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A Pluto-Slam Design to Fit a Maximum Fineness Ratio Missile Into a Polaris Launch Tube

R. S. Cornwell

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M E M O R A N D U M --- October 12, 1962

TO: J. Hadley

FROM: R. S. Cornwell

SUBJECT: A Pluto-Slam Design to Fit a Maximum Fineness Ratio Missile Into a Polaris Launch Tube ~~(SRD)~~ e (U)

To fit a Pluto-Slam missile into a Polaris tube requires another look at the boost system. A storable liquid propellant rocket boost system can give a maximum fineness ratio to the missile and still allow full use to be made of all the available space in the tube. Figure 1 shows a proposed configuration for a 48 inch diameter reactor used in a 54 inch diameter missile weighing 30,000 pounds. The rocket motor shown develops 130,000 pounds of thrust at a chamber pressure of 1000 psi and burns for 60 seconds using inhibited red fuming nitric acid (IRFNA) as an oxidizer and JP-X (50% unsymmetrical dimethyl hydrazine, 50% JP jet fuel) as a fuel. (See Appendix I for calculations). It should be noted that the Navy has been using at least one storable liquid propellant missile aboard ship since 1960. The Bullpup, an air to surface missile, uses IRFNA (83.5 lbs) and MAF (Mixed Amine Fuel) (28.6 lbs) as propellants and is described in Bureau of Naval Weapons Notice 8023 of June 1960. These are stored on carriers and used on attack aircraft. Other storable liquid propellant systems in use are listed in Ref. 1.

Many configurations of tanks and rocket motors are certainly possible but only two are considered here. Aerodynamic and structural stability would strongly influence an optimum design. Possibly full airflow should be available through the reactor throughout the boost period.

Geometry of the first configuration considered here is to wrap the propellant tanks around the missile with a "chin" tank under the nose. The "chin" tank is ejected as soon as possible to allow limited air flow through the reactor. If full air flow through the reactor all the way through boost is considered necessary, this "chin" tank could be modified to a slipper tank for top and side of the nose. Wings are folded in the aft tanks and are carried "wet", that is, simply immersed in the fluid. Single fold wings are likely limited to a maximum area of 50 ft.^{2}. This area could possibly be increased with the use of full highly swept delta planforms or multiple folding.

The flight schedule would call for lighting off the rocket at water exit and simultaneously bringing up reactor temperature on a schedule giving about 1000°F at the side support. About a 40 second warmup is estimated to bring the reactor to full temperature. Location of the rocket motor in the nozzle allows some airflow at all times after the "chin" tank is blown off, which takes place about 30 seconds after launch. About 65% reactor air flow can then be maintained with the rocket motor in the nozzle. By shortening the vehicle and extending the rocket motor further aft, more airflow is obtainable.

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Figure 2 shows performance of the proposed missile based on a 3000°F wall temperature, which appears to be quite feasible when a dome supported reactor is used. Performance calculations were "by hand" but compare satisfactorily to the optimum ramjet of Ref. 2, which is also shown on Figure 2 for comparison. The drag curve is based on Ref. 3 and a comparison to other drag data (Ref. 4,5). Calculations are shown in Appendix II.

Shown on Figure 1 is a variation using hybrid rocket boost in which the fuel is carried as a solid and the oxidizer as a liquid. A shorter vehicle results, but no "chin" tank is necessary. The nozzle shown is a cooled metal structure using ceramic "sub-domes" and expanding through multiple nozzles. A great saving in length results but stability problems are increased due to the far aft center of gravity location. Also the question of losses due to wake drag from mixing of the multiple exhaust streams remains to be answered. Since the supersonic exhaust streams are parallel and at equal pressure, mixing should be limited. However, the basic design appears feasible and cooling air requirements for the metal nozzle parts are acceptable. Hybrid rockets are being developed but are not yet in the thrust range for this application. The problem of smooth burning in hybrids is still not completely solved and may pose a development problem.

A minimum size reactor makes this scheme of tube launching even more attractive. The increased performance due to higher wall temperature allows a smaller reactor and hence a smaller overall diameter. A 46 inch diameter reactor allows a 52 inch diameter missile and a corresponding decrease in the size of the "chin" tank, allowing flow through the reactor at an earlier time after launch.

Ref. (3) shows that a smaller vehicle than those originally contemplated is feasible even with a II-C reactor and becomes more feasible with a high wall temperature reactor. A short period of higher speed dash up to about $M = 3.8$ is indicated by Figure 2 (depending on the actual C_D). This period would be quite short as high stagnation temperatures and pressures would quickly damage the vehicle. However, the ability to change missile velocity by almost a whole mach number, even for short periods, would be a penetration aid as well as enhancing manuverability.

