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Performance Potential and Research Needs of a Hypersonic, Airbreathing, Lifting Missile Concept

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This paper reports the results of a conceptual study to define the performance potential, scaling relationships, and research needs of a hypersonic missile concept formulated about an airframe-integrated propulsion system. A baseline missile design is developed and the rationale for selecting a heat-sink structural approach is described. Modifications to a NASA-Langley-developed fixed-geometry, modular scramjet are presented that may be required to achieve the dual-mode, "hydrocarbon-fueled" engine operational characteristics utilized in predicting missile performance potential. A heat-sink structure is shown to be a viable concept for Mach 6 missiles for ranges of up to 200 n.mi. A heat-sink structure protected by multiwall heat shields is suggested as a potentially attractive system for long-range cruise missiles. Airframe-integrated propulsion systems are shown to offer potentially large benefits for missiles that require high maneuverability. The use of the missile forebody for generating inlet precompression as well as lift and use of the afterbody as a high-expansion ratio, low-drag nozzle also offer improved missile cruise performance. It is concluded that technologies being developed for hypersonic aircraft, with appropriate modifications, also can be useful for hypersonic, airbreathing, lifting missiles.

Nomenclature

A	= area
C_D	= drag coefficient
g	= gravitational constant
I_{sp}	= specific impulse
L/D	= lift-to-drag ratio
l/d	= length-to-diameter ratio
M	= freestream Mach number
q	= freestream dynamic pressure
t	= time
TPS	= thermal protection system
V	= velocity
X	= horizontal distance
Y	= offset distance
α	= angle of attack
ϕ	= fuel equivalence ratio
ρ	= density

Subscripts

c	= inlet cowl
t	= trim
2	= first minimum

Introduction

IN recent years, hypersonic research programs at Langley have contributed to the development of a hydrogen-fueled modular scramjet,[1] actively-cooled structures,[2] transport technology,[3] airbreathing launch vehicle concepts,[4,5] and research airplane concepts that utilized heat-sink structures.[6,7] This research has resulted in a growing capability in vehicle synthesis along with an increasing awareness of applicability. Of interest in this progression is the applicability of these technologies to hypersonic, airbreathing missiles.

Missiles are relatively small, volume-constrained vehicles which do not lend themselves, logistically or volumetrically, to the use of hydrogen fuel, but rather to noncryogenic fuels having densities more than an order of magnitude greater than hydrogen. Clearly, this switch from hydrogen to "hydrocarbon" fuels will have an impact on the details of the needed technology; however, there are areas—including propulsion, propulsion integration, configuration synthesis, thermal protection, and structures—where portions of Langley's present hypersonic technology program could possibly contribute to a technology base for the development of hypersonic, airbreathing missiles.

The purpose of this study was to define an example baseline concept for a hypersonic missile having a modular airframe-integrated, dual-mode scramjet propulsion system, to determine the potential performance and scaling relationships, and to identify potential areas for further research. In order to specify weight and determine the magnitude of the technological obstacles, it was necessary both to define a structural approach for the engine, airframe, and radome, and to assess the thermostructural limitations. Operating Mach numbers from 4 to 9 were investigated; however, the focus here is from Mach 4 to 7. Concepts with ranges from 20 to 3500 n.mi. and payloads from 33 to 227 kg were considered.

Baseline Concept

The baseline concept was formulated about a simple shell, heat-sink structure which lends itself to range/size tradeoffs. A length of 3.7 m and effective diameter of 0.386 m was selected as representative of a reasonable missile size (Phoenix class); however, missile compatibility with various launch aircraft was not assessed. A modular, airframe-integrated, dual-mode scramjet propulsion system using an alkylated borane as fuel was assumed. A 45 kg payload was assigned. The missile was assumed to be boosted by a solid propellant rocket from the launch condition at Mach 0.8 and 12 km altitude to Mach 4, at constant altitude where staging occurred. The vehicle was to accelerate on the airbreathing propulsion system from staging to Mach 6 at 31 km altitude, where the fuel-flow rate was reduced for cruise. These

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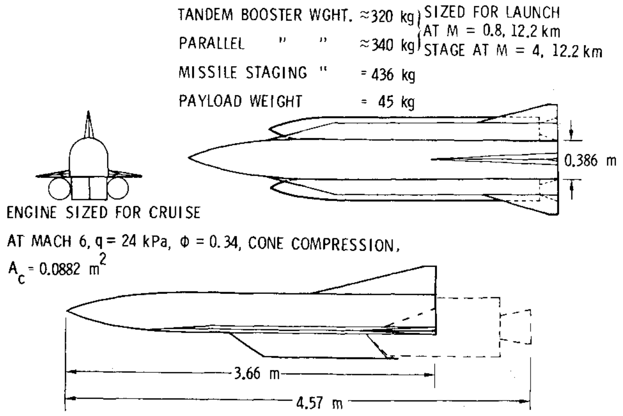

Fig. 1 Baseline configuration.

