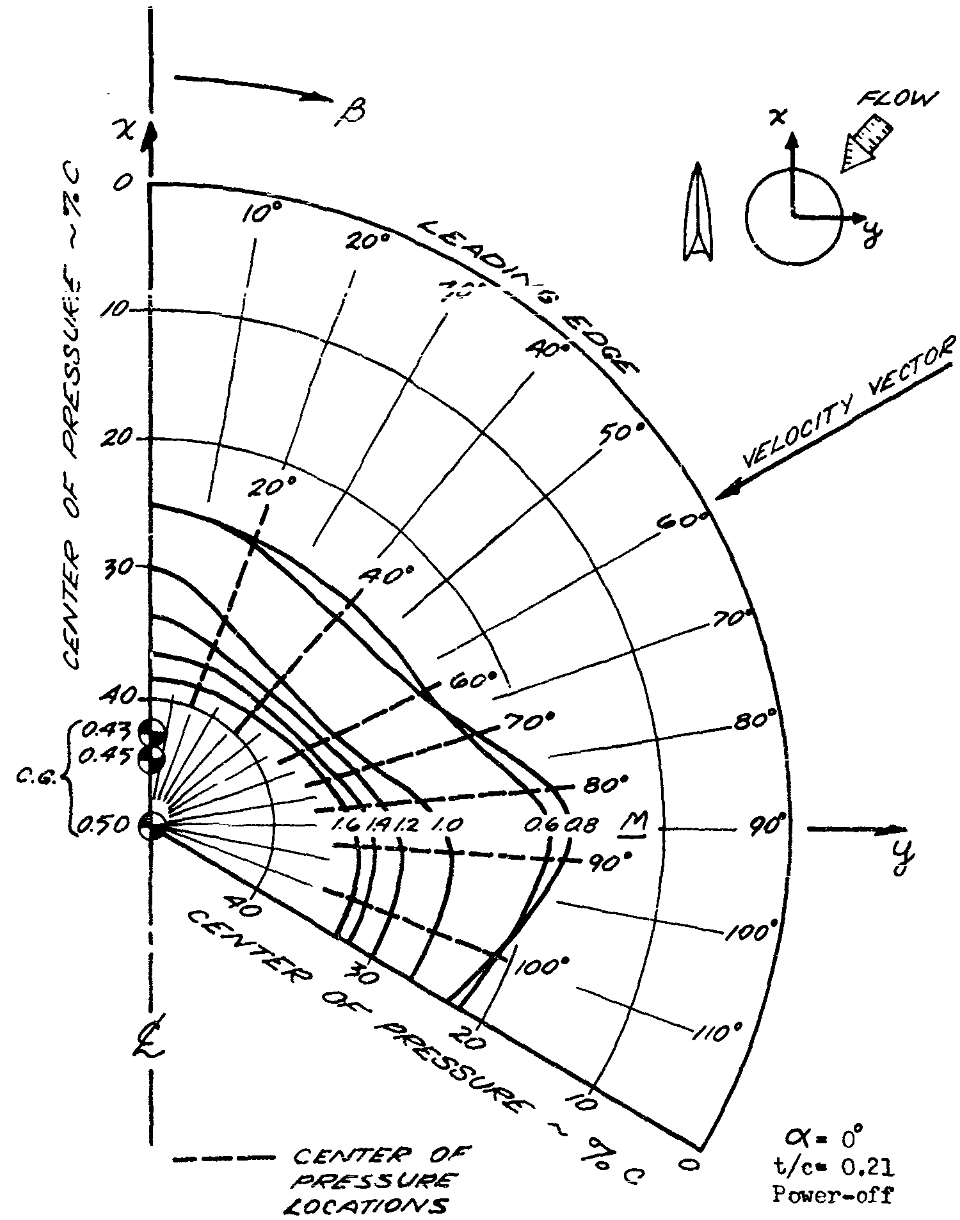


Figure 2.4 Variation of Center of Pressure with Angle of Sideslip

bluntness of a leading edge causes the center of pressure to move upstream. Figure 2.4 also indicates that the center of pressure does not lie on the velocity vector for sideslip angles greater than 20 degrees. This fact is caused by the asymmetry of the vehicle at sideslip angles other than 0 and 180 degrees and results in a small rolling moment about the velocity vector.

The axial drag coefficient as a function of Mach number is plotted in Figure 2.5 for the forward, crosswind and aft launch positions. The aft launch ($\beta = 180^{\circ}$) curve is an estimation based on the data obtained at Mach 0.6. A malfunction of the wind-tunnel balance precluded obtaining drag data at other aft velocities.

The main booster-rocket motors could not be simulated during the wind-tunnel tests thus preventing the determination of the exact effect of the exhaust on the basic aerodynamic characteristics. However, the power-on effects were estimated on the basis of the results obtained from control jet simulation. These estimates indicate that the booster-motor exhaust does significantly alter the lateral stability characteristics for the crosswind launch and the longitudinal stability characteristics for the aft launch. These estimates should be verified by actual test.

The reaction control jets, when operating, induce a change in the pressure field on the surface of the body. This induced pressure change can increase the effective control thrust if the resultant pressure differentials act in the same direction as the jet forces. Figure 2.6 is a fluorescent-oil photograph of the vehicle planform with one control

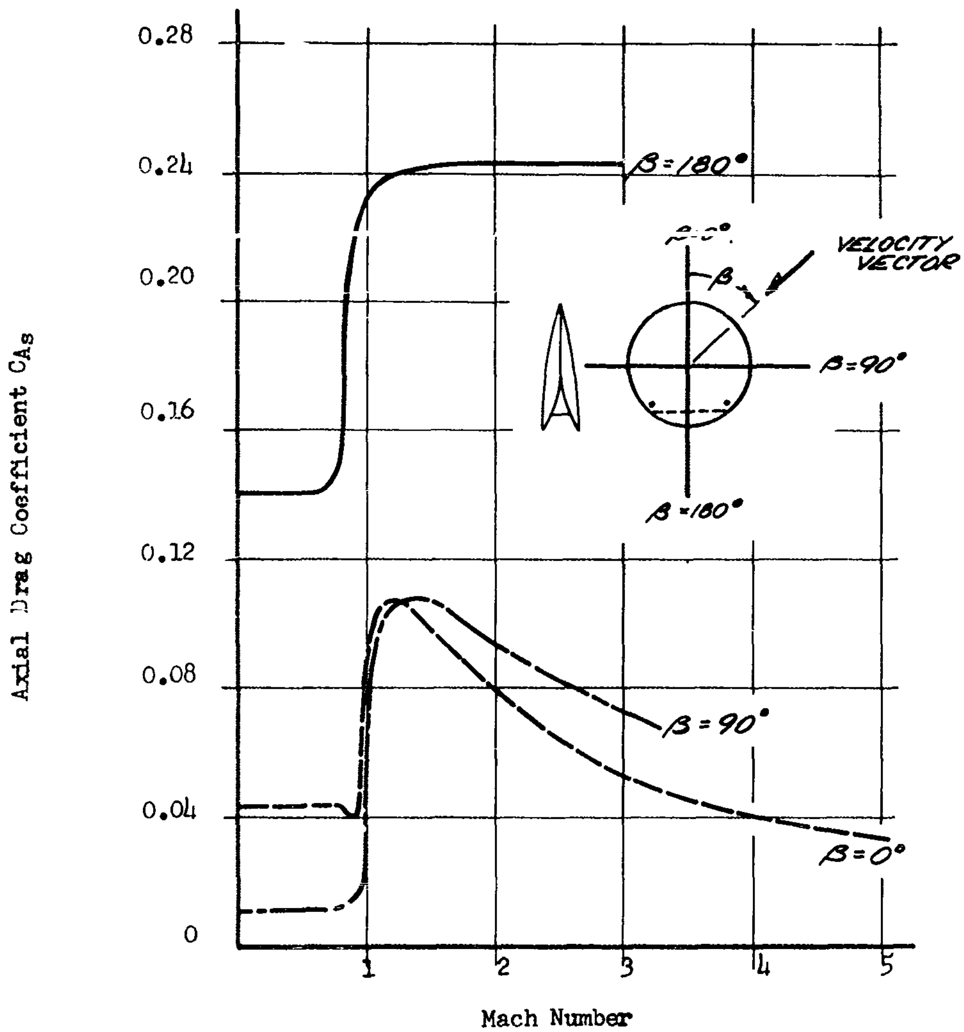


Figure 2.5 Variation of the Axial Drag Coefficient with Mach Number

Booster Motors Off

Control jets off

$\alpha = 0°$

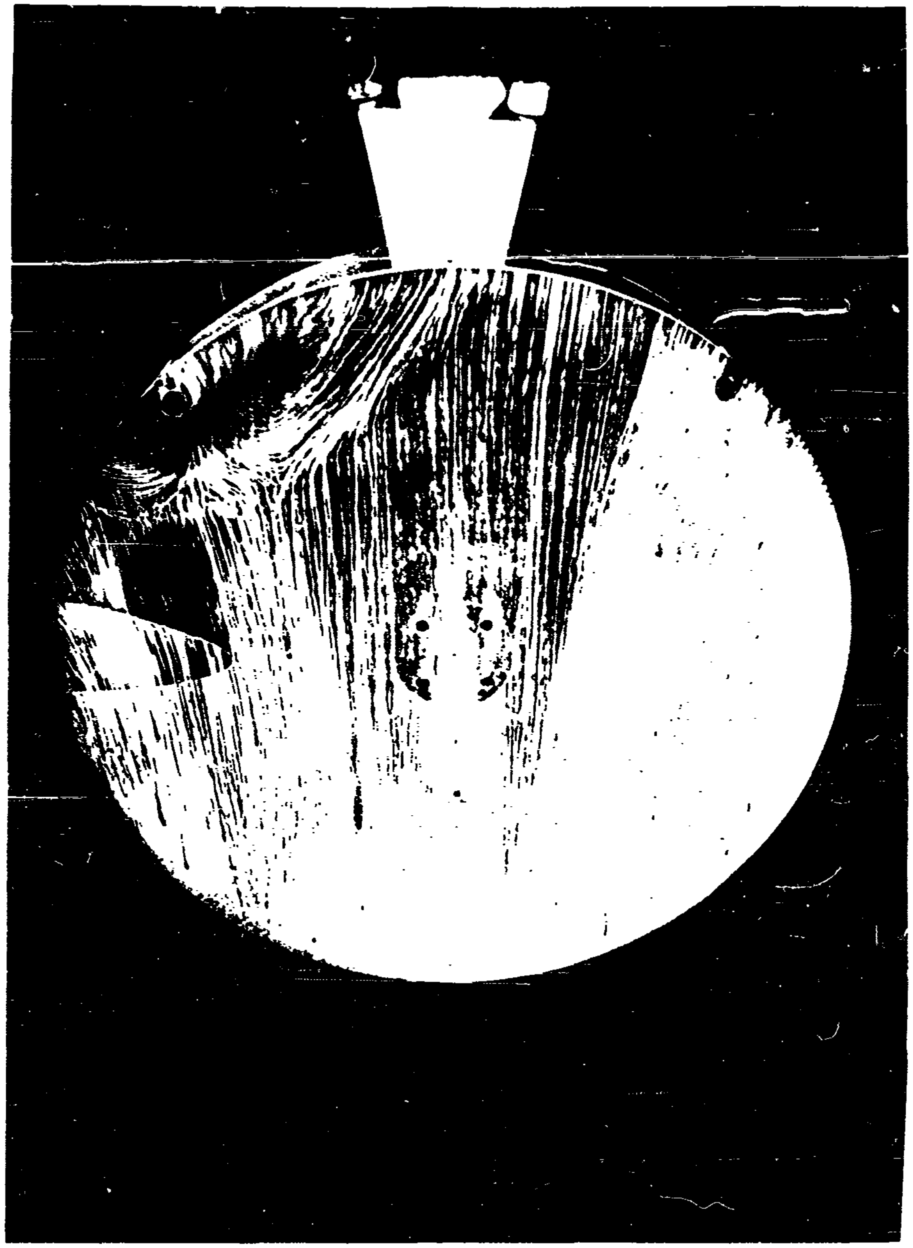


Figure 2.6 Fluorescent-Oil Photograph of Windward Side Right Jet On, M = 2.0, $\alpha = 6°$

motor operating. The lines shown on the photograph represent streamlines adjacent to the vehicle surface and indicate the disturbing effect of the control motor exhaust. The pressure measurements reveal an increased upstream pressure and a decreased downstream pressure as a result of the control exhaust. The variation in the pressure adjacent to the exhaust nozzles is illustrated in Figure 2.7. The force obtained by integrating the induced pressure is negative in the subsonic flow regime and positive in the supersonic flow regime. That is, the effective thrust of the control motors is decreased subsonically but is significantly increased supersonically. These results, obtained in the wind tunnel, are applicable to steady-state conditions only, i.e., constant velocity and constant control thrust. The nonlinear control system is such that the thrust will be produced in relatively short pulses. Since the flow field induced by the jet exhaust does require a finite time to form, the results presented here represent an absolute maximum effective thrust alteration.