Weight estimates shown in Appendix III indicate that lower vehicle weights may be obtainable, but the weights used in calculations are considered conservative.

This missile could also be based on land and used by other services. One method would be to take the complete system including the tube and fire from inland water sites such as lakes and quarries, the water providing a cheap blast shelter. Land firing is also feasible using a small initial boost. By using a longer nose section and a slightly different guidance system, more weapons could be carried without building a completely new missile, giving even more flexibility to the system.

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RSC:ph

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2) Finke, R.G. et al. Sea Level Nuclear Ramjet Parameter Study, LRL, Livermore UCRL-5397 (SRD).

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APPENDIX I

Rocket Motor Sizing:

Boost flight requires that a final velocity of $M_o = 3(3360$ ft/sec) be obtained. Ref (a) develops the expression:

$$V_{final} = -V_j \ln \left(\frac{m_o - m_p}{m_o}\right) - \bar{g}\tau_p - q_1 \bar{C}_D \left(\frac{S}{m_o}\right) + V_o$$

V_j = effective jet velocity

m_o = original mass

m_p = propellant mass

$\bar{g}$ = average gravitational acceleration

τ_p = time of burning

$$q_1 = \int_0^{\tau_p} \left(\frac{\frac{1}{2}\rho V^2}{1 - y\frac{\tau}{\tau_p}} \right) d\tau$$

$\bar{C}_D$ = average drag coefficient

S = area

V_o = initial velocity (zero in this case)

V_j, the effective jet velocity is defined as $V_j = g\ I_{sp}$. Specific impulse for the IRFNA/JP-X system is 269 (Ref (b))

$$V_j = 32.2 \times 269 = 8660$$

Time of burning is assumed as: $\tau_p = 60$ seconds.

g is assumed for a $45°$ climb.

$g\tau_p = 32.2 \cdot \sin 45° = 1368$ ft/sec

The actual flight path would be a curved trajectory ending at some fairly low altitude like 15,000 feet.

The dynamic pressure term q_1 depends on altitude versus speed and time. Assume constant sea level density to give a conservative value. Assume a straight line velocity time curve.

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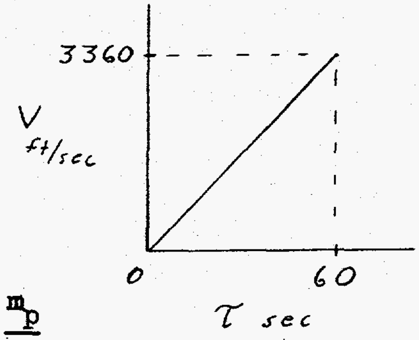


Then $V = \frac{3360}{60}\tau$

The value y is $\frac{m_p}{m_o}$

Use m_p = 29,200 lbs. and m_o = 59,200 lbs.

Then $y = \frac{29200}{59200} = 0.494$

Integrating the modified expression

$$q_1 = \frac{\rho}{2}\int_0^{\tau_p}\left(\frac{\left(\frac{3360}{6}\tau\right)^2}{1 - \frac{.494}{\tau_p}}\right)d\tau \qquad \text{to } \tau_p = 60 \text{ seconds}$$

gives $q_1 = 1.747\,\tau_p^{\,3}$

Surface area is derived later and is 17.7 ft^2

Solving for a value $\frac{m_o - m_p}{m_o}$ to give

V_{final} = 3360 ft/sec gives m_p = 29,200 lbs.

For 60 seconds firing time, thrust will be $F = \frac{29{,}200 \times 269}{60}$ = 131,000 lbs.

Volume required = 29,200/1.33 x 62.4 = 352 ft^3

For the IRFNA, JP-X system, the characteristic velocity (ref 7), C* is 5320.

$$C_f = \frac{V_j}{C^*} = \frac{8700}{5320} = 1.634$$

also $C_f = \frac{F}{P_c A_t}$ where p_c = chamber pressure

and A_t = throat area

$A_t = \frac{131{,}000}{1000 \times 1.634}$ = 80 in^2 (radius = 5.04 in.)

Using the curves of pg. 444 of (ref 6):

Ae/A_t = 8.2

Then A_e = 652 in^2 radius = 14.4 in.

For the $Cl\ F_3$ (Chlorine Trifluoride), $N_2\ H_4$ (Nitrogen tertroxide) system, the specific impulse if 294 and the density 1.51.