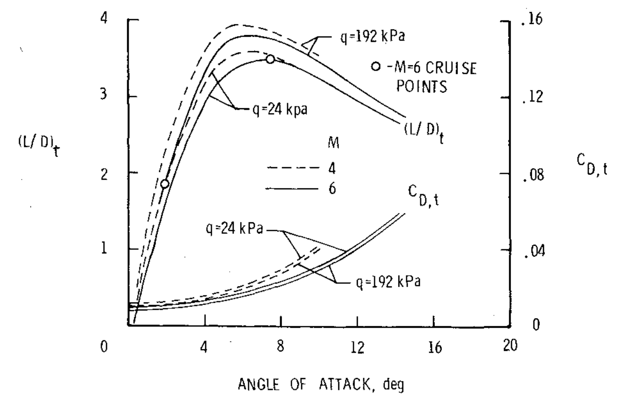

Fig. 2 Aerodynamics of the baseline missile; reference area = 2.31 m².

guidelines and assumptions resulted in the conceptual baseline vehicle configuration shown in Fig. 1, for which most performance evaluations were conducted.

Configuration/Aerodynamics

The baseline configuration (Fig. 1) was designed to cruise at maximum L/D at Mach 6 at 31 km altitude ($q = 24$ kPa). The low-aspect-ratio wings with short, swept leading edges are structurally efficient and have relatively low heat loads. The fuselage, which has a fineness ratio of 9.5, incorporates a simple ogival nose shape attached to a constant cross-section shell structure. The flat lower forebody provides a precompressed inlet flow to the dual-mode scramjet modules nested side-by-side on the lower mid-fuselage. The low wing placement also contributes to the precompression and uniformity of the inlet flow as well as providing additional space for the engine installation. The lower afterbody provides a high-expansion-ratio, low-drag nozzle for the scramjet propulsion system. Longitudinal and lateral aerodynamic control was provided by triangular, all-flying tail surfaces.

Trimmed aerodynamics of the baseline configuration without engine modules were calculated using the panel methods described in Refs. 8 and 9. Skin friction and aerodynamic heating were calculated assuming an all turbulent boundary layer. Trimmed L/D and drag coefficient are presented in Fig. 2 as a function of angle of attack, for Mach 4 and 6 cruise at dynamic pressures of 24 and 192 kPa. These aerodynamic characteristics without the engine modules attached were used in all performance calculations, inasmuch as Ref. 10 indicates that careful integration of the scramjet with the airframe results in minimal elevon deflections and trim penalties for both power-on and power-off flight conditions at Mach 5, 6, and 7. Thus, for the scope of this study, it was assumed that the axial location of the engine modules, the cowl length, and the nozzle angle could be adjusted to avoid engine/airframe integration trim penalties.

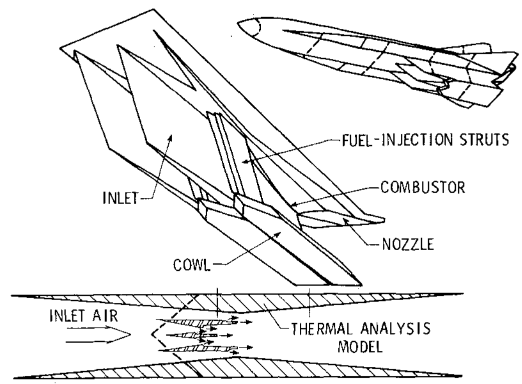

Fig. 3 Airframe-integrated, modular scramjet.

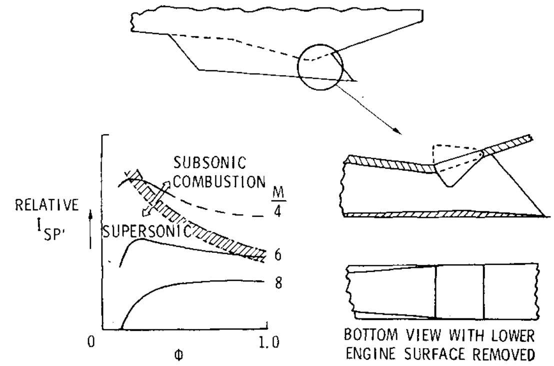

Fig. 4 Dual-mode scramjet concept and performance.