The near sea level atmosphere in which the Feasibility Test Vehicle will be launched presents an extremely severe environment for a highly maneuverable vehicle such as PYE WACKET. One of the several major advantages of this configuration is its maneuverability at high altitudes. As a result of this high maneuverability, extremely large "g" loading occurs for moderate angles of attack in the relatively dense sea level atmosphere. The maximum "g" control capability of the Feasibility Test Vehicle at sea level is presented in Figure 2.8. At approximately Mach 2.4, the configuration becomes neutrally stable and hence is

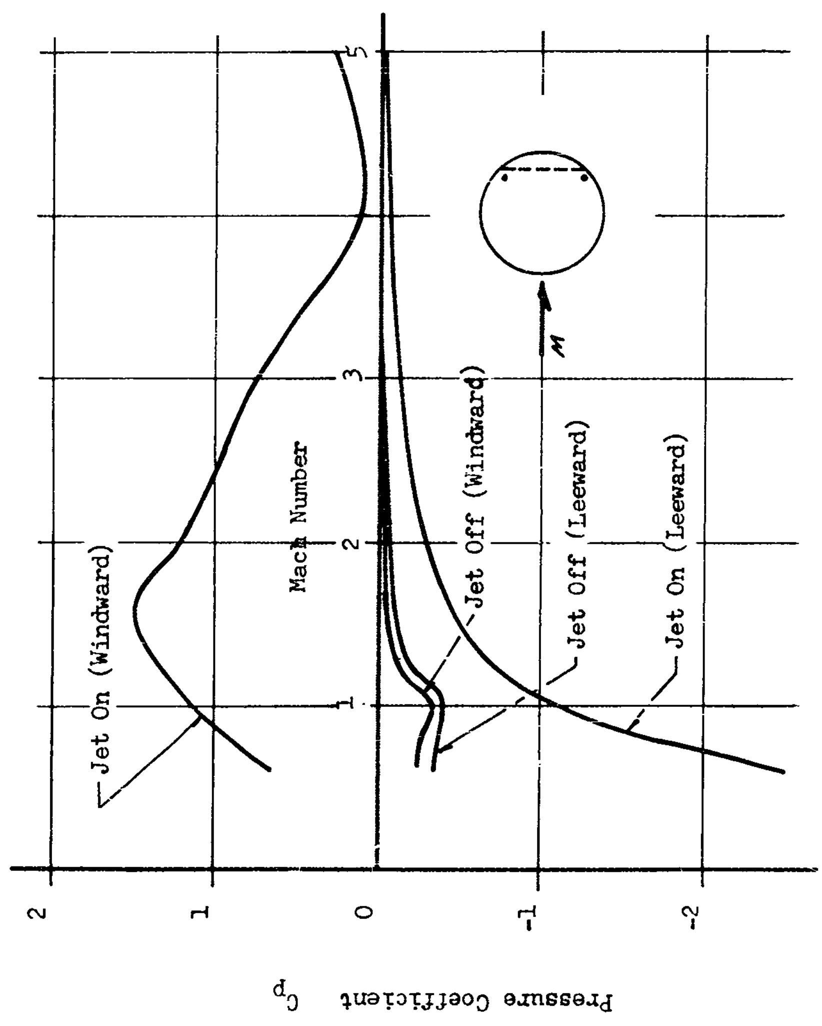


Figure 2.7 Comparison of Maximum and Minimum Pressure Adjacent to Reaction Jets

$\alpha = 0°$ $\beta = 0°$ $P_j = 700$ psia

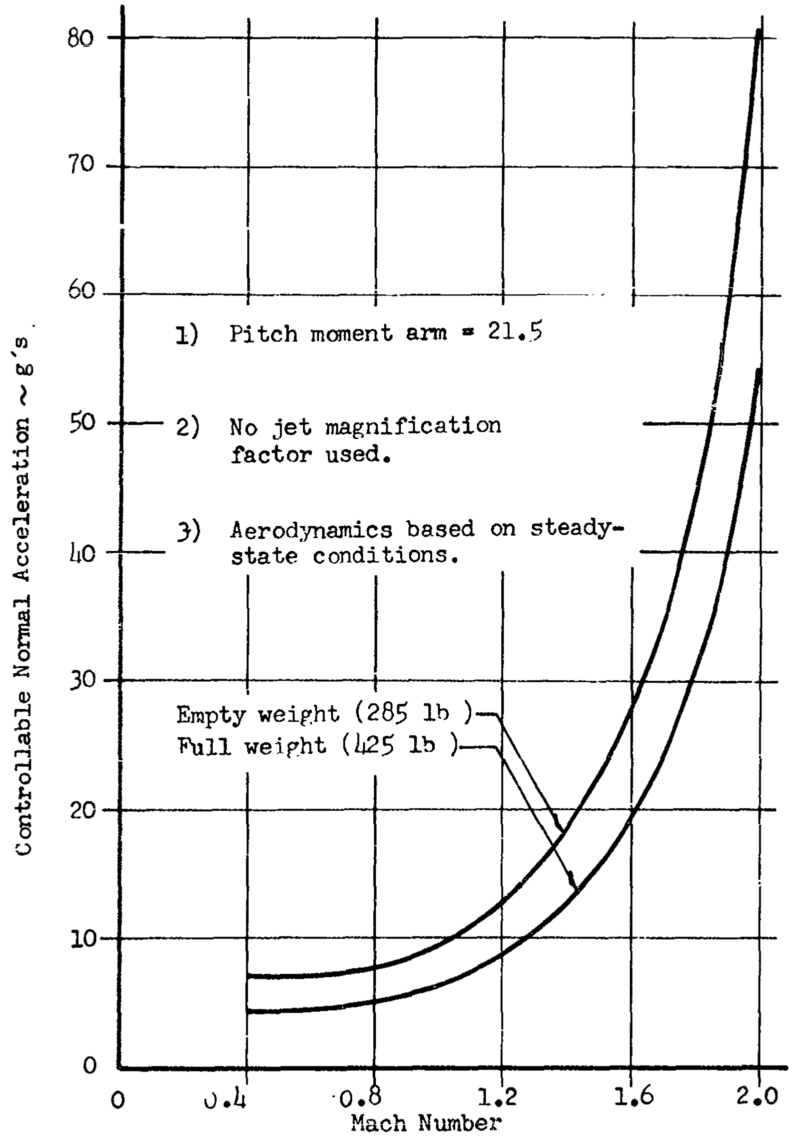


Figure 2.8 Variation of Maximum Maneuverability with Mach Number

$\beta = 0^{\circ}$ $T_j = 500 \text{ lb}/_{\text{jet}}$

Sea Level C.G. = 0.43 C

controllable at very large normal accelerations.

The launch velocities for the Feasibility Test Vehicle at sea level are related to the equivalent velocities at higher altitudes at which a tactical version of the missile system would function. The two main factors affecting this correlation are that the dynamic pressure decreases as altitude increases (Figure 2.9), and that the center of pressure shifts in the downstream direction as the velocity increases and thereby decreases the aerodynamic moment arm. These correlation factors are applied to the sea level conditions to indicate the equivalent altitude maneuverability. The theoretical controllable maneuvers for the Feasibility Test Vehicle exceeded 250 g's for Mach 3 and higher velocities. To illustrate the high altitude, high velocity maneuvering capability within the structural limitations, a 15-degree angle of attack at 60,000 feet altitude was chosen. The results are illustrated in Figure 2.10. A comparison of the data presented in Figure 2.10 and 2.8 clearly indicates the severity of the proposed Feasibility Test Vehicle launching environment.

Aerodynamic wind-tunnel data are obtained for steady-state conditions. The exact effect of the basic aerodynamics can only be determined by introducing them into a dynamic, time varying simulation such as that utilized in the autopilot studies.

The preceding summary of aerodynamic data obtained and analyzed in this contract is extremely cursory. For a detailed presentation of the aerodynamics, see Volume II of this report.

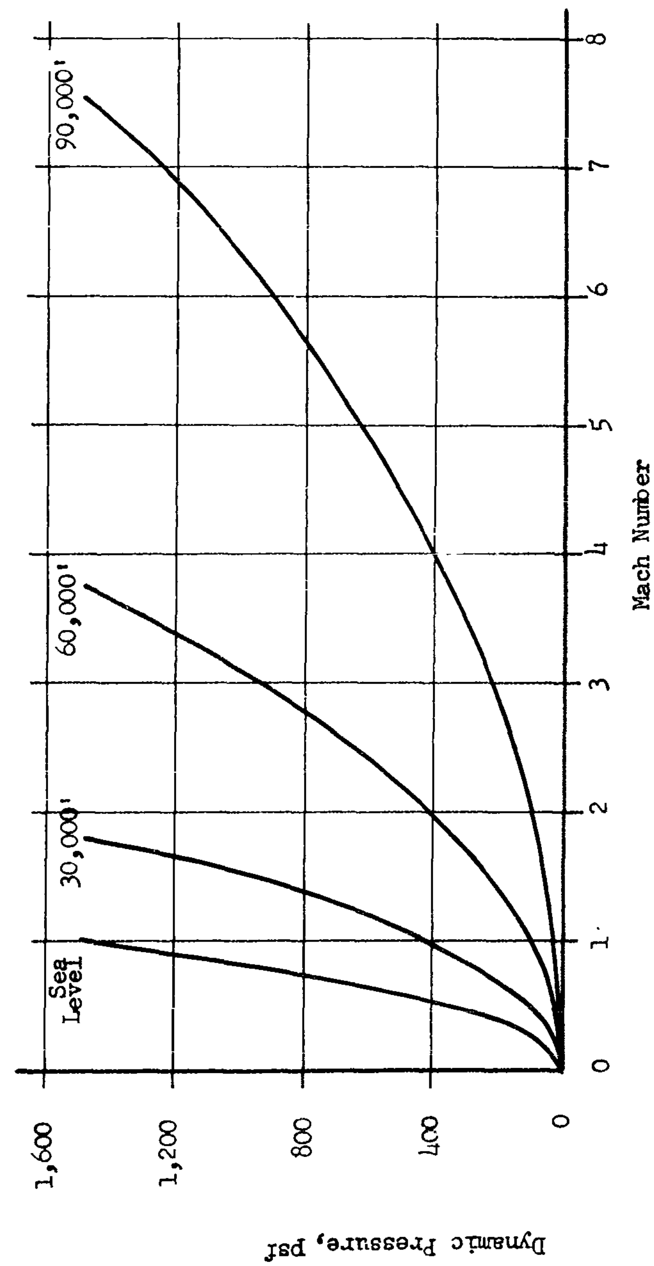


Figure 2.9 Variation of Dynamic Pressure with Mach Number

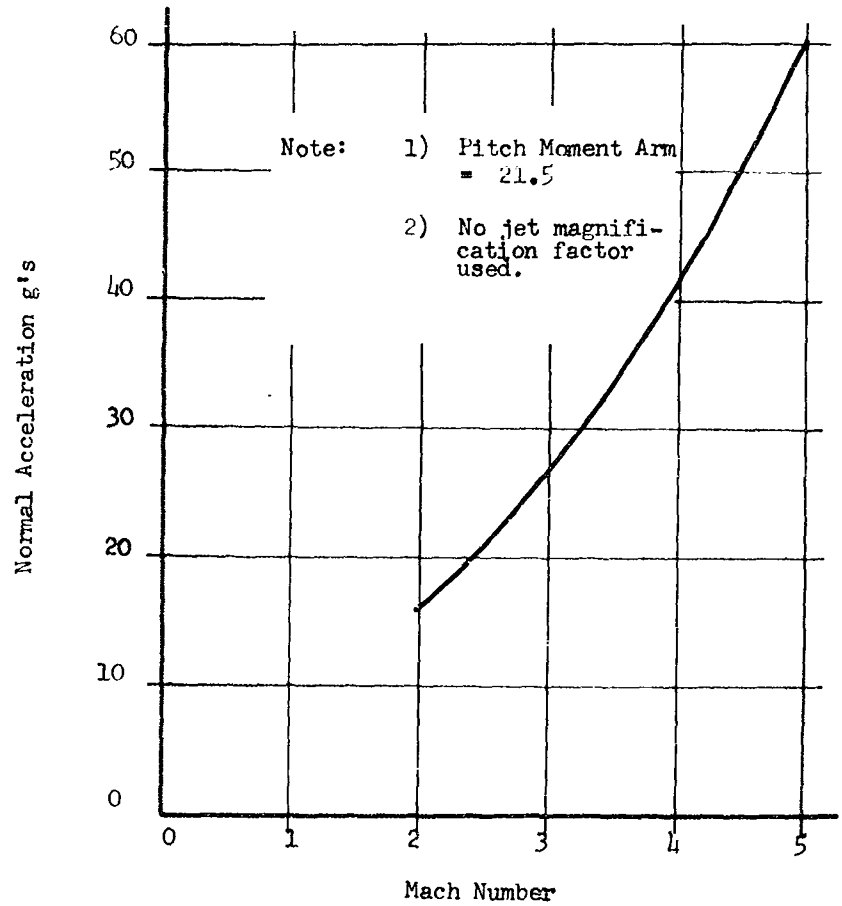


Figure 2.10 Resultant Maneuverability

$\alpha = 15^{\circ}$ Altitude = 60,000 ft. Weight = 285 lb

T_j = 500 lb/jet CG = 0.43 C

Section 3.0
STRUCTURE

The basic design configuration of the PYE WACKET Feasibility Test Vehicle was greatly influenced by the aerodynamic and control system considerations, booster-motor configuration, and component cost and producibility. The vehicle is 60 inches in diameter with a 0.21 thickness-to-chord ratio. The main propulsion is provided by three M58A2 rocket motors aligned parallel to the missile longitudinal axis. The M58A2 rocket motor is currently in production status and as such is readily available. The booster motor has demonstrated a high degree of reliability through its extensive use in the Falcon missile. The burnout velocity attainable with the three M58A2 motors in the Feasibility Test Vehicle is presented in Figure 3.1. The boosters are located in a forward position compatible with a component arrangement (Figure 3.2) that ensures a 43% chord center-of-gravity location. This forward c.g. location decreases the aerodynamic moments and thus decreases the development work required for the control system. The weight distribution of the major components, the corresponding moments and center-of-gravity locations are presented in Table 3.1 for the launch, booster-motor burnout, and flight termination conditions. The resulting moments of inertia for the pitch, roll and yaw planes are listed in Table 3.2.