For the same thrust $m_p = \dfrac{60 \times 131{,}000}{294}$

$= 26{,}700$ lbs.

Volume required $= \dfrac{26{,}700}{1.51 \times 62.4} = 283\ ft^3$

This is 80.5% of the propellant storage volume of the IRFNA/JP-X system. However, ClF_3 has a boiling point of $52^{\circ}F$ and must be kept pressurized to extend the liquid range above this temperature.

APPENDIX II

RAMJET PERFORMANCE

Flight conditions:

$M_o = 3.0$

Altitude - sea level

Pressure - 14.7 psi

Temperature - $60^\circ F$

Scaling from Tory IIC for airflow

$$\frac{48^2}{54^2} \times 1800 = 1425 \text{ lbs/sec}$$

$$\text{Intake Area} = A_i = \dot{m}/\rho v = \frac{1400}{.002378 \times 32.2 \times 3360}$$

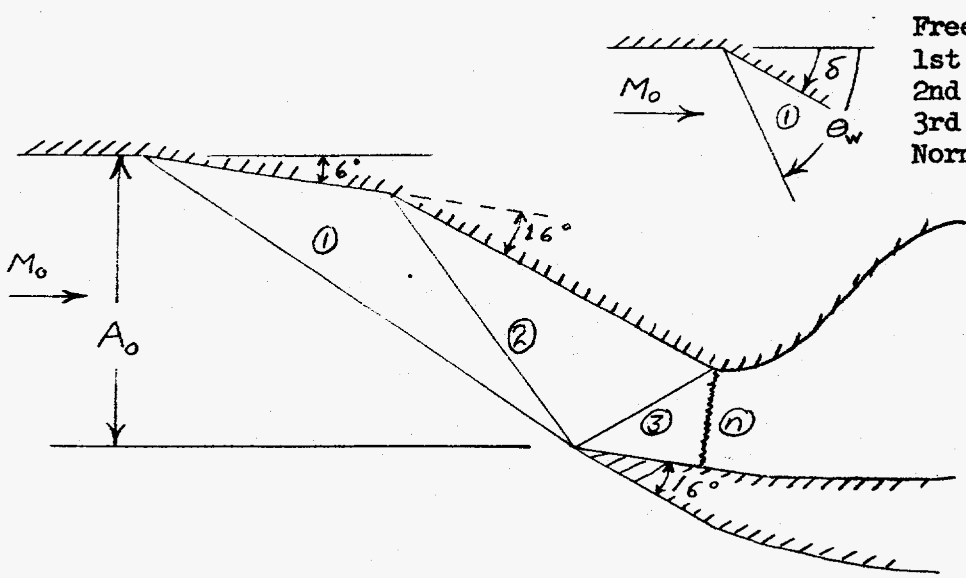
$$= 5.45 \text{ ft}^2$$

Reference Area = 14.72 ft^2 (52 in. dia. missile)

$$A_i/A_{ref} = 0.37$$

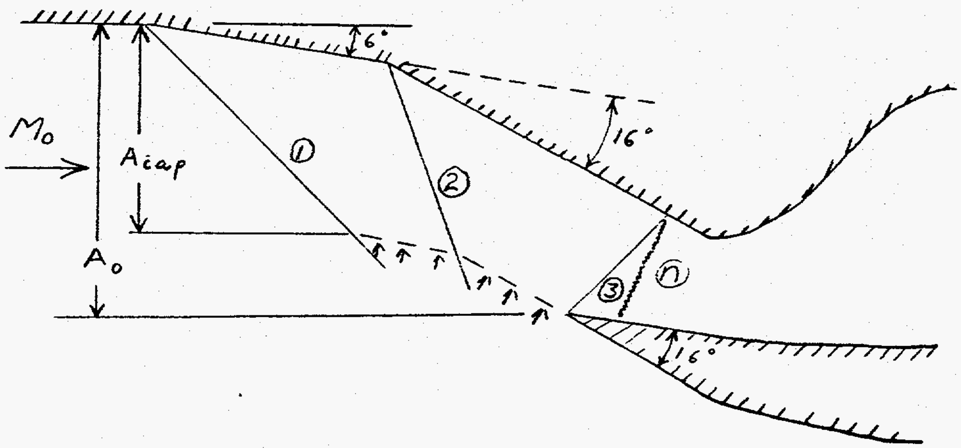
As a basis for calculation, a simple 3 oblique, 1 normal shock ramp inlet is used. This choice is purely aribtrary, but as will be shown later, it provides a handy method to estimate off design performance and compares well with actual and optimum inlets.