Propulsion

The Langley modular scramjet[1] illustrated in Fig. 3, although developed for use with hydrogen fuel, has some features in addition to the advantages afforded by airframe integration[11] which are attractive for "hydrocarbon-fueled" missile applications. The fixed-geometry inlet has demonstrated excellent starting, mass capture, and pressure recovery characteristics for flight speeds from Mach 3 to Mach 8.[12] Swept fuel injection struts enhance the precompression and allow a combination of normal and parallel fuel injection to control the mixing and heat release, which provides efficient operation over a wide Mach number range with low heat transfer to the combustor walls.[13] Recent thrust measurements[14] on a module (20.3 cm by 16.3 cm inlet capture area) at Mach 4 and 7 lend credence to the potential performance predicted for this engine.

Although the Langley fixed-geometry engine has some attractive characteristics, the present study indicated that for the best overall hypersonic missile performance a dual-mode engine is needed. Such an engine could provide efficient operation with subsonic combustion at a high equivalence ratio for acceleration from staging at Mach 3 or 4 and convert to supersonic combustion between Mach 5 and 6 for further acceleration and/or cruise. To investigate this type of dual-mode operation for the Langley scramjet, subsonic combustion cycle analyses were performed[15] at Mach 4 flight conditions assuming that the fuel was injected from the wall downstream of the normal shock. (The Langley engine has an experimentally measured inlet contraction ratio of 5.7 at Mach 4, and a combustor-exit to combustor-entrance effective area ratio of approximately 3.7.) These analyses indicated that a variable-geometry second minimum which would force the normal shock in the combustor toward the first minimum would substantially improve performance in the subsonic combustion mode.

This study suggests that an attractive hypersonic propulsion system for a missile would be a modular, dual-mode scramjet incorporating a fixed-geometry inlet, a combustor with a gradually increasing expansion rate culminating in a combustor-exit to combustor-inlet area ratio greater than about 2.5, a variable- or two-position second minimum, and a combination of strut(s) and wall fuel injection.[16] These features should provide efficient subsonic combustion performance at high equivalence ratios for acceleration from Mach 3 or 4 to about Mach 5.5. (A rearward-facing step[14] and/or distributed fuel injection[17] may be required to prevent combustion effects from feeding forward into the inlet at equivalence ratios greater than about 0.5.) In the vicinity of Mach 5.5, the combustor exit could be enlarged, with a simple pivoting device or by removal of an insert, to provide for efficient supersonic combustion. An alternate attractive possibility for achieving efficient subsonic combustion performance is the use of distributed, flush-wall fuel injection to provide a controlled thermal choke in the downstream region of the combustor. This capability has been demonstrated[17] in experimental tests at Mach 3 and 5 with hydrogen.

The fuel selected for use in this study was an alkylated-borane, HiCal 3-D, which is storable, but toxic and expensive. This fuel was chosen because its nonpiloted combustion has been demonstrated in free-jet tests of a dual-mode scramjet[18] at Mach 5.0, 5.8, and 7.0, and because its effective heat-of-combustion is 10 to 15% higher than conventional hydrocarbon fuels at Mach 6 flight conditions. It is recognized that conventional hydrocarbon fuels such as JP-4 and JP-7 or the higher density ones such as Shelldyne may be more appropriate for hypersonic missile applications. The supersonic combustion of these fuels has been experimentally investigated[16,19] using a hypergolic oxidizer as an ignitor and pilot. The specific impulse realized from such fuels may be somewhat less than with HiCal 3-D; however, the higher density afforded by, say, Shelldyne may be a compensating factor in terms of cruise range.