The main structure (weldment) of the Feasibility Test Vehicle consists of four magnesium alloy (AZ 31B) channels criss-crossing the

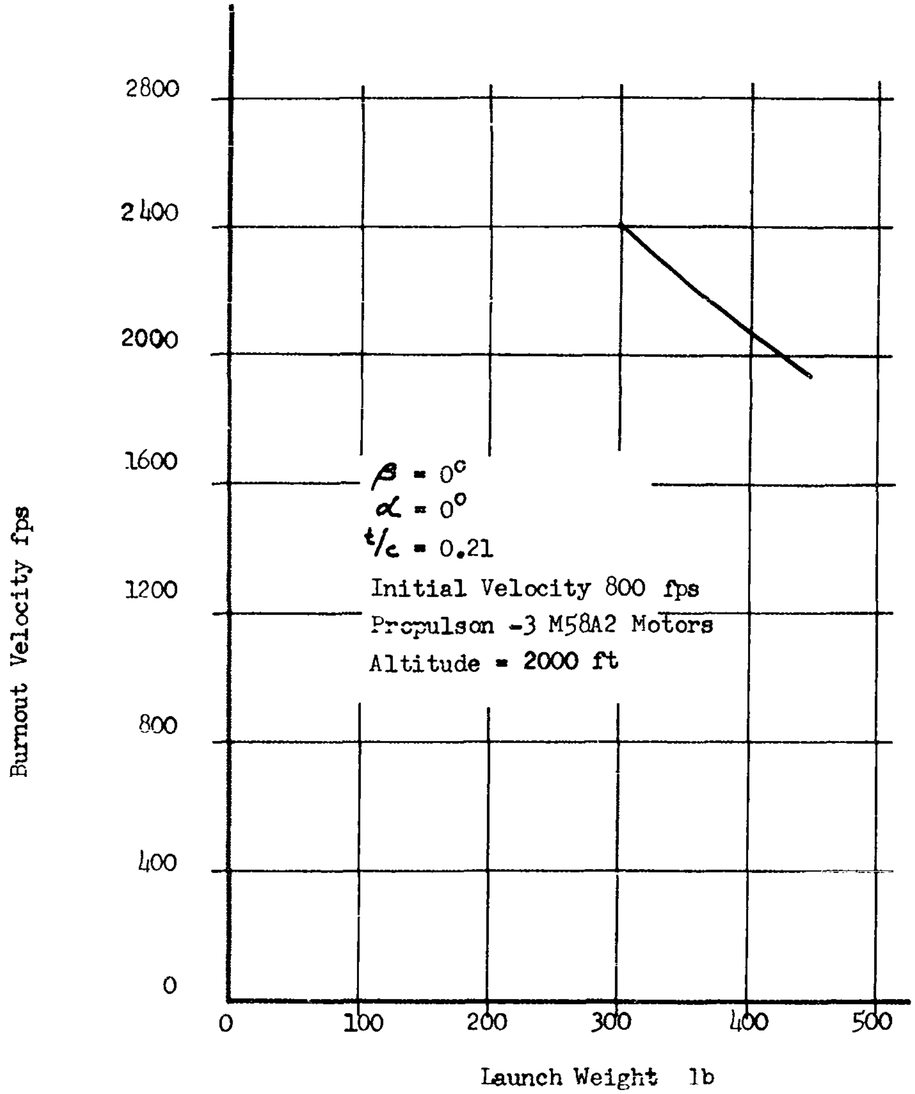


Figure 3.1 Variation of Burnout Velocity With Launch Weight

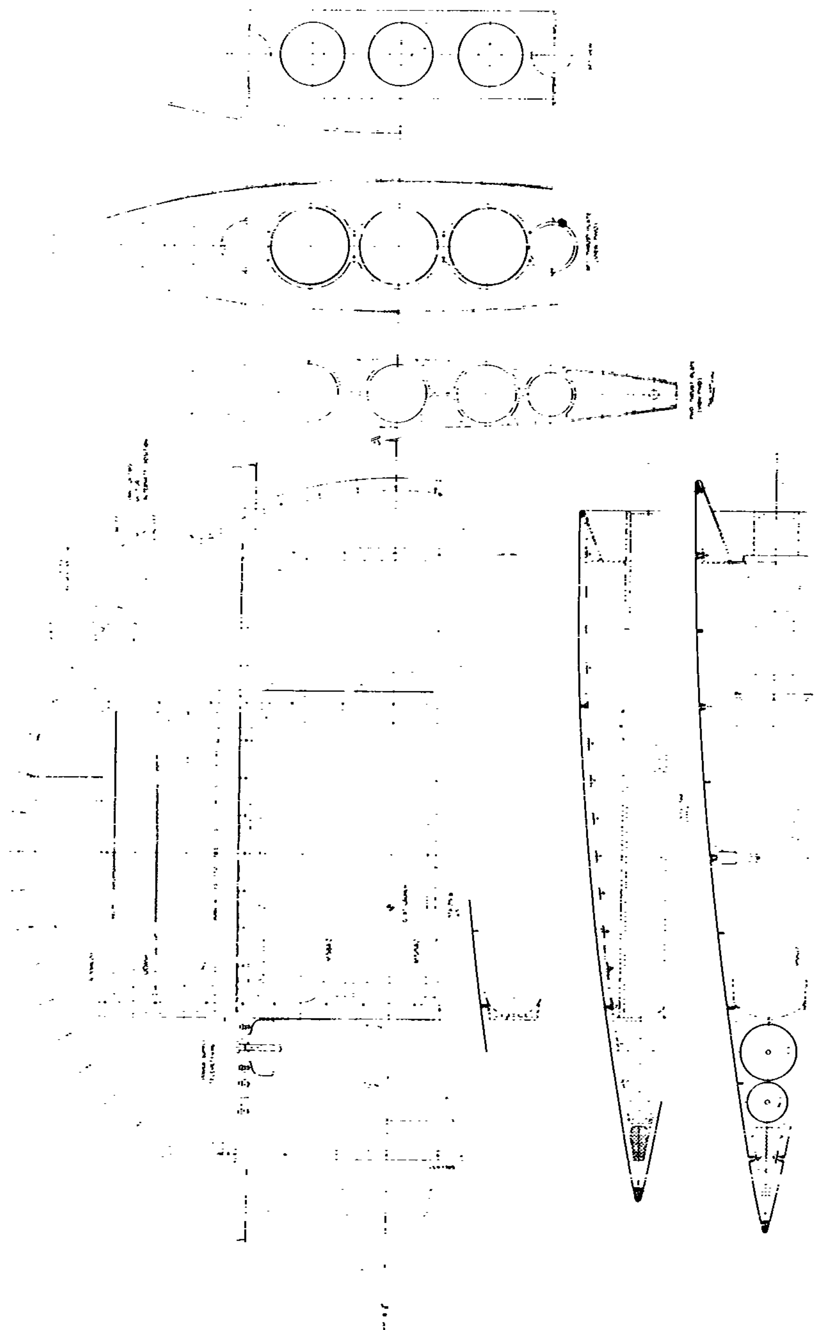


Figure 3.2 Feasibility Test Vehicle Layout

Table 3.1 Weight and Balance

COMPONENT		WEIGHT lb			MOMENT in-lb		
		LAUNCH	BURNOUT	EMPTY	LAUNCH	BURNOUT	EMPTY
STRUCTURES		66	66	66	2112.0	2112	2112
PROPELLANT SYSTEM		37	37	37	938.0	938	938
PROPELLANT		35	18	--	632.0	348	---
CONTROL MOTOR		30	30	30	1500.0	1500	1500
ELECTRONICS		40	40	40	640.0	640	640
BOOSTERS		137	42	42	4450.0	1605.	1605
MISC.		11	11	11	279.0	279	279
BALLAST		69	69	69	408.0	408	408
TOTALS		425	313	295	10959	7830	7482
Center of Gravity	(in)	25.8	25.0	25.3			
	% C	43	41.6	42.0			

missile planform (see Figure 3.3). This weldment, when coupled with the missile skin, provides structural rigidity in all planes. The launcher tubes and booster support are an integral part of this weldment in order to ensure maximum parallelism and alignment. The configuration was designed such that the skin can be removed to allow access to all components for easy assembly, checkout and service operation.

Table 3.2 Feasibility Test Vehicle Moments of Inertia

Condition	Weight (lb)	Inertia (slug-ft^2)	
Launch	425	31.7	Yaw
		11.6	Roll
		21.4	Pitch
Burnout	330	28.7	Yaw
		10.1	Roll
		19.8	Pitch
Empty	295	26.7	Yaw
		9.8	Roll
		18.9	Pitch

The conditions noted in the above table are:

launch ----- basic missile with propulsion propellant and reaction control propellant,

burnout ---- basic missile plus reaction control propellant, and

empty ------ basic missile only.

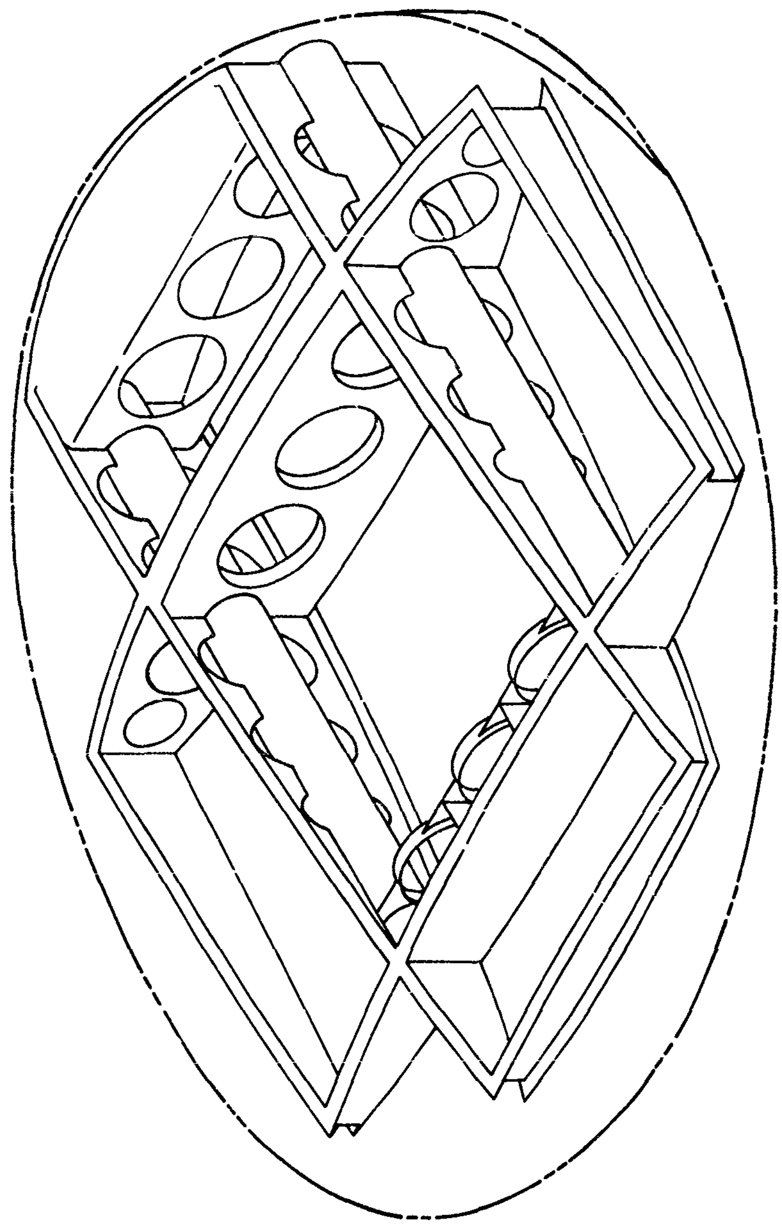


Figure 3.3 Main Structure Weldment

The cast magnesium-alloy skin shown in Figure 3.4 was selected for the Feasibility Test Vehicle. The choice was governed by the current demonstrated capabilities in forming a configuration equivalent to the missile skin. If the capability of fabricating laminated honeycomb-sandwich structures into the required compound surface can be proven, this type of construction will be ideal for the vehicle shroud.

Figure 3.5 illustrates the proposed launcher mounted on a high speed rocket sled. The launcher is a twin-rail system capable of 360 degree rotation in the horizontal plane and 90 degrees in the vertical plane. The launcher rails slide into the two longitudinal cylinders on either side of the booster motors (see Figure 3.3). A simple locking device to hold the missile prior to booster ignition will be designed into the system as a part of the igniter-circuit interlock. The launcher design incorporates sufficient ground clearance to allow at least 100% safety margin beyond the predicted missile "drop-off" during aft launch and to minimize the sled and ground turbulence imposed upon the missile. The launcher-body characteristics are listed in Table 3.3 for three diameters of the launcher rail. The final choice of these diameters will depend upon the detailed environmental studies proposed as a first task in the next phase of the program. Since the divergent Mach number and natural frequencies are high in all three configurations, sufficient latitude is available in the space allocated to design a system detuned from the critical disturbing frequencies.