Geometry is as shown below.



Free stream $()_o$
1st shock (1)
2nd shock (2)
3rd shock (3)
Normal shock (n)

Off design flight will result in an altered shock pattern as shown:



Capture area A_{cap} can now be estimated at off-design conditions.

The total pressure recovery is computed from the expression:

$$\frac{P_{t_n}}{P_{t_o}} = \frac{P_n}{P_o}\left(\frac{1+\frac{\gamma-1}{2}M_n^2}{1+\frac{\gamma-1}{2}M_o^2}\right)^{\gamma/\gamma-1}$$

where the t subscript refers to total pressure.

Needed now is static pressure ratio across each shock. Using an oblique shock chart from ref 8, the following information is obtained:

δ	θ_w	Mach No. Behind Shock	Static press $p_{after\ shock}/p_{before\ shock}$
6^o	24^o	2.7	1.50
16^o	36^o	1.96	2.75
16^o	47^o	1.40	2.32
Normal Shock	90^o	.736	2.1

$$\frac{P_n}{P_o} = \frac{P_1}{P_o}\ \frac{P_2}{P_1}\ \frac{P_3}{P_2}\ \frac{P_n}{P_3}$$

Then
$$\frac{P_{t_{(n)}}}{P_{t_o}} = 1.5 \times 2.75 \times 2.32 \times 2.1\left(\frac{1+\frac{1.4-1}{2}(.736)^2}{1+\frac{1.4-1}{2}(3)^2}\right)^{\frac{1.4}{1.4-1}}$$

$= 0.785$

For $M_o = 2.5$ a similar calculation yields a pressure ratio of 0.824 and $A_{cap}/A_o = 0.834$. Ref. 6 gives a pressure recovery of about 0.84 for 3 oblique and 1 normal shock conical diffusers and about 0.75 for 2 oblique and 1 normal shock. Properly, a variable γ should be used to account for temperature changes. Throughout the remainder of these calculations, an average γ is used based on the temperature into and out of the applicable section being considered.

Reference 6 discusses subsonic diffusers and for the geometry of this section of the missile a pressure recovery of 0.8 appears reasonable and is used throughout these calculations. For $M_o = 3$ the pressure at the reactor entrance will be:

$$P_{t_{reat.}} = 0.785 \times 0.8 \times 14.7 \left(1 + \frac{1.4-1}{2}(3)^2\right)^{\frac{1.4}{.4}}$$
$$= 348 \text{ lbs/in}^2$$

Pressure drop through the reactor is taken as the same as IIC, giving 0.643 for the pressure ratio at $M_o = 3$. This ratio is assumed as 0.62 for $M_o = 3.5$, and 0.66 for $M_o = 2.5$. The nozzle coefficient is assumed is 0.98 (From IIC)

The stagnation temperature rise through the supersonic diffuser is:

$$T_{t_n} = T_{t_u}\left(1 + \frac{\gamma-1}{2} M_o^2\right)$$
$$= 520\left(1 + \frac{1.4-1}{2}(3)^2\right)$$
$$= 1456^\circ \text{ R}$$

$$\gamma_{avg} = \frac{\gamma 520 + \gamma 1456}{2} = \frac{1.4 + 1.35}{2} = 1.375$$

The corrected temperature is then:

$$T_{t_n} = 1410^\circ \text{ R; } (950^\circ\text{F})$$

No temperature rise is assumed to take place through the subsonic diffuser.

Ramjet thrust, F, is defined as:

$$F = \dot{m}(V_{inlet} - V_{exit}) + (P_{exit} - P_{ambient}) A_{exit}$$
$$- D_{additive} - D_{pressure} - D_{friction}$$

[redacted]

A drag coefficient for the entire vehicle will be used to replace $D_{pressure}$ and $D_{friction}$, so only additive drag will be considered for these calculations. Below design mach numbers will cause a contraction in the inlet stream tube as shown in the off design flight sketch of the inlet. Additive drag is then the axial component of the pressure on the inclined part of the stream.

For $M_o = 3.0$ and the temperature of the exit gas at 2600°F, the pressure at the nozzle entrance (reactor exit) is:

$$P = 0.643 \times 348 = 223 \text{ lbs/in}^2$$

Size the nozzle for expansion to stream pressure. Then:

$$\frac{P_{exit}}{P} = \frac{14.7}{224} = 0.0656$$

Sonic speed at 2600°F (3060°R)

$$a = 49.1 \sqrt{3060} = 2720$$

From Ref. 9 table 34 (One dimensional Isentropic Compressible - Flow Functions for $\gamma = 1.3$).