The engine performance characteristics used in this study (Fig. 4) were based on specific impulse estimates for a freestream inlet, annular nozzle, dual-mode scramjet concept,[18] fueled with HiCal 3-D injected perpendicularly from the wall. These performance characteristics are consistent with the results of Ref. 20, assuming a combustion efficiency of 0.94 at $\phi = 1.0$ and 0.99 at $\phi = 0.4$. At Mach 6, the specific impulse ranged from 900 s to 1200 s depending on equivalence ratio, altitude, etc. The specific impulse values for subsonic combustion at Mach 4 were extrapolated (dashed line in Fig. 4) to an equivalence ratio of one, since engine operational constraints are believed to be relaxed by the use of a variable-geometry second minimum, strut(s) and/or wall fuel injection, and the higher inlet pressures afforded by forebody precompression. Credibility for this extrapolation and for the assumption that an engine similar to the one suggested herein can be developed is supported by the test results of a fixed-geometry dual-mode scramjet at Mach 3 and Mach 5 with hydrogen[17] and at Mach 6 with hydrocarbons.[16]

The assumption of a dual-mode scramjet throughout this study does not preclude the use of ramjet propulsion systems for missiles having cruise speeds approaching Mach 6; however, the supersonic combustion mode of operation, with its lower combustion pressures and temperatures, is attractive.

Structure/Materials

Definition of a feasible structural concept to withstand the severe thermal environment of hypersonic flight without regenerative cooling, other than perhaps the fuel injection struts, was one of the more challenging aspects of this study. The dynamic pressures which a hypersonic missile experiences can be in the hundreds of kPa. Such extreme dynamic pressures place severe stiffness demands on the missile structure in order to prevent flutter. High dynamic pressures also cause high heating rates and attendant thermal stresses unless high thermal conductivity materials can be used to relieve temperature gradients. Hypersonic missile end-game maneuvers may impose large normal load factors, 30 g's or more, which also increase strength requirements. The only relaxation of requirements compared to the structure of, say, a hypersonic cruise airplane is in the cycle life and the size. Missiles are one-shot vehicles which allow structural design liberties that other vehicles could not tolerate.

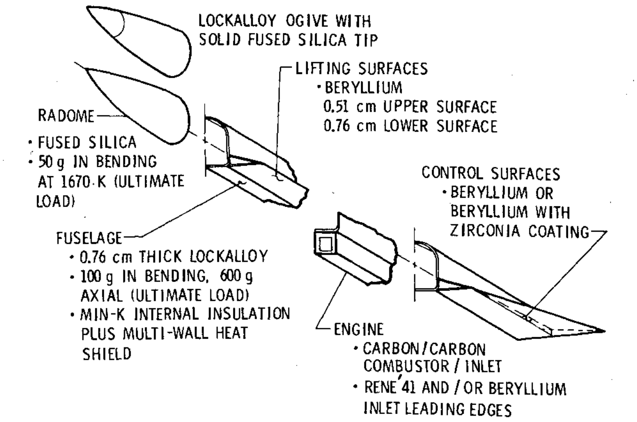

Fig. 5 Structures and materials approach.

For short flight times, the thermostructural problem is similar to that of the hypersonic research aircraft[6,7] which led to the investigation of a heat-sink system. One feasible lightweight heat-sink structural approach is illustrated in Fig. 5. The fuselage structure was designed with a 7.6 mm thick Lockalloy material to provide a heat-sink range of about 200 n.mi. at Mach 6 cruise conditions and 32 km altitude. The fuselage structure was not allowed to exceed 530 K (even though the operational limit of Lockalloy is near 700 K) in order to keep the internal insulation requirements manageable. Such a heat-sink designed shell could withstand an ultimate boost load of 600 g's axially at 300 K and 100 g's in bending at 550 K. Thus, it appears that shell-structure approaches are sufficient for most tactical applications and that only a modest amount of internal structure would be required with the larger, longer range missiles.

The low-aspect-ratio wings were designed with beryllium plate to allow prolonged flight at Mach 6 and 32 km altitude. Beryllium was also selected for the control surfaces because of potentially high interference heating rates. Lockalloy and beryllium were considered because of their high conductivity (14 to 25 times higher than that of the superalloys or titanium, respectively) and high ratios of stiffness to weight, heat-sink capacity to weight and strength to weight (Fig 4 of Ref. 15). The structural stiffness necessary to prevent flutter during high-dynamic-pressure flight conditions is also provided by these two materials, which have the highest stiffness to weight characteristics of any high-temperature heat-sink material. Lockalloy was selected for the fuselage structure, rather than beryllium, because of its more desirable fabrication characteristics. Although Lockalloy has not been used as envisioned herein, some material characterization data are available, and it has been used to construct a ventral fin for the YF-12.[21]