For a detailed discussion of the structural aspects of the Feasibility Test Vehicle, see Volume III, Section 1.0, of the report.

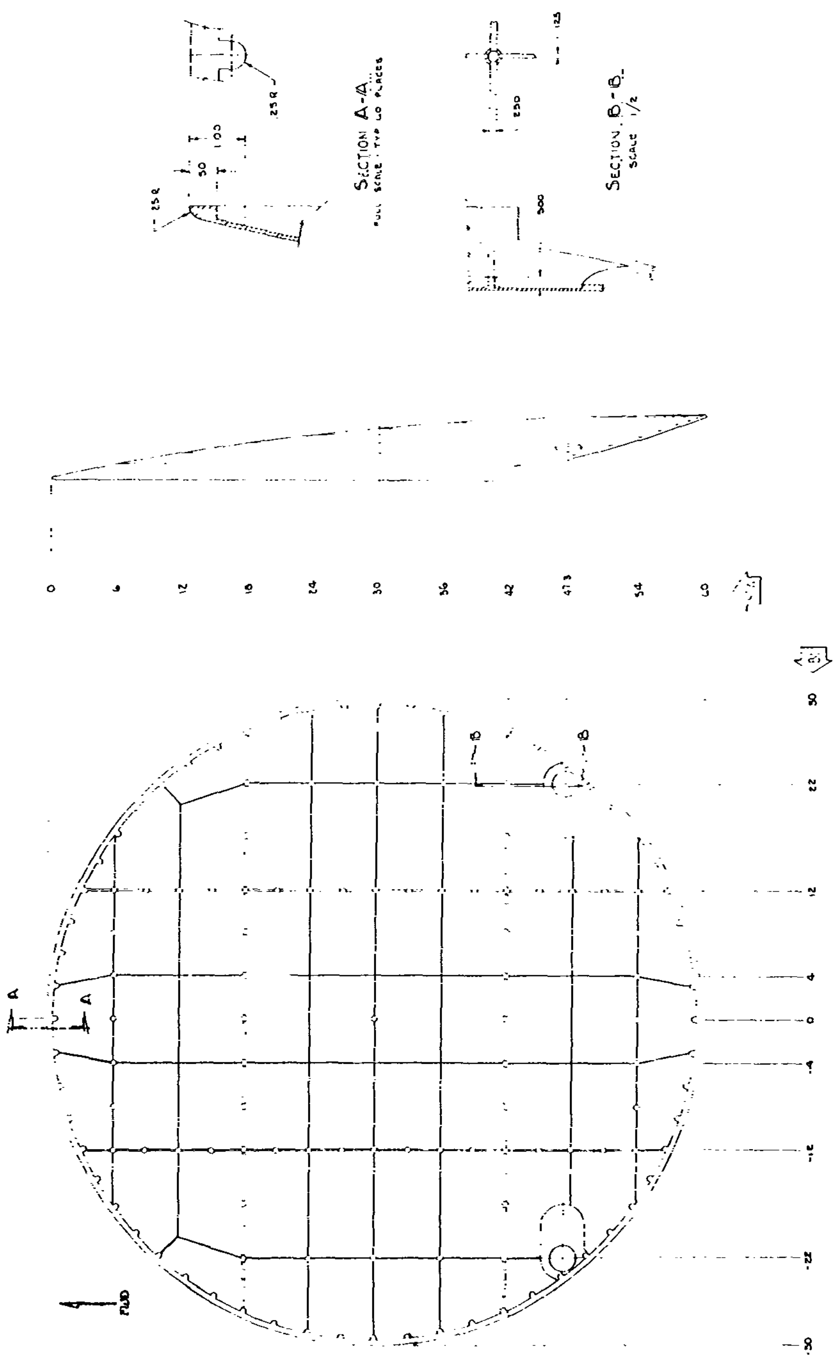


Figure 3.4 Layout of Cast Magnesium Skin

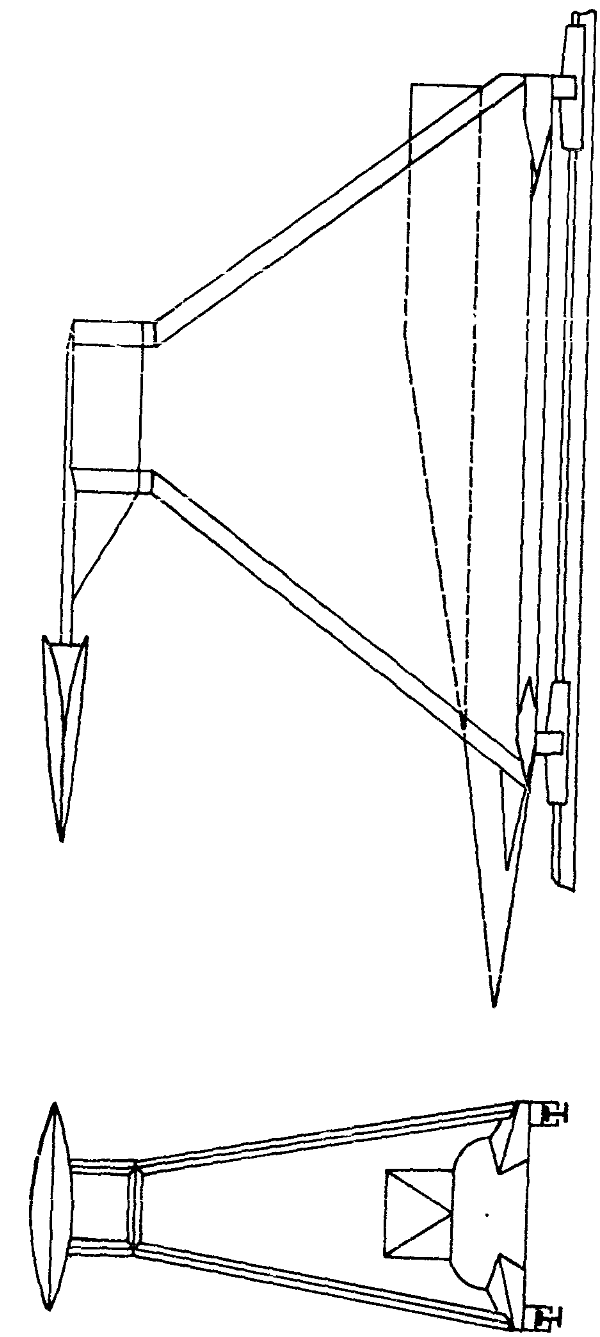


Figure 3.5 Launcher Configuration (forward launch position)

Table 3.3 Launcher-Body Characteristics

Launcher Rail Diameter inches	Resonant Frequencies - cps					Divergence Mach No.
	Coupled Pitch and Vertical Translation		Roll	Coupled Yaw and Lateral Translation		
	First Mode	Second Mode		First Mode	Second Mode	
2.75	14.2	337	75.8	30.8	253	2.35
3.00	16.9	402	90.3	36.8	301	2.79
3.25	19.8	472	106.0	43.2	355	3.28

Section 4.0
REACTION CONTROL SYSTEM

The blunted lenticular, circular planform configuration of the PYE WACKET possesses many advantages as an airborne weapon. In general, these advantages are high lift-to-drag ratio, high maneuverability and omnidirectional launch capability. The vehicle must not only be stabilized and controlled but this must be accomplished in a manner which best exploits the full potential of this unique shape.

The control system studies conducted under this contract were directed toward an application in a Feasibility Test Vehicle. The intended mission of this vehicle is a flight test to prove the feasibility of the concept as an airborne weapon. Inherent in the specific application is the severe but restricted sea level flight environment, relatively short flight time, and the required maximum use of off-the-shelf items (minimum development effort).

The control moments will be developed in the three planes, pitch, roll and yaw, from six jet nozzles. The pitch and roll signals are electronically combined to drive four common jets. The yaw system is separate and operates two control jets independent of the pitch-roll autopilot. Although three-dimensional control can be obtained with four nozzles, the use of six results in a lower maximum thrust requirement, an economy of control propellant, and a decrease in development work required for the propellant tankage, control motors and the autopilot.

The reaction motors will be driven by a bistable element, i.e., on demand from the autopilot, the control jets will be commanded to a fully open or a fully closed position. This philosophy alleviates the problem of developing a propellant control valve whose flow is directly proportional to the input signal level, and whose natural frequency is sufficiently high to satisfy the required response times.

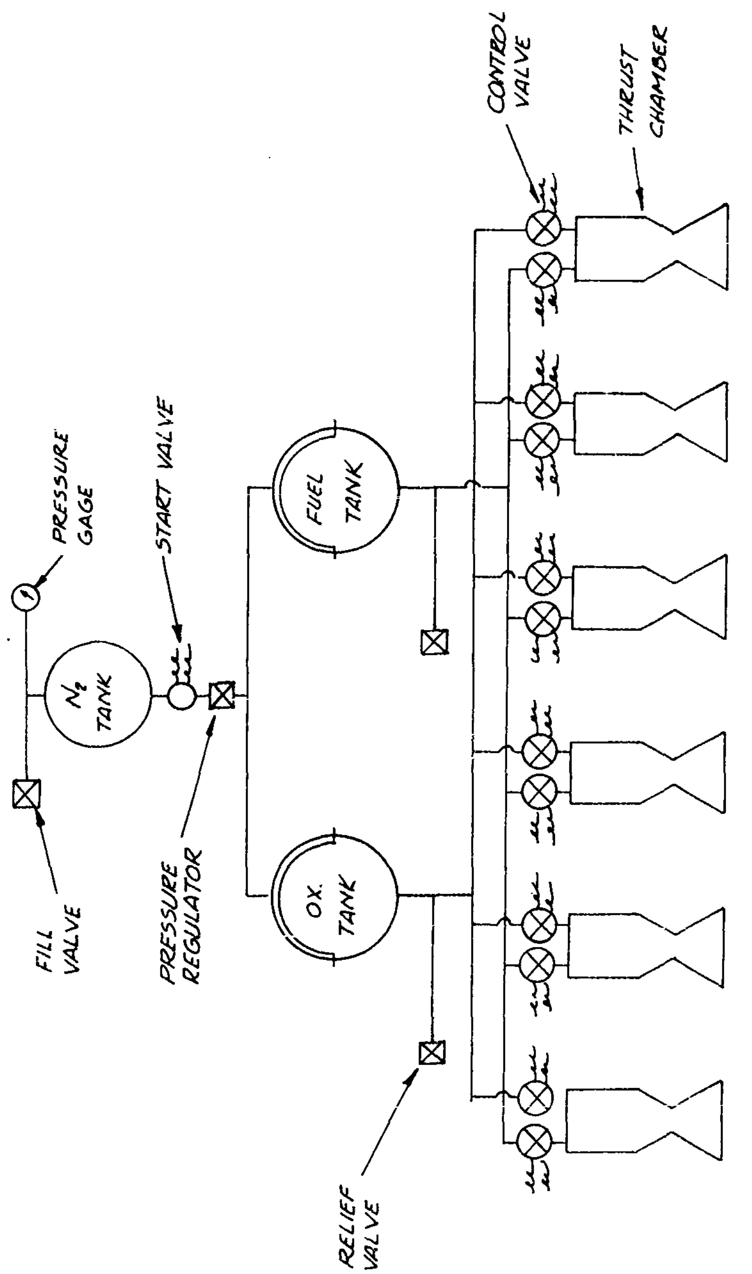
The thrust will be produced by a nitrogen pressurized, hypergolic bipropellant system (Figure 4.1). This conclusion emanated from a detailed analytical study and an industry survey of the available systems and components. The arrangement of the fuel and oxidizer tanks, the pressurization system and the control motors is illustrated in Figure 3.2.

The interaction of the control jet exhaust and the free stream has a significant effect on the control forces. The reaction jet induces a high pressure area upstream of the jet analogous to the stagnation region upstream of a solid body. This high pressure acts in the same direction as the thrust vector. However this effect is opposed by the extremely low pressure on the leeward side of the jet. This pressure is lower than that expected behind a solid body and therefore suggests that a jet pumping effect is present. The total effective control moment is then a summation of the moment caused by the jet and the additional moment occuring due to disturbed aerodynamics.