M^*	A/A^*	P/P_o	T/T_o
1.000	1.000	0.5457	0.8696
1.893	2.100	0.0656	0.5332

$$a^*/a_o = T^*/T_o = \sqrt{0.8696} = 0.932$$

$$a^* = 0.932 \times 2720 = 2535$$

$$v = M^* a^* C_n = 1.893 \times 2535 \times .98 = 4720 \text{ ft/sec}$$

$$F = \dot{m} \Delta V = \frac{1400 \text{ lbs/sec}}{32.2 \text{ ft/sec}^2} \times (4720 - 3350) = 59{,}600 \text{ lbs.}$$

$$C_f = \frac{F}{\frac{\rho}{2} V^2 A} = 0.303$$

The nozzle throat area is computed from the expression for weight flow in a nozzle:

$$\dot{m} = \frac{A p_o}{\sqrt{T_o}} \left(\frac{\gamma g}{R}\right)^{\frac{1}{2}} \left[\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{2(\gamma-1)}}\right]$$

Using $\gamma = 1.3$

$$A_t = 4.64 \text{ ft}^2 = A^*$$

To fully expand this flow; $A_{exit} = A^* \times 2.700 = 12.52 \text{ ft}^2$

Below design performance with a fixed nozzle geometry is restricted by the mass flow and the temperature to which this flow can be raised. Using the capture ratio of $A_{cap}/A_o = 0.834$, the pressure ratio of 0.824 for supersonic recovery, the pressure ratio of 0.8 for subsonic recovery and the pressure drop through the reactor of 0.66 results in $\dot{m} = 972$ lbs/sec and a pressure at the nozzle entrance of 110 lbs/in^2. The allowable temperature for these conditions is 1468°R.

$$a^* = 1752$$

$$V = 3330 \qquad \Delta V = 3330-2790 = 540$$

$$P_{exit} = .0639 \times 110 = 7.02$$

$$F = \frac{972 \times 540}{32.2} - (14.7-7.02) \times 144 \times 12.52 = 16300-13850$$

$$= 2450 \text{ lbs}$$

Additive drag at $M_o = 2.5$ is the pressure acting on the stream tube area projected in the axial direction. Using graphical means based on the off design sketch shown previously the additive drag is 6052 lbs. The total thrust at $M_o = 2.5$ is then

$$F = 2450-6052 = -3602$$

$$I_F = \frac{-3602}{\frac{\rho}{2}V^2 A} = -.0264$$

At $M_o = 3.5$ the capture area ratio is assumed to be 1 and the mass flow is then 1632 lbs/sec. An actual inlet would not allow this high a recovery due to a shock forming at the lip but this factor is not considered here.

[redacted] [redacted]

The temperature at the reactor inlet is 1715°R and the pressure 660 lbs/in². Using a reactor pressure ratio of 0.62, the pressure at the nozzle entrance is 369 lbs/in²

Then:

$a^* = 2535$

$V = 1.898 \times 2535 = 4810$

$V\ C_n = 4810 \times 0.98 = 4720$

$P_{exit} = .0639 \times 369 = 23.6$

$F = \dfrac{1632 \times 810}{32.2} + (23.6-14.7) \times 144 \times 12.52 =$

$41100 + 16050 = 57150$

$C_F = 0.216$

[redacted]

[redacted]

APPENDIX III

Vehicle Structure and Stores

The schematic vehicle outlined in Figure 1 has a net length of 30 feet in the liquid propellant launched version and 26-$\frac{1}{4}$ feet in the hybrid solid launched variation. The overall body diameter of both versions is 52 inches. The estimated flight weight of the vehicle is 27,500 lbs., $\pm$ 1,500 lbs.

<u>Vehicle Structure</u>

The fuselage structure is assumed to consist of three primary portions. The foreward most postion extending aft to the region of Fuselage Station 164 consists of the inlet plus weapons and guidance bays. The structure of this portion is assumed to be fabricated in a conventional skin-stringer-bulkhead fashion of precipitation hardening high temperature alloy and is unpressurized.

The mid section of the fuselage extending from approximately Fuselage Station 164 to Fuselage Station 316 is an integrally stiffened skin pressure vessel with fore and aft ring bulkheads.

This region contains the reactor power plant, inlet plenum, reactor control and coolant storage bays, and an integral radiation shadow shield. Structural weights quoted are based on the use of solid solution high temperature alloys.