For ranges exceeding the Lockalloy fuselage structure heat-sink limits, a radiation TPS[22] was added to the fuselage, which prevented the underlying structure from exceeding 530 K at Mach 6 cruise conditions. This multiwall radiation shield concept (Fig. 6) consists of layers of dimpled and flat sheets welded at the crests of the dimpled sheets to form a sandwich vented to the atmosphere. Pressure loads are transmitted through the insulating layers to the inner Lockalloy structure with negligible deformation normal to the insulating layers. According to the research of Jackson et al,[22] this multiwall

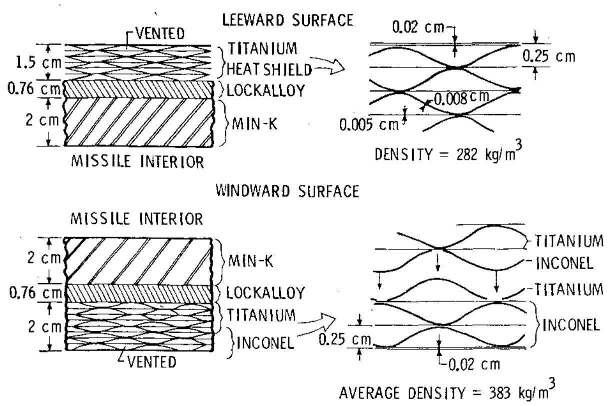

Fig. 6 Fuselage thermal protection system.

sandwich can be fabricated from several metals without having to weld dissimilar metals. A 2000 n.mi. cruise range at Mach 6 for example, requires bimetal multiwall composed of three outer layers of Inconel and four of titanium for the windward surface (Fig. 22 of Ref. 15) with a weight per unit surface area of 7.8 kg/m^2. (TPS is not required for beryllium wings and control surfaces since beryllium has extremely high heat-sink capability and could also operate at radiation equilibrium temperatures at Mach 6 and 32 km altitude.) The multiwall for the lee surface may consist solely of titanium since the radiation equilibrium temperatures are less severe than on the windward surface. For shorter cruise ranges, of course, the number of inner layers may be reduced to accommodate the smaller thermal loads.

Slip-cast fused silica appears to have promise as a radome material for high-Mach-number missiles,[23] at least up to Mach 6. For applications not requiring a radome, a Lockalloy shell with a solid fused silica or a carbon-carbon nose tip was assumed.

The definition of a practical high-temperature, lightweight structural concept for the engine is obviously critical to the potential performance of a hypersonic missile. Rene'41 is an attractive material candidate for the inlet leading edges. The temperature limit of this superalloy is sufficient to withstand the equilibrium temperatures of the inlet's sharp, swept (48 deg) leading edges at Mach 6 cruise conditions, although a zirconia coating may be advantageous at the stagnation points of the lower cowl (Fig. 3). Also, heat-sink leading edges of beryllium would suffice for moderate flight times (Fig. 20 of Ref. 15). A three-dimensional woven, carbon-carbon material[24] could be used for the remainder of the rectangular inlet and the combustor. This coated carbon-carbon retains adequate strength for the combustor wall up to about 3000 K and is very lightweight (1520 kg/m^3). Internal pressure loads are relatively modest. For example, combustor pressures for supersonic combustion in the baseline engine at Mach 6 will not exceed 1050 kPa even at dynamic pressures of 480 kPa.

Fuel injection from struts as well as the combustor wall may be desirable[16]; however, the struts will require cooling. Calculations patterned after those of Ref. 1 indicate that at Mach 6, $q = 48$ kPa, and $\phi = 0.76$ sufficient cooling capacity exists in the alkylated borane fuel flow (allowing 167 K fuel temperature rise) to cool three swept fuel injection struts, if needed. The structural material assumed for the fuel struts was Rene'41; however, beryllium is also a good candidate material.

Results of extensive transient heating analyses of the foregoing thermostructural concepts are presented in the appendix of Ref. 15. These results indicate, for example, that for Mach 6 cruise conditions ($q = 24$ kPa), essentially sharp, swept (74 deg) leading edges of beryllium or advanced titanium could be used without protective coatings for almost indefinite flight time.

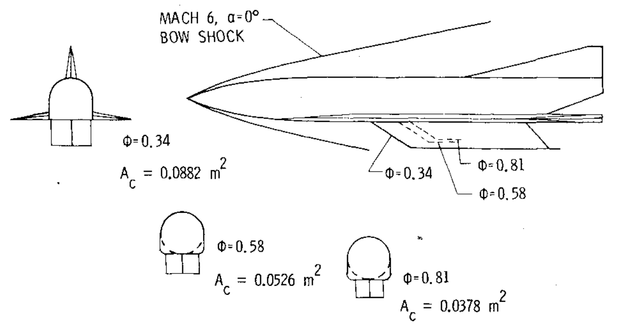

Fig. 7 Engine sizing on baseline configuration; Mach 6, $q = 24$ kPa, cone precompression.