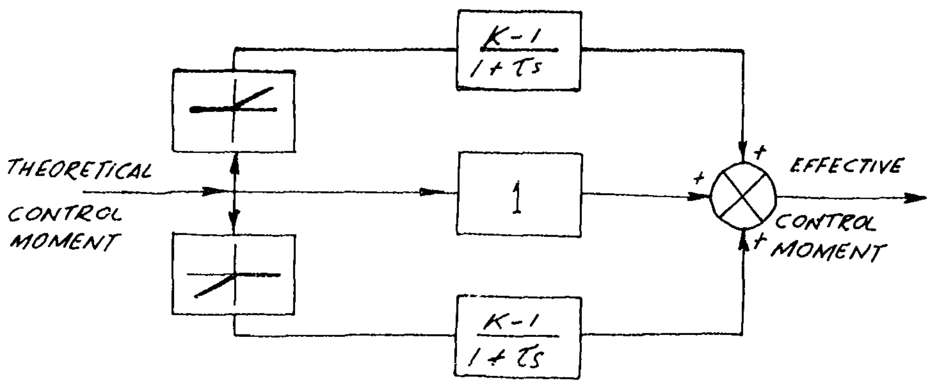
The interaction effect is reported as a magnification factor, K, defined by the ratio of the actual moment on the vehicle (total effect

Figure 4.1

SCHEMATIC OF A BIPROPELLANT REACTION CONTROL SYSTEM



of reaction jets) to the theoretical moment (thrust vector alone). The magnification factors for the wind-tunnel conditions were converted to the Feasibility Test Vehicle conditions and are shown in Figure 4.2 for the forward, crosswind and aft launch positions. Since the background information for the magnification factors originated in a wind tunnel, the data are applicable to steady-state, power-off conditions only. The shock pattern induced on the body by the interaction effect takes a finite time to develop. This time delay is especially significant when the pulsing nature of the control thrust produced by the nonlinear autopilot is considered. The effect is easily accountable in analog simulation as shown in Figure 4.3.



K = magnification factor

τ = time constant for interaction shock wave development

Figure 4.3 Analog Simulation of Interaction Effect

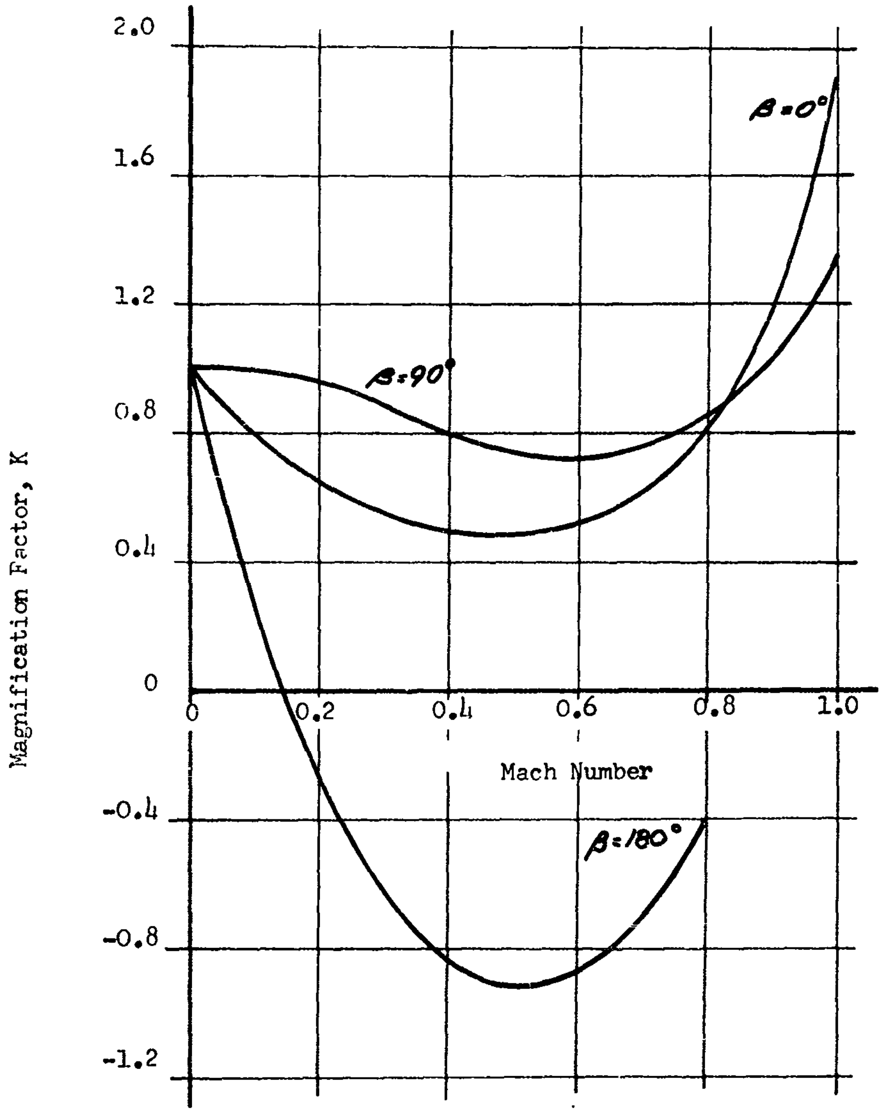


Figure 4.2 Variation of Magnification Factor With Mach Number

Sea Level 500.lb/jet c.g. 0.43 C

(Steady-state Conditions Only)

The effect can also be computed analytically in the manner discussed in Volume III, Section 2.2, of this report. The true effect of the magnification factor is best evaluated by incorporating the steady-state results into the time varying missile simulation used in the finalized autopilot studies.

The studies involving the reaction control, including the contributions from the aerodynamic and autopilot studies, culminated in the dictate of the required 500-pound thrust level and the 5 millisecond response time. Figure 4.4 illustrates a typical 300-pound thrust motor response obtained in a Convair company-sponsored test program. The motor does not represent the ideal case for the Feasibility Test Vehicle; however, it does indicate the present state-of-the-art and points toward the development work required in this field.

The detailed study and analysis of the reaction control motors is discussed in Volume III Section 2.2, of the report.

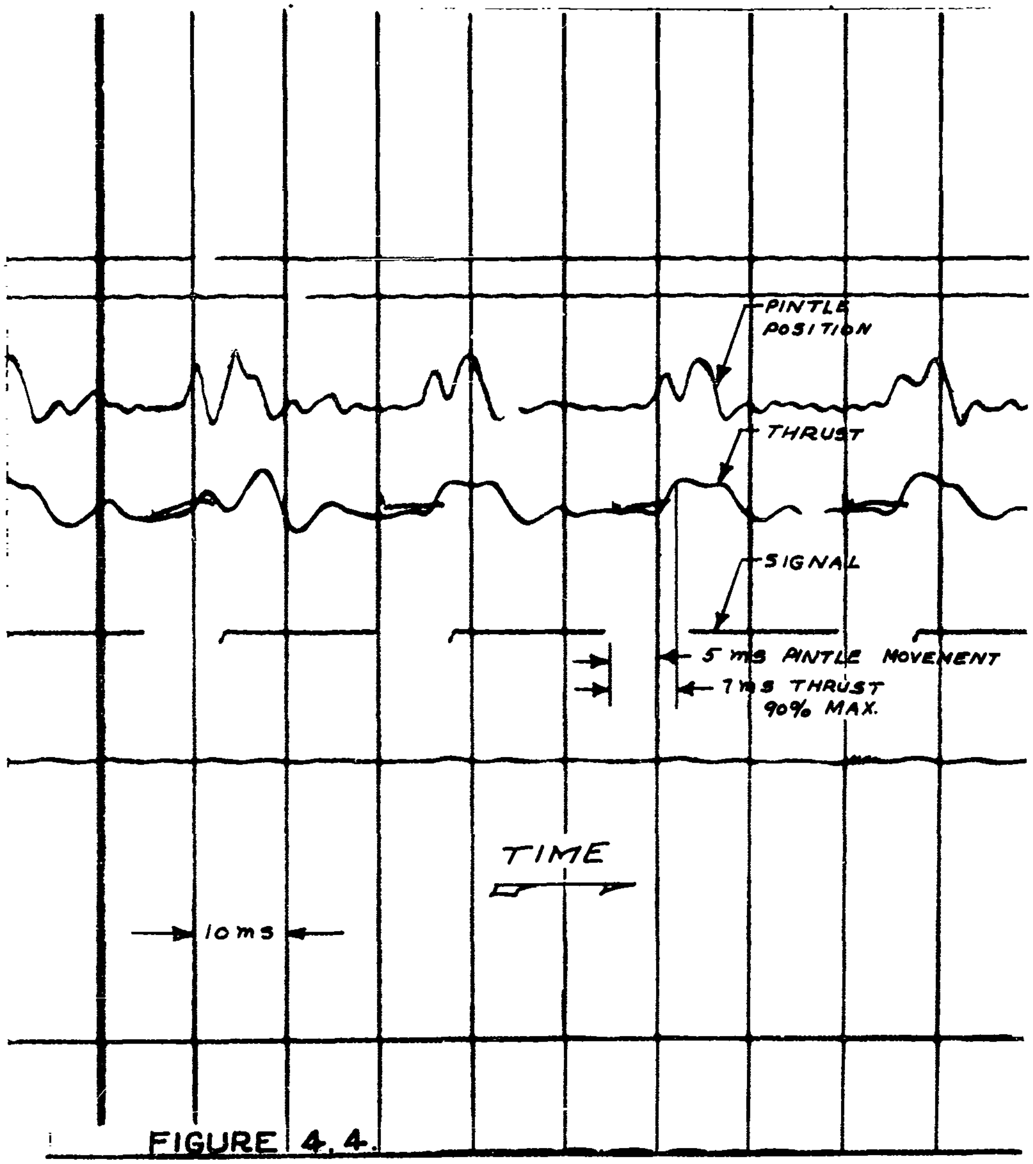


FIGURE 4.4.
TYPICAL 300 LB MOTOR RESPONSE

Section 5.0
AUTOPILOT

The autopilot developmental studies were conducted on the premise of a 60-inch diameter, 21% thickness-to-chord ratio Feasibility Test Vehicle launched at or near sea level from a high-speed rocket-sled. Since the main purpose of the vehicle is the demonstration, through eventual flight test, of stabilized and controlled flight from an omni-directional launch, the autopilot was studied to the greatest extent possible in forward, crosswind and aft launch conditions. The work was initiated on a simplified basis with control in each plane considered separately for each mode of operation at the various launch attitudes. These separate autopilots were then combined and finalized parameters established on a time varying three-dimensional simulation of the entire missile.

The autopilot for the Feasibility Test Vehicle is nonlinear in nature. The controlling error signals are first developed according to linear control equations and then fed to a signum computer, the output of which is proportional only to the sign of the input signal. The output of the signum computer is used to drive the reaction control motors to a state of either full thrust or zero thrust. Each plane of control therefore has a characteristic dither frequency at which it oscillates during steady-state conditions. The frequency and amplitude of these oscillations are maintained such that almost zero airframe perturbations are present during the steady-state mode of operation.

The roll and pitch signals are combined to operate four common control motors in a time sharing sequence. As such only pure roll or pure pitch control moment is possible at one time. To ensure stability, it is essential that the system controlling the motion about the unstable aerodynamic axis exercise the dominant control. That is, during forward launch the airframe is unstable in the pitch plane, hence the pitch autopilot dominates; during crosswind launch the airframe is initially unstable in the roll plane, hence the roll control dominates. The change in the controlling mode is reflected in the gain changes programmed to occur when β is approximately 20 degrees. The autopilot block diagram in Figure 5.1 indicates the proposed implementation of the control methods. The control equations for the pitch and roll control are respectively,

$$\mathcal{E}_p = K_1(\eta - \eta_c) - K_2\dot{\theta}(1+\tau_p s), \text{ and}$$