The rearmost structural division consists of the exit nozzle, boat-tail fairing and lifting surface erection actuator bay. The construction is presumed to consist of a monolithic nozzle supplanted by a web supported fairing shell.

The lifting and stabilizing surfaces shown are a simple single fold design of arbitrary plan form. The half-span projection is on the order of 4 feet. The net area of a hypothetical delta configuration would approximate 45 square feet.

All structural estimates were based on 0.1% strain in 10 hours at 1000°F material strength extrapolated from published data. The design safety factor used was 1.25.

Reactor

The reactor configurations shown are rear supported designs requiring no internal metal structure. The primary configuration makes use of a "dome" like axial support of self bonded silicon carbide such as have recently been fabricated and tested by LRL.

The auxilliary configuration shown makes use of an internally air cooled metal support structure incorporating multiple full expansion nozzles. In each instance, the reactor is capable of being separated completely from the vehicle systems and structure. The sole mechanical attachments occur at the axial support structure seat and foreward shear joint. The reactor weights detailed below are based on a 46-inch overall ceramic matrix diameter, 40-inch fueled matrix diameter, silicon carbide "dome" supported BeO core assembly.

Systems, Auxilliaries and Stores

1. Weapons: Optional configurations range from a single 32-inch diameter warhead or pair of 21-inch diameter warheads to as many as six 15-inch diameter ejectable weapons. The net volume of the bay is 45-cubic feet. In each variant the total yield should be on the order of 10 megatons.

2. Shielding: Provision is made for mass attenuation of direct beam radiation seen by the main guidance and weapons bays. Additional provisions for scatter shielding may be necessary but was not considered in this treatment.

3. Guidance: The foreward guidance bay is assumed to house antennae and other receptors along with the canard control surface actuators. The net available volume is about 6 cu ft. The main (aft) guidance bay has a volume of about 18 cu ft and is presumed to house all active electronic and radiation sensitive control systems.

4. Reactor Control Actuators: Radiation resistant control rod servo and "scram" actuators are housed in a nacelle projecting into the inlet plenum.

5. Coolant: Provision for an evaporative or mechanical coolant system is based on a presumed requirement of the guidance and/or weapons systems and is arbitrarily sized at 5 cu ft.

Weight Summary

The following brief summary of weights is derived from calculations based on the foregoing considerations. The numbers quoted are intended to be conservative, i.e. a little on the high side.

Weight Summary (Continued)

1. Reactor:

Ceramic matrix, 46" dia. - 60" long	5,750 lb	
Axial support structure, SiC dome	800	
Front preload structure & fittings	700	
Side support structure & fittings	400	
Integral control hardware	300	
Dome seat support structure	200	
		8,150 lb

2. Air Frame:

Nose & inlet to fuse sta. 164	2,900 lb	
Main fuselage - F.S. 164 to F.S. 316	5,800	
Nozzle & boat-tail-aft of F.S. 316	1,300	
Wing and erecting mechanism	1,500	
		11,500 lb

3. Other:

Weapons stores	3000 - 4000	
Shadow shield	2000 - 4000	
Fwd. guidance bay	200	
Aft guidance bay	500	
Reactor controls	300	
Coolant system	300 - 500	
		6,300 - 9,500