Engine Sizing

The size and weight of the baseline engine is based on the scramjet concept shown in Fig. 3.[1] The two modular engines for the baseline configuration are sized for Mach 6 cruise with an equivalence ratio of 0.34 at a freestream dynamic pressure of 24 kPa, assuming cone surface precompression and the measured inlet performance of this fixed-geometry design.[12] The engine size for cruise could be reduced by a factor of two or more by increasing the cruise equivalence ratio from 0.34 to 0.81, as shown in Fig. 7. However, for a given cruise Mach number, the larger engines, though heavier, will operate with lower heating rates and possibly higher specific impulse at the lower equivalence ratios. Furthermore, this oversizing of the engines also provides acceleration capability from staging at Mach 4 to cruise at Mach 6 and allows a relatively large and efficient nozzle with a nozzle exit to inlet cowl area ratio of two. Since the objective is to determine "across the spectrum performance potential" and not to optimize for a given mission, the larger engines ($A_c = 0.09$ m^2) were chosen for the baseline configuration.

Packaging and Weights (Missile and Boosters)

The packaging of the baseline missile fuselage (antenna, payload, electronics, controls, auxiliary power unit, etc.) is consistent with current missile design studies. The weight statement for the baseline configuration is given in Table 1. In establishing the weight of the engine, no consideration was given to the lengthening of the combustor (estimated between 15 and 25%) that probably would be required in converting from hydrogen to "hydrocarbon" fuel or to a second minimum. However, the weight estimate should be representative, considering that the lighter weight carbon-carbon material could be used instead of Rene'41 or beryllium for a majority of the inlet leading-edge structure.

The boosters (Fig. 1) were sized assuming a propellant mass fraction of 0.82 and a specific impulse of 235 s. A tandem booster sized to accelerate the baseline missile from Mach 0.8 to 4 at 40,000 ft altitude in 4.5 s would weigh approximately 700 lb. A similar booster sized for staging at Mach 6 would weigh approximately 1250 lb.

Table 1 Baseline configuration weight statement, $l/d = 9.5$

Item	Material	Weight, kg
Radome	Fused Silica	15
Fuselage	Lockalloy	56
Wings	Beryllium	25
Fins (3)	Beryllium	16
Engines	Carbon-carbon/Rene'41 and/or beryllium	83
Payload		45
Fuel	(Aklylated borane)	64
Internal parts		125
Miscellaneous		7
		436

Performance

Cruise

Cruise ranges which were calculated for the baseline configuration at Mach numbers of 4, 5, 6, and 7, assuming staging occurs at cruise Mach number, are presented in Fig. 8 for flight dynamic pressures ranging from 24 kPa to 335 kPa. A cruise range of 740 n.mi. at Mach 6 at a dynamic pressure of 24 kPa would be possible with the baseline configuration. Cruise range drops rapidly with increasing dynamic pressure because of the decreasing cruise L/D. The maximum ranges occur at Mach 5 across the dynamic pressure range shown; however, there is little difference in range between Mach 5 and Mach 6 cruise above a dynamic pressure of 100 kPa. Although not indicated in Fig. 8, lengthening of the missile could be an effective means of increasing range since fuel is the predominant weight added. For example, a 160% improvement in the Mach 6 cruise range can be realized by lengthening the baseline missile 17%, allowing a 75% increase in fuel.

Acceleration

The airbreathing engine could be used to accelerate the missile from a lower staging Mach number to a higher cruise Mach number, but the range from the staging point would be substantially reduced. For example, calculations indicated that, operating at stoichiometric conditions, the baseline missile could accelerate from staging at Mach 4, 12 km altitude to Mach 6, 31 km altitude in 24 s. Cruise at Mach 6 was assumed to be initiated at $\phi = 0.34$. Fuel burnout occurred 300 n.mi. downrange from the staging point as compared to 740 n.mi. for cruise at Mach 6 from a Mach 6 staging at 31 km altitude. The acceleration-cruise range penalty was sensitive to the rate of acceleration. For example, the range penalty for accelerating from Mach 4 to Mach 6 at equivalence ratios below 0.8 averaged approximately 40% as compared to using all the fuel for cruise at Mach 6.