$$\mathcal{E}_r = K_4(\phi_c - \phi) - K_3\dot{\phi}(1+\tau_r s)$$

where $\mathcal{E}_p$ = controlling error signal in the pitch plane

η = normal acceleration of the airframe

$\dot{\theta}$ = angular pitch rate,

$\mathcal{E}_r$ = controlling error signal in the roll plane,

ϕ = roll angle,

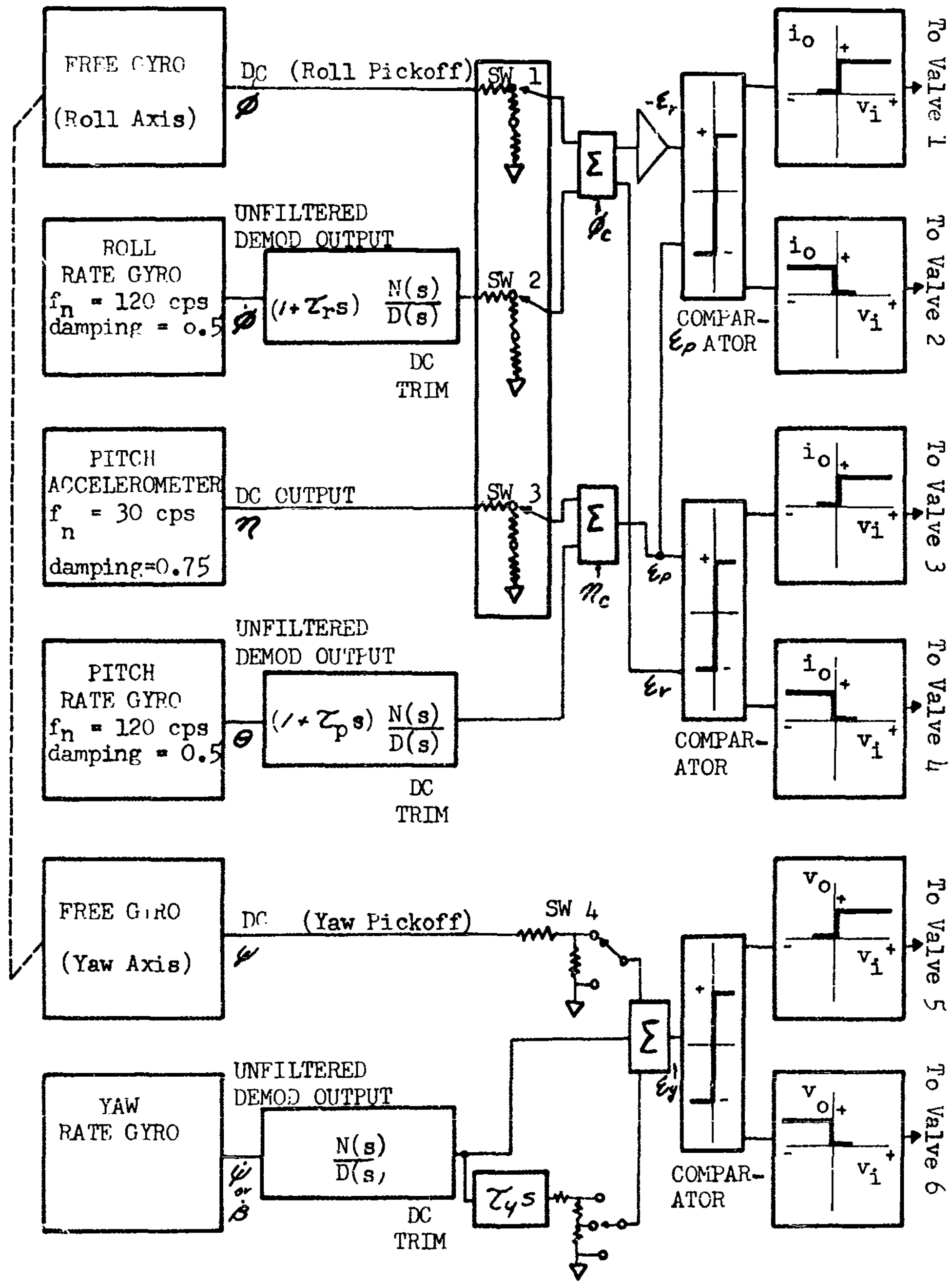


Figure 5.1 Autopilot Block Diagram

K = autopilot gain, and

τ = network time constant

Table 5.1 indicates the autopilot gains for the various control modes.

The yaw autopilot is separate and distinct from the roll-pitch autopilot and as such has two separate reaction motors for control in the yaw plane. The missile is aerodynamically stable in the yaw plane for small angles of sideslip and extremely underdamped. The yaw autopilot was designed for three possible modes of operation. Beginning with a crosswind launch, the heading angle (angle between the missile longitudinal axis and the desired direction of missile travel) is held to a minimum while the sideslip angle (angle between the velocity vector and the missile longitudinal axis) is decreased from 90 degrees to approximately 20 degrees. At this point the main propulsion unit of the Feasibility Test Vehicle is essentially "burned out", and the missile is allowed to align itself with the velocity vector (sideslip angle decreases to zero degrees). The launch perturbations are now corrected and the yaw autopilot functions to maintain the sideslip angle at a minimum. Initially, while the autopilot controls the heading angle (ψ), the control equation is,

$$\mathcal{E}_y = -K_7 \psi - K_8 \dot{\psi} (1 + \tau_y s)$$

where $\mathcal{E}_y$ = controlling error signal,

ψ = heading angle,

K = autopilot gains, and

τ_y = network time constant.

		SIDE LAUNCH			FORWARD LAUNCH	
		Launch Phase	Post Burnout		Launch Phase	Post Burnout
		$20° \leq \beta \leq 90°$	$-20° \leq \beta \leq 20°$	$\beta = 0$	$\beta = 0°$	$\beta = 0°$
Pitch Autopilot	K_1	5.0	2.0 →	→	5.0	2.0
	K_2	0.55 →	→	→	→	→
	τ_P	0.0015 →	→	→	→	→
Roll Autopilot	K_4	8.0	0.8 →	→	→	→
	K_5	0.375	0.08 →	→	→	→
	τ_R	0.005 →	→	→	→	→
Yaw Autopilot	K_7	10.0	0.00 →	→	→	→
	K_8	1.5 →	→	→	→	→
	τ_Y	0.01	0.10	0.00 →	→	→

Table 5.1 AUTOPILOT GAIN PARAMETERS

During the succeeding control modes, the control equation becomes,

$$\varepsilon_y = -K_B \dot{\beta} \left(1 + \tau_y s\right)$$

where $\dot{\beta}$ = sideslip angular rate. The yaw autopilot is also displayed in Figure 5.1 with the gain parameters listed in Table 5.1.

Since the aft launch power-on aerodynamics are not available, only a cursory study was conducted for control during this launch phase. If the jet demagnification is moderate during aft launch, there is no doubt that the missile can be controlled. The predominately low velocity environment increases the ratio of available control moment to aerodynamic moment. Control has been demonstrated for simulated flight with a zero launch velocity.

The autopilot parameters listed in Table 5.1 were verified on a time varying three-dimensional simulation. Boundary conditions for the initial launch phase of the Feasibility Test Vehicle were established. These initial boundary conditions are illustrated in Figure 5.2. The final study consisted of simulated programmed flights of the missile on the three-dimensional analog simulation. While, "flying" these trajectories, the missile performed normally. Two examples of the trajectory runs are shown in Figure 5.3 with the detailed missile operation during trajectory A of Figure 5.3 being presented on the computer strip recordings of Figures 5.4 and 5.5. Table 5.2 supplies additional information to simplify the interpretation of the computer recordings. Figure 5.3 illustrates a typical test trajectory in which no attempt was made to approach the maximum of maneuverability.