Total Flight Weight 25,950 -29,150 lb

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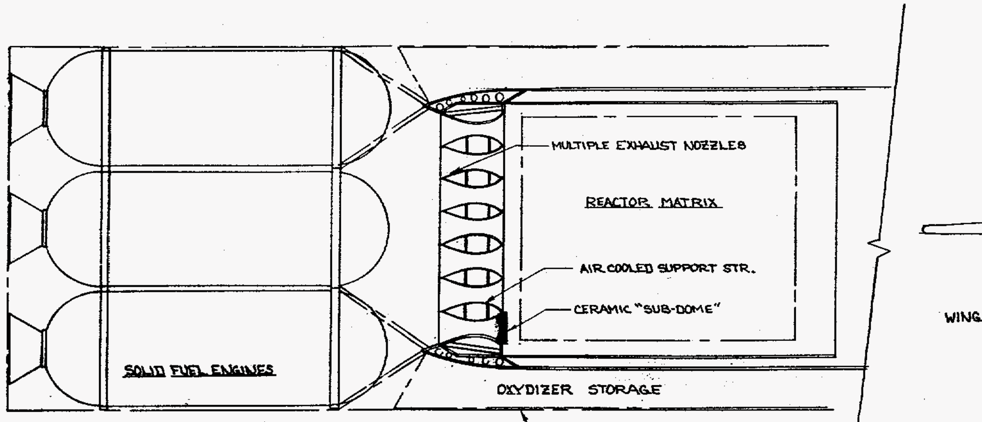
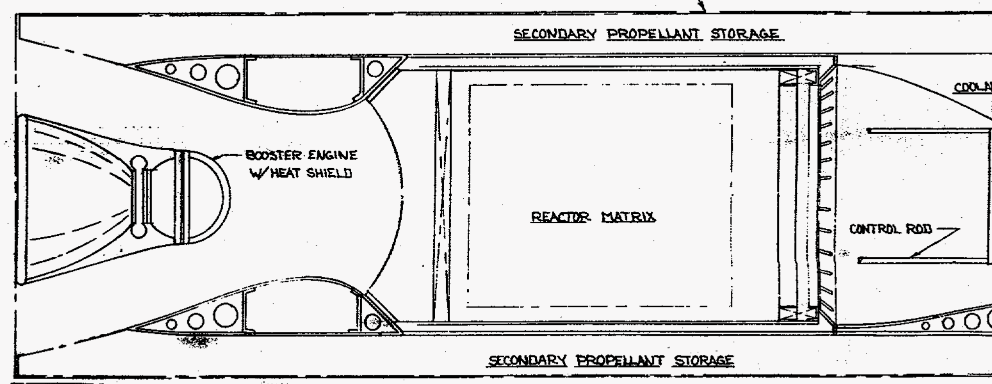


PROPOSED MULTIPLE NOZZLE VEHICLE CONFIGURATION WITH HYBRID SOLID BOOSTER

F.B.M. NUCLEAR SUBMARINE LAUNCH TUBE PROFILE



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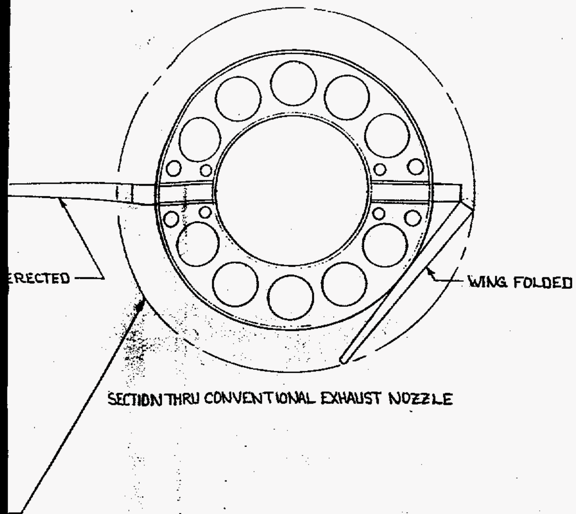
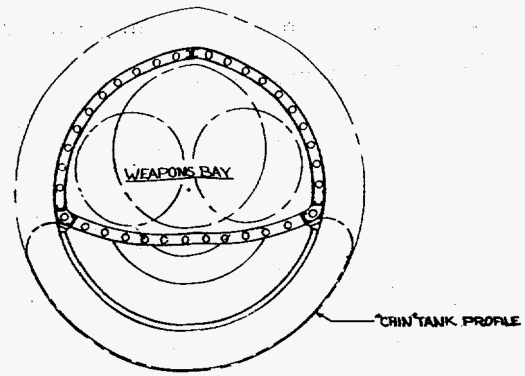
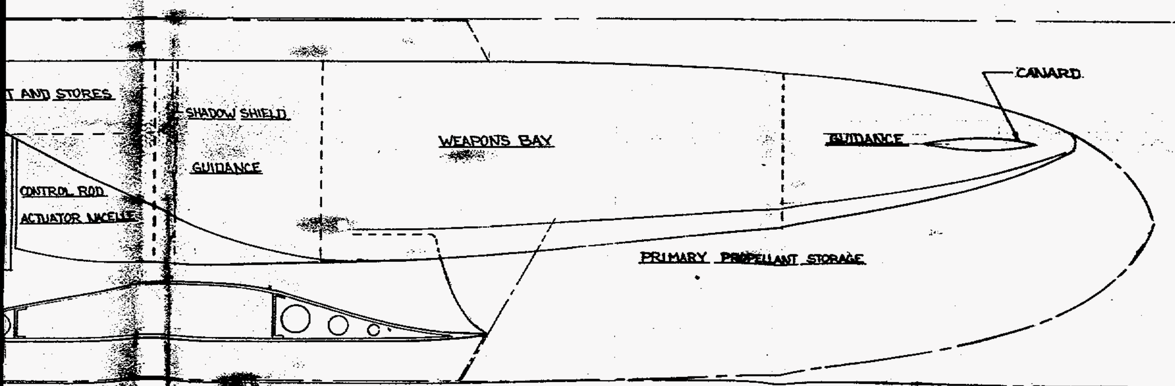