Constant altitude trajectories were calculated for an initial Mach number of 4 and equivalence ratios of 0.6 and 1.0 to indicate the acceleration potential of the baseline missile. The results are shown in Fig. 9 as a line of range maxima for the two values of equivalence ratio. The maximum range varied from about 15 n.mi. at 11 km altitude for $\phi = 1$ to about 250 n.mi. at 26 km altitude for $\phi = 0.6$; fuel burnout Mach numbers are 7.6 and 7.3, respectively. Horizontal accelerations of up to 9 g's at altitudes between 8 and 9 km were calculated for Mach numbers of 4 to 7. At these high dynamic pressures, very little precompression is realized because the airframe is operating near zero lift. Additional acceleration/range information for a modified baseline missile which achieved maximum speeds approaching Mach 9 is presented in Ref. 15.

Maneuvers

Horizontal Turns

The thrust of an airframe-integrated propulsion system increases with angle of attack because a large portion of the air compressed by the vehicle for lift is delivered to the engine inlet. For modest angles of attack, the thrust in the flight-path direction increases faster than the drag, as shown for the baseline configuration at Mach 6 in Fig. 10. This characteristic results in large thrust-to-drag ratios occurring at substantial lift coefficients, thus providing high-turn rate-capability without loss of velocity.

An illustration of the constant-velocity turns that may be accomplished as a function of dynamic pressure at Mach 6 is also presented in Fig. 10. The number of g's pulled, the turn radius and the bank angle for angles of attack of 2, 8, and 16 deg are given. This figure indicates the high g turn capability of this type of missile at relativley low angles of attack. The thrust equals the drag at an angle of attack of approximately 14 deg indicating that the baseline airbreathing missile could accelerate in turns requiring angles of attack less than 14 deg (assuming that the bow shock does not enter the inlet). This type of acceleration capability is desirable in applications for which high maneuverability is required. For instance, in following moving targets at a given altitude, speed becomes an advantage since the response time and consequently the miss distance decreases as dynamic pressure increases. (Response times decrease at high dynamic pressure because only a small angle of attack change, and thus small elevon deflections and short actuator strokes, are required to produce a large g forces normal to the established trajectory direction.)

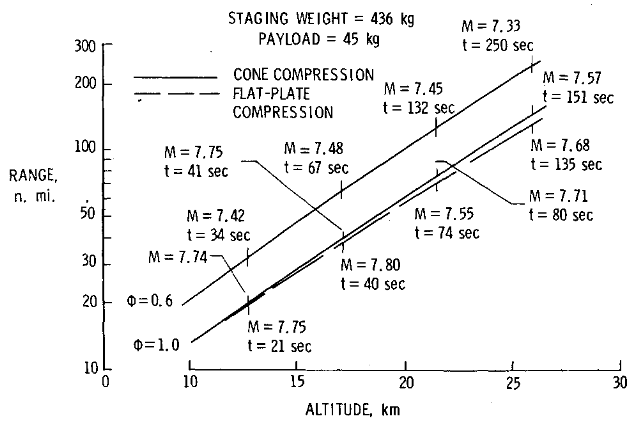

Fig. 9 Range of baseline during acceleration from Mach 4 at various constant altitudes; $A_c = 0.0882\ m^2$.

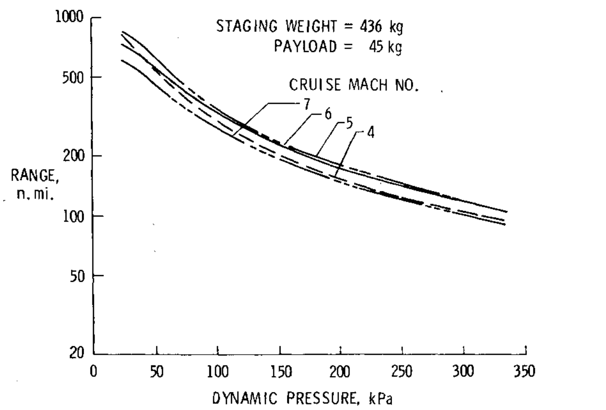

Fig. 8 Cruise range variation with dynamic pressure; baseline configuration, cone precompression.