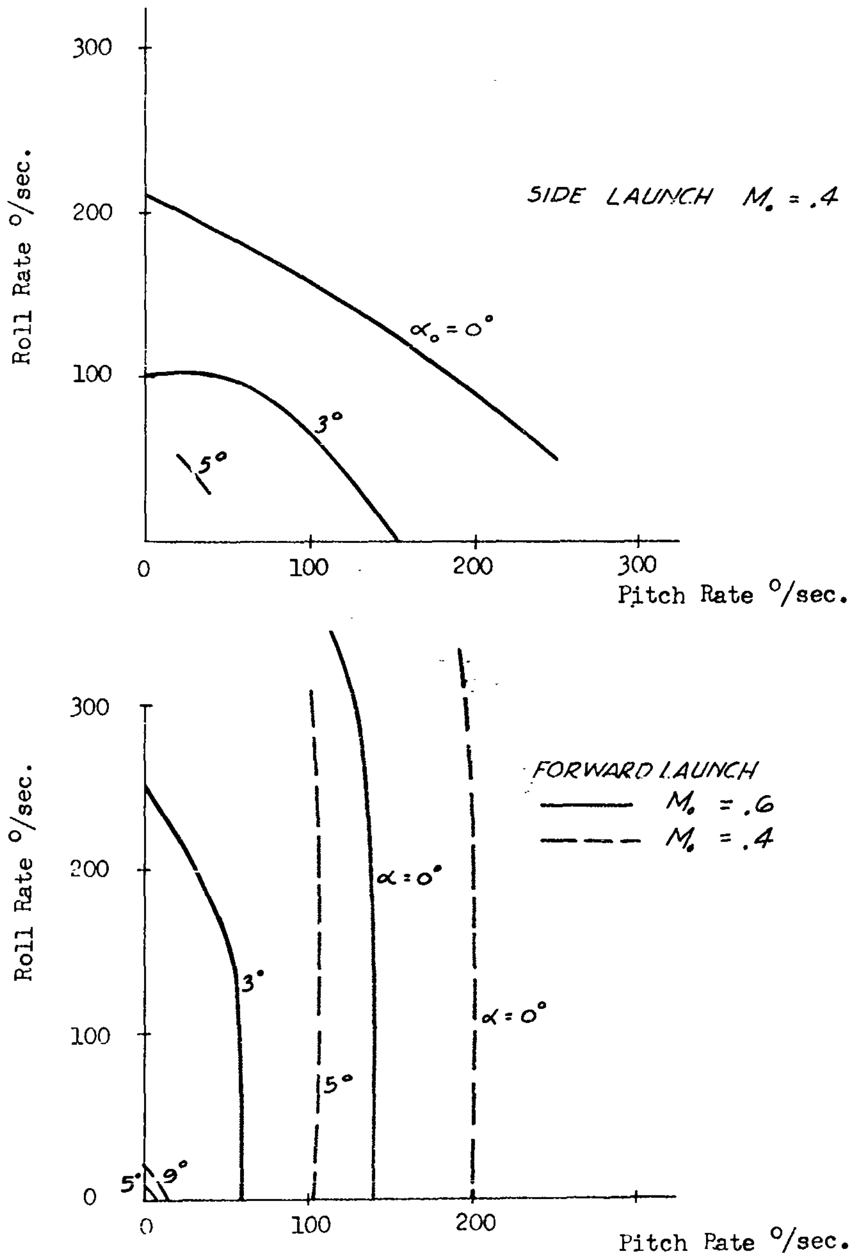


Figure 5.2 Initial Boundary Conditions

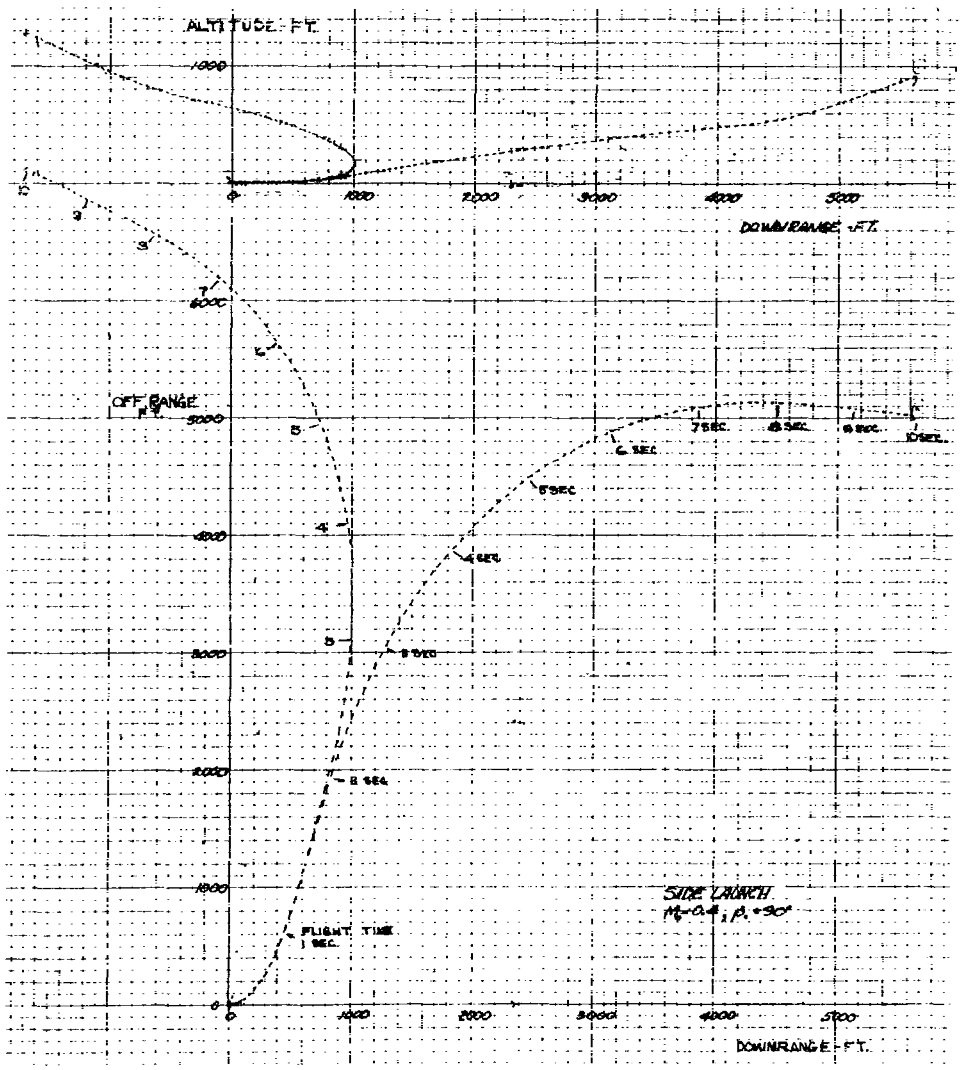


Figure 5.3 Side Launch Trajectories, $M_o = 0.4$, $\beta_o = 90°$

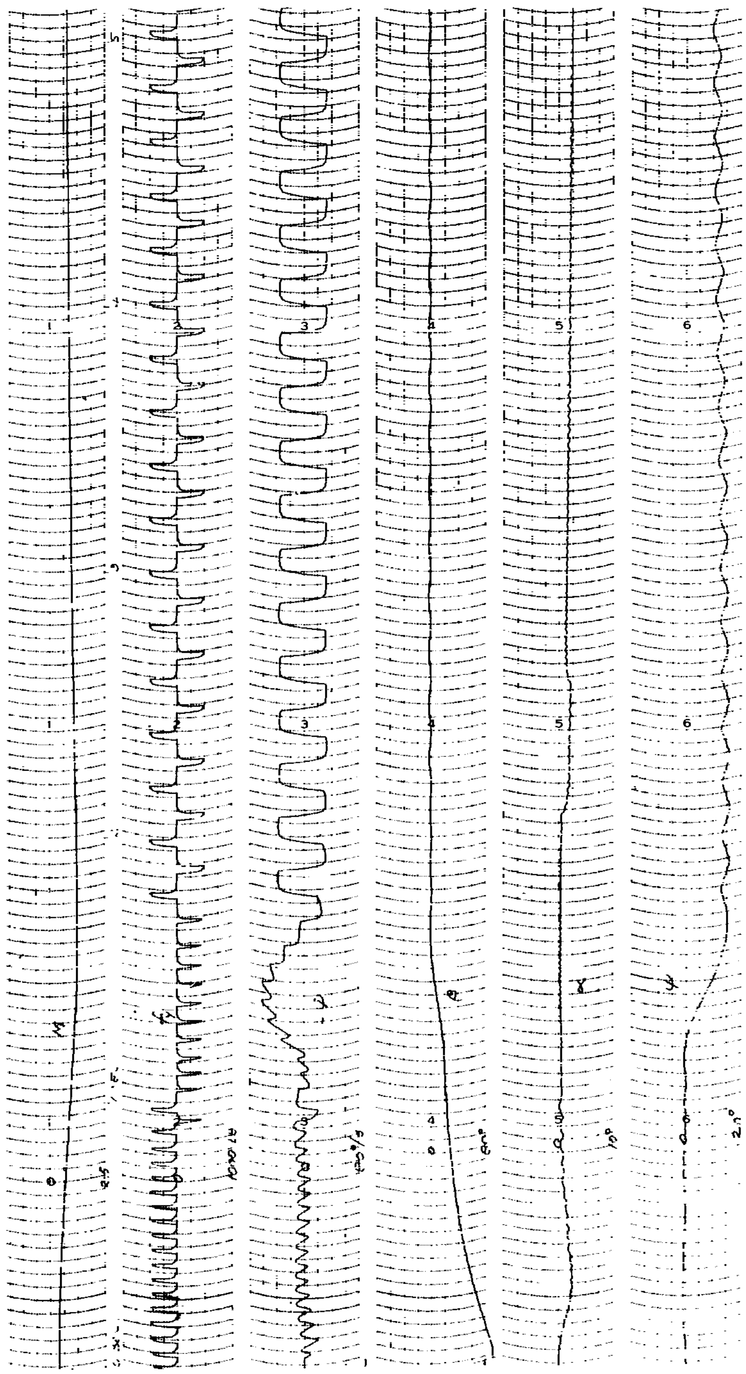


Figure 5.4a Missile Functions, Side Launch (0-5 Seconds)

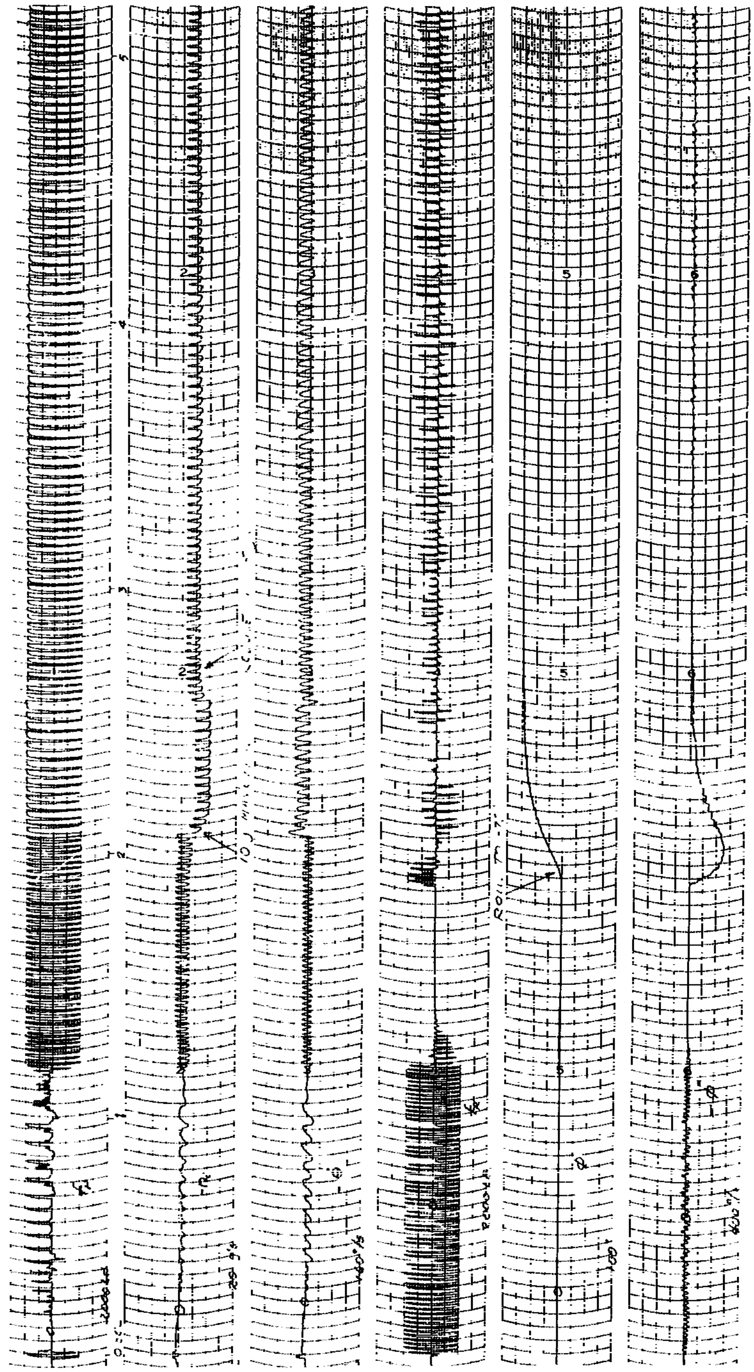


Figure 5.4b Additional Missile Functions, Side Launch (0-5 Seconds)

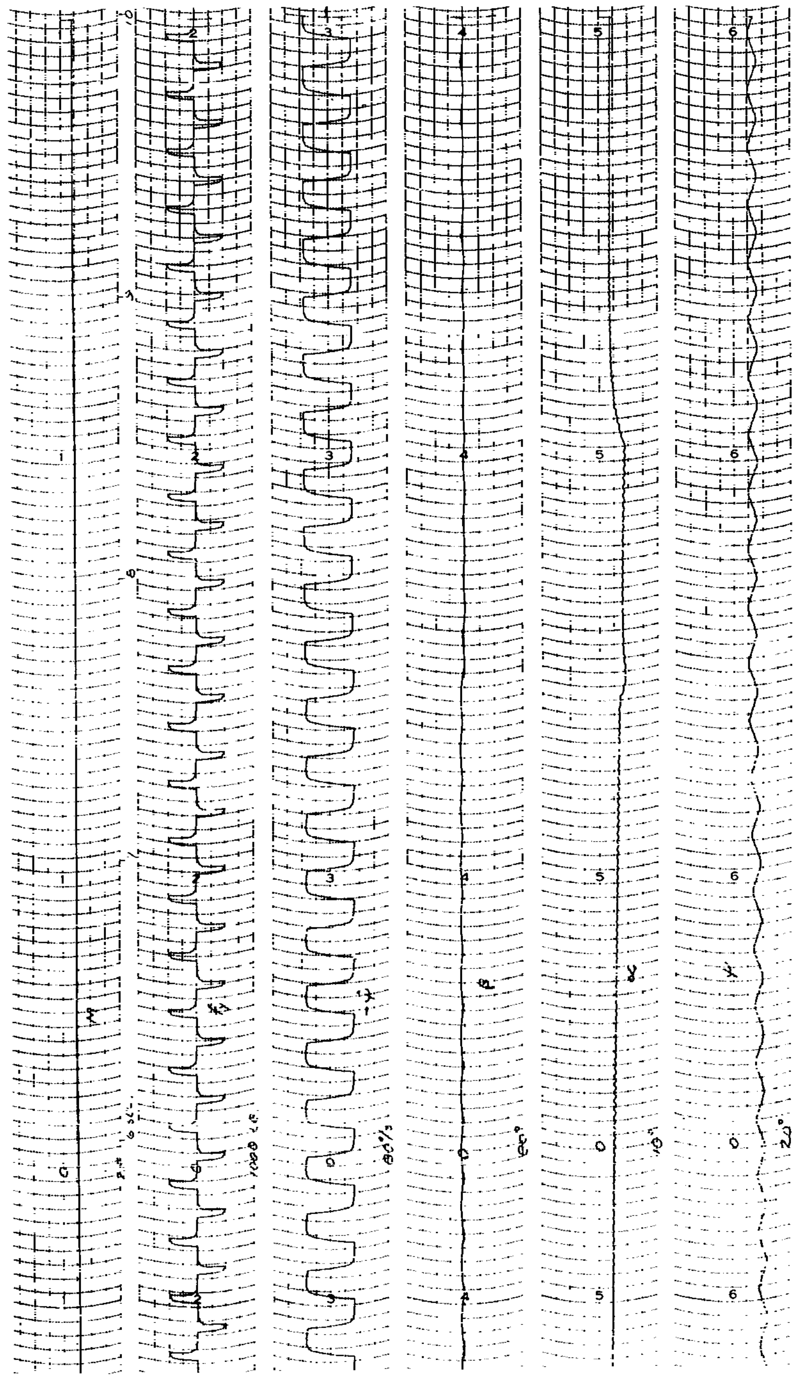


Figure 5.5a Missile Functions, Side Launch (5-10 Seconds)

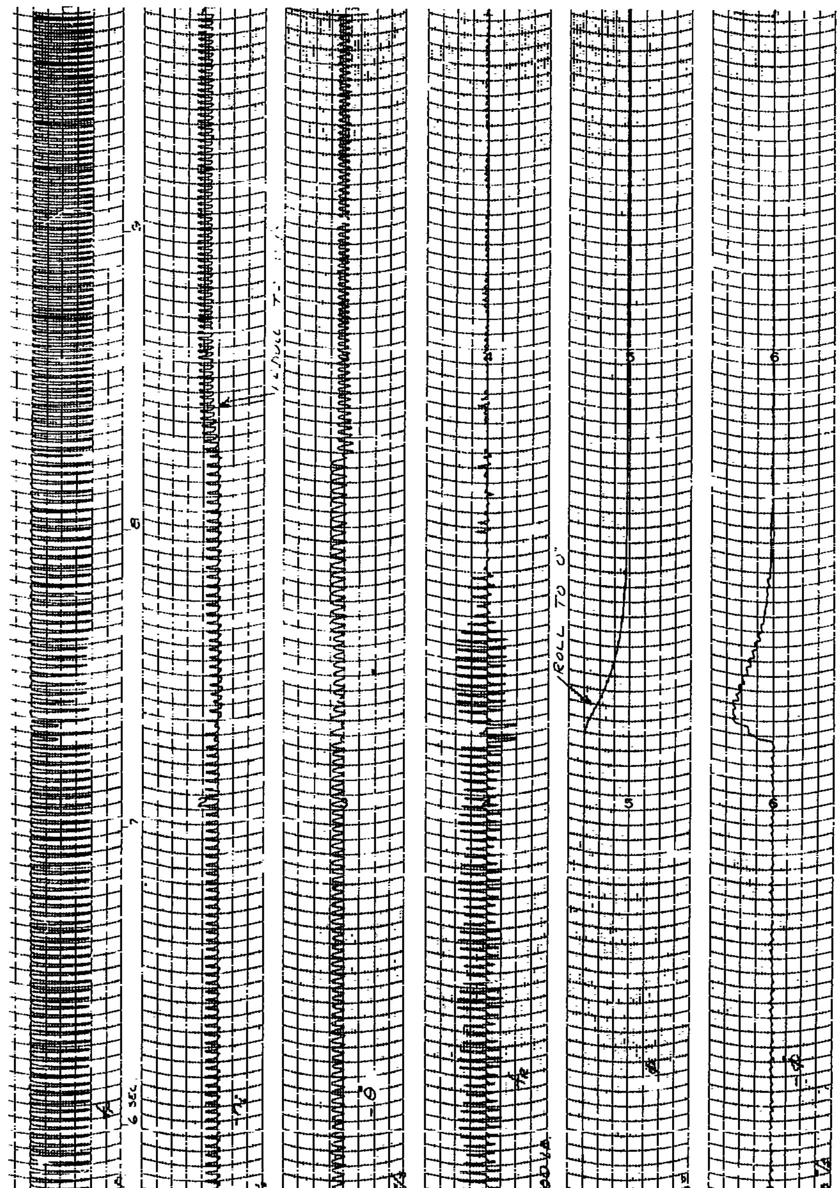


Figure 5.5b Additional Missile Functions Side Launch (5-10 Seconds)

For a detailed discussion and analysis of the autopilot studies, refer to Volume III, Section 2.3, of this report.

Section 6.0
PROTOTYPE CONSIDERATIONS

The primary effort in this study was directed specifically toward a Feasibility Test Vehicle application. The intended mission of this vehicle is a flight test from a high-speed rocket-sled to demonstrate the feasibility of the concept. However, much of the data obtained, both analytical and experimental, is directly applicable to weapons which could be developed from this basic concept. The wind tunnel results provide sound parametric data for configuration evaluation or analysis of the launch problem for any specific aircraft; the autopilot/control studies, likewise, provide the basis for proceeding with hardware development of this important subsystem.