SECTION THRU CONVENTIONAL EXHAUST NOZZLE



SECTION THRU INLET



AMJET VEHICLE

[redacted]

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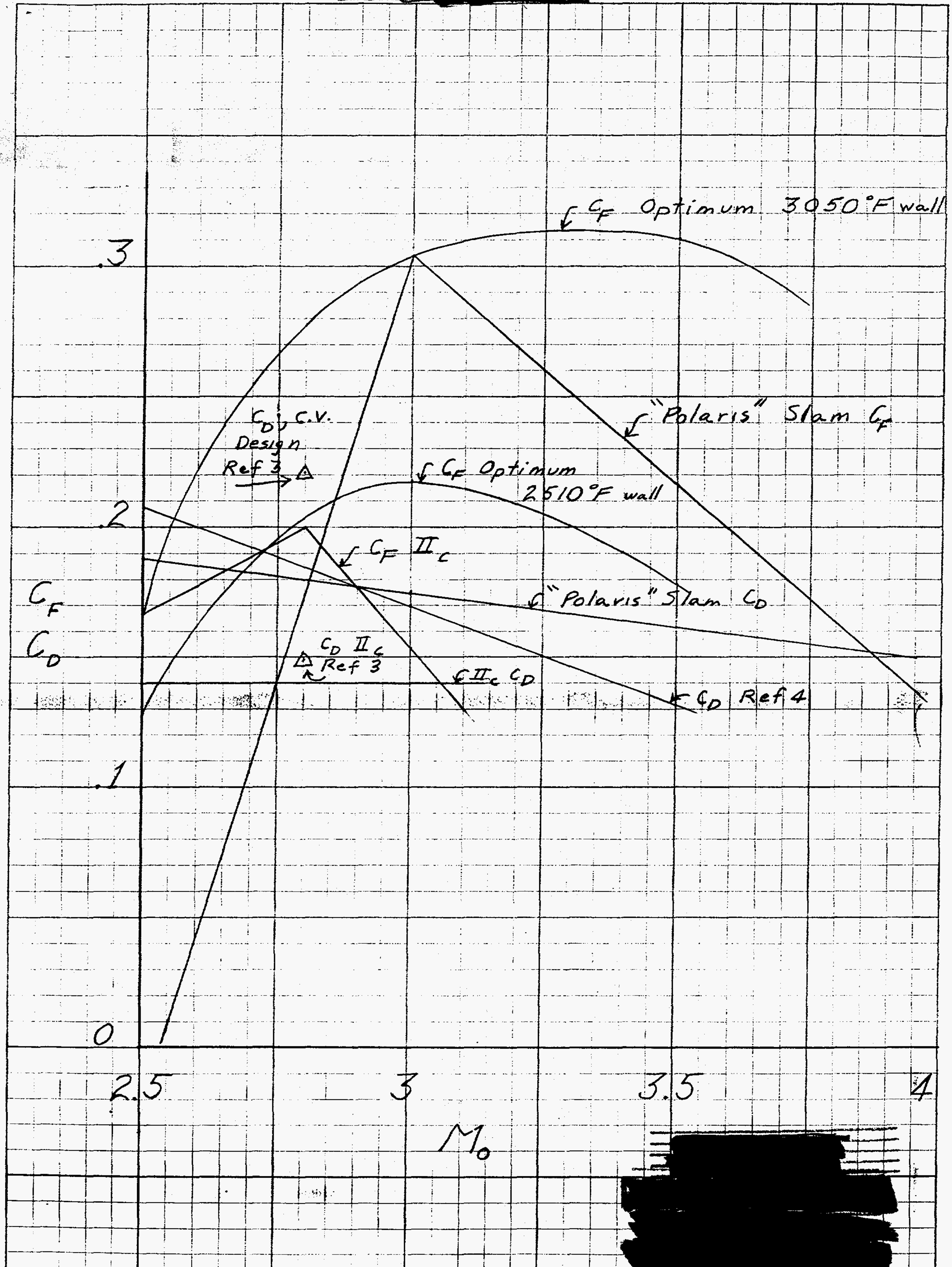

[redacted]

FIGURE 2

[redacted]

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