Applications where this advanced concept will prove advantageous are primarily those where omnidirectional launch is required and/or high maneuverability at altitude is essential. Several specific missions immediately become apparent:

(1) Manned Vehicle Defense. The omnidirectional launch capability is particularly important for this application, even when considering a high performance aircraft such as the B-70. Missile attackers pose a real threat whether they are surface-launched or launched from an advanced interceptor (e.g. F-108 type). More advanced missile-launching platforms including very high speed rocket-powered interceptors (X-15 types),

may be expected in the future. To combat these threats, a defensive missile must possess not only omnidirectional launch capability but high maneuverability in order to be effective at high closing velocities. The PYE WACKET concept embodies these basic characteristics.

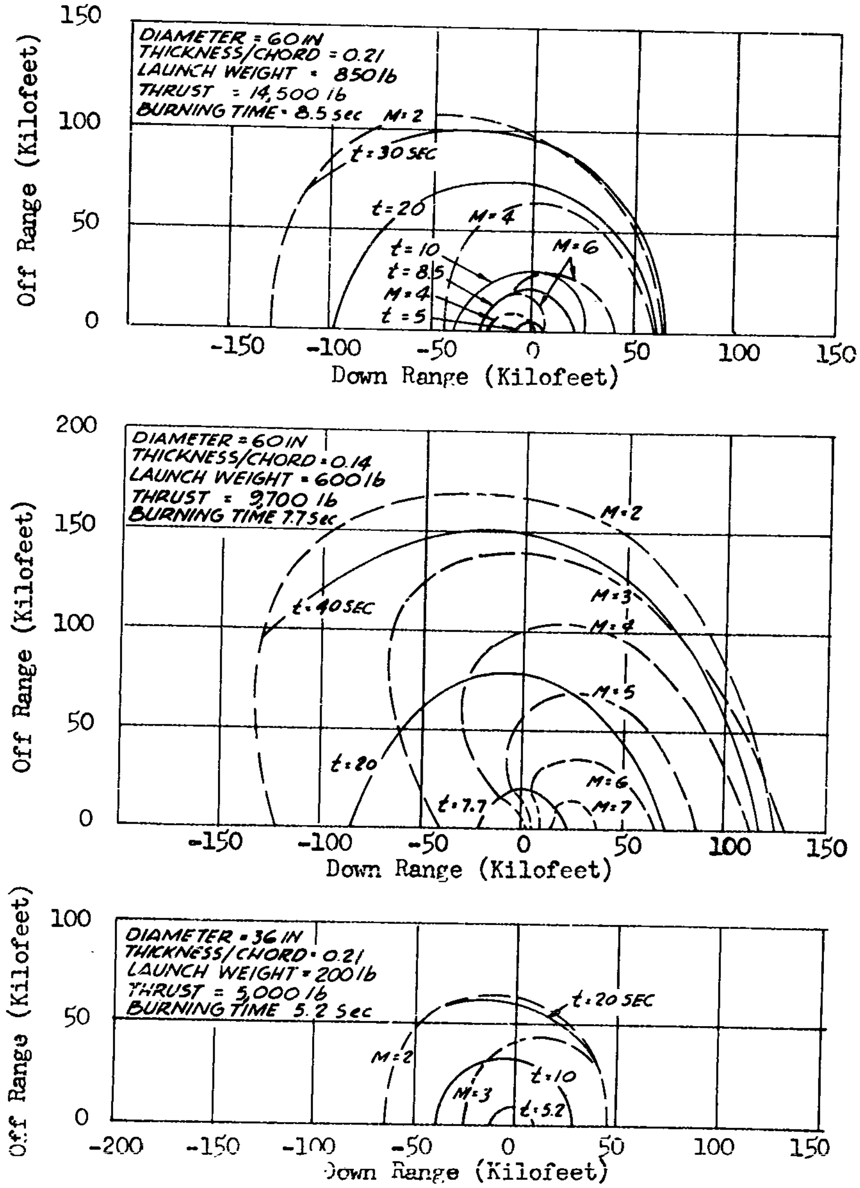
(2) Air-to-Surface Missions. The particular characteristic that "pays off" for this mission whether it be a high- or low-altitude penetration, is the omnidirectional launch capability. For "targets of opportunity" or those heavily defended so as to make direct overhead flights infeasible, the ability to deliver a warhead laterally to the target (offset bombing) is particularly attractive. An inertially guided PYE WACKET type of weapon receiving pre-launch input data from the airplane system may very satisfactorily accomplish this task.

(3) Controllable Reentry Vehicle. The circular planform, lifting-body concept is advantageous for reentry missions where controlled landings are required. The high drag of the bluff-body coupled with the lifting characteristics obtained at less than 90 degrees angle of attack permit adjustment or variation of the reentry trajectory. This capability provides a wider reentry corridor, and further, permits gliding flight to a pre-selected landing area. This concept is of particular interest for manned vehicles.

In the Phase I study, three "feasibility designs" for an air-to-air weapon were described: a 60-inch diameter, 21 percent thickness-to-chord ratio (t/c) vehicle, a 60-inch diameter 14 percent t/c configuration, and a 36-inch diameter 21 percent t/c missile. Performance envelopes for these missiles are presented on Figure 6.1. It should be noted that all distances are in terms of aircraft coordinates; i.e. down range or off range represents the distance between the missile and the launching aircraft at time of target intercept. Actual missile range, of course, is greater than the ranges shown for the forward launch case and less for the aft launch situation. Separation, or standoff, however, is approximately equal for forward and aft flights.

Figure 6.2 presents performance data for a 48-inch diameter, 21 percent t/c vehicle launched at Mach 3 at 70,000 ft altitude and Mach 0.8 at 40,000 ft altitude; this 315-lb missile employs a 50 lb warhead and utilizes cylindrical rocket motors manifolded to a single nozzle, a type of motor capable of development in a reasonable period of time. The separation distances of 80,000 ft or more are attained in all directions for the 70,000 ft altitude case and in excess of 30,000 ft for the 40,000 ft altitude launch. In both cases very substantial maneuver capability exists for the terminal Mach numbers shown.

The brief performance summaries presented in Figures 6.1 and 6.2 do not represent limits or bounds on vehicle capability; they merely serve to illustrate the characteristics of several different sizes and thicknesses. For any specific mission, a careful study is required to evaluate missile size, warhead type and yield, guidance accuracy, missile range or standoff, etc., since experience with conventional missiles has shown that invariably a tradeoff among these factors results.



Altitude = 60,000 ft
Launch Velocity = 2500 ft/sec

Figure 6.1 Performance Boundaries
(Aircraft-to-Missile Separation)

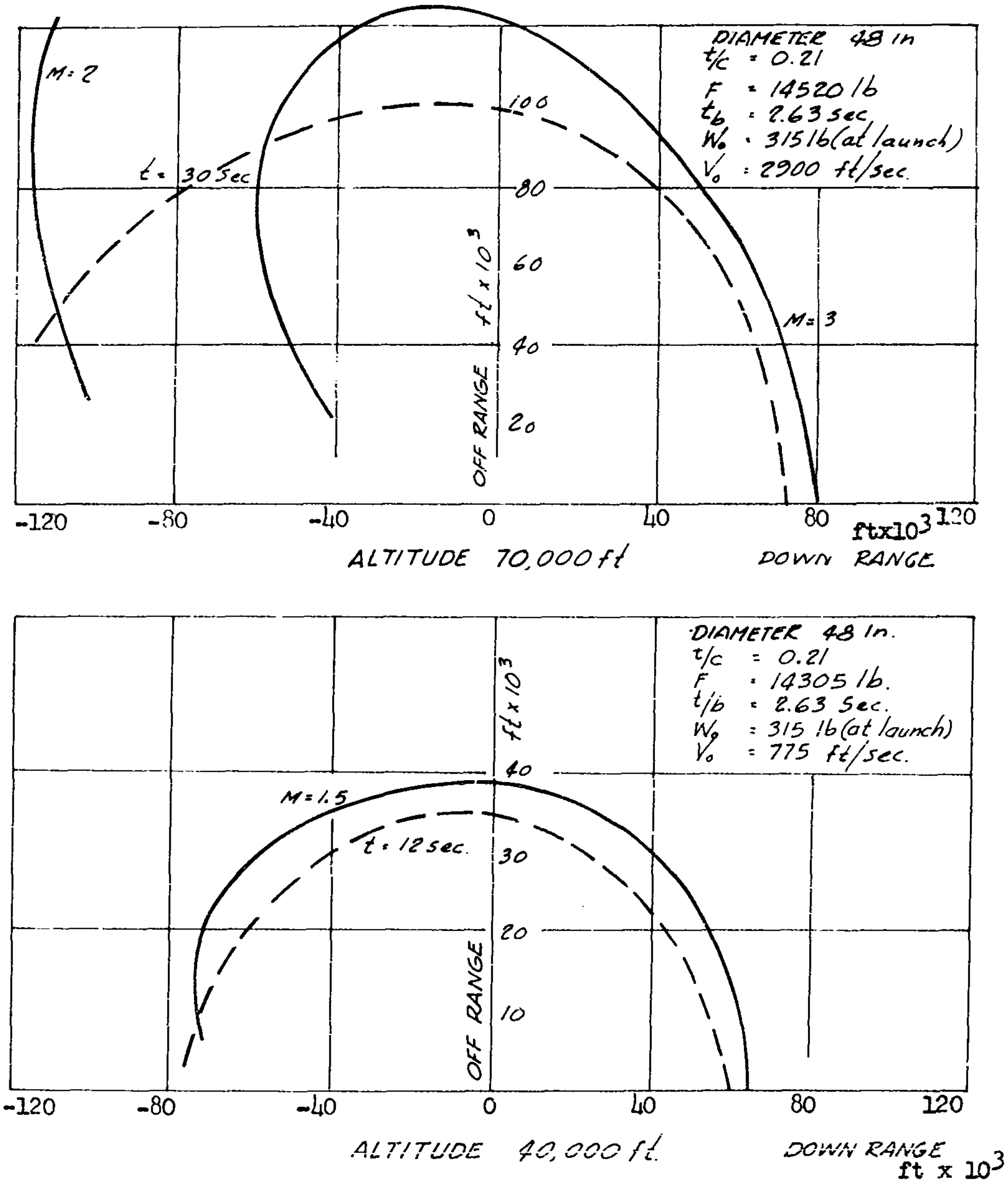


Figure 6.2 Performance Boundaries

(Aircraft-to-Missile Separation)

Section 7.0
RECOMMENDATIONS

The Phase II studies have greatly added to the knowledge concerning the circular-planform, lifting-body type of missile. The detailed analyses conducted in this phase, both aerodynamic and stability/control studies, provided further verification of the advantages of the circular planform, blunted lenticular configuration for certain missions, viz., missions where omnidirectional launch is required and/or where high altitude maneuverability is necessary. Some of the specific missions where the advanced lifting-body type of vehicle will show definite advantages are:

(a) manned vehicle defense,

(b) air-to-surface missions where offset bombing is required, and

(c) controllable reentry flights.

Recognizing the potential of this advanced concept for important future missions the following recommendations are made.

(1) A flight test program should be initiated immediately to demonstrate proof of the basic concept. This task (Phase III of the Feasibility Studies) involves fabrication of a number of flight vehicles and flight test of these missiles from a high speed sled (12 vehicles proposed -- reference Convair/Pomona Letter No. 11-8837/2217 dated 22 December 1960 to Directorate of Procurement, Eglin AFB, Florida). In addition to proof-of-concept, the flight

test program will provide the important data still lacking -- the dynamic effects of transients encountered at launch and in flight manuevers. It should be pointed out that all wind tunnel data were obtained under steady state conditions as is customary in such tests. A secondary benefit to be derived from the flight test program is state-of-the-art advancement -- not only in missile concept but in important subsystems. An example of this is the non-linear autopilot and high response reaction control system. This work will be applicable to most future vehicles operating at very high altitudes or in space.

The flight test program, the logical next step in realizing the advantages of this concept, comprises four major tasks:

(a) Development of the autopilot/reaction control system. The program todate has involved no hardware. The first six months of the Phase III effort would be concentrated on the necessary development and proof-testing of an autopilot and reaction control system.

(b) Detailed design of the Feasibility Test Vehicle. This task is primarily one of producing experimental fabrication drawings based on the Phase II configuration studies and the Phase III experimental program.

(c) Fabrication of the test vehicles and associated launching gear.

(d) Flight tests from a high speed sled (such as the Edwards Flight Test Center Facility). Flights from the sled would be made for forward, side, and aft-launch conditions. Photographic coverage would be obtained as well as data from the missile telemetering system and ground surveillance.

(2) Conduct of the flight test phase is strongly recommended since it is essential to a future weapon development, regardless of the mission to which this advanced concept is first applied. Only by conduct of such a program can the necessary data be obtained and real proof of feasibility demonstrated.

(3) A further recommendation is that a specific requirement be issued for a weapon which can exploit the inherent advantages of this advanced lifting-body concept. The preliminary work should be initiated while the Phase III task is in progress in order to expedite development of the weapon, and in turn advance its operational availability date.

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This report, consisting of three volumes contains the results of a series of studies conducted to form the basis for the design of PYE WACKET Feasibility Test Vehicles. Detailed studies (including wind tunnel tests), were conducted in the areas of aerodynamics, control system, structures and test vehicle performance. All studies were based on sea level flight environment. The complete task is reported in three volumes: Volume I - Summary, Volume II -- Aerodynamics, and Volume III -- Configuration and Autopilot/Control.

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