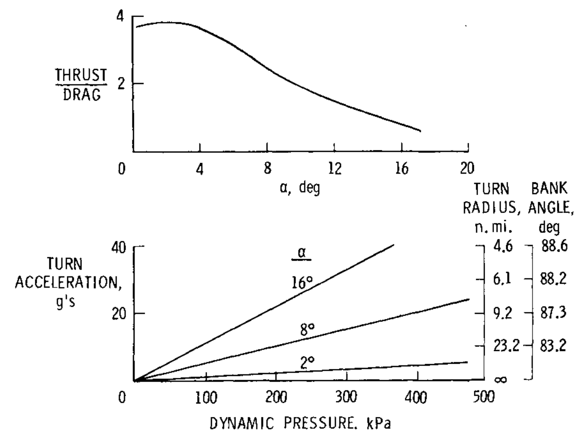

Fig. 10 Maneuverability at Mach 6.

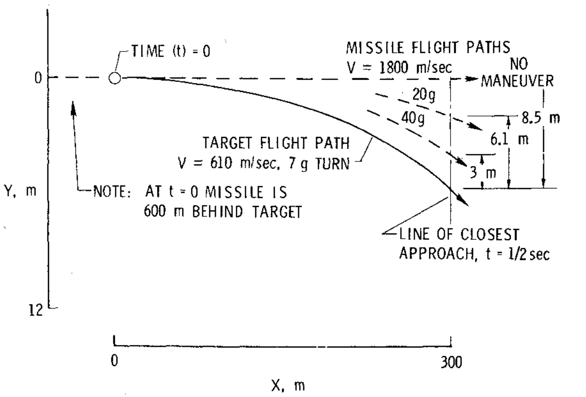

Fig. 11 Intercepting moving target from the rear.

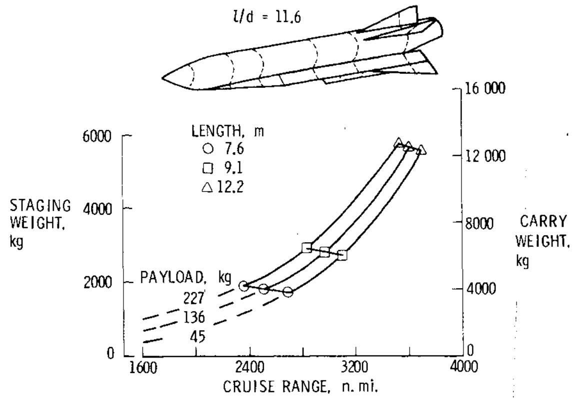

Fig. 13 Range, weight, and payload relationship for cruise missile; $l/d = 11.6$.

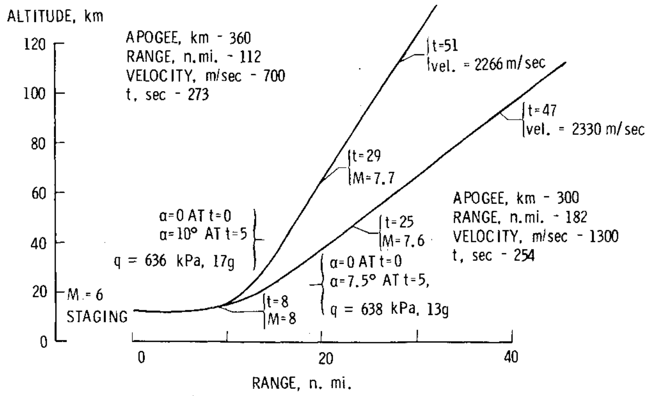

Fig. 12 Pull-up from Mach 6 stage at 12.2 km.

Intercepting maneuvering targets was not analyzed in detail; however, one example of a Mach 6 missile rearward approach against a Mach 2 target was considered. In this case the aerodynamic response time of the Mach 6 missile was estimated to be 1/3 s and the Mach 2 target was allowed an optimal "sidestep" maneuver for evasion; that is, the target continued on course until the missile was 1/3 s away and then initiated, instantaneously, a 7 g evasive turn. As shown in Fig. 11, even if the Mach 6 missile does not maneuver at all the miss distance is only 8.5 m. The miss distance can be reduced to about 3 m if the missile performs a 40 g turn.

Climbs and Dives

A hypersonic missile has a high ratio of kinetic to potential energy and thus has the capability to reach high altitudes by trading kinetic for potential energy. The baseline configuration could make this trade very effectively since the normally energy-consuming transition from zero flight-path angle to a steeply climbing trajectory could be made with a gain in velocity because of the large thrust-to-drag ratio and added lift available during pullup. (This rationale also applies to powered dives.) Example climb trajectories are shown in Fig. 12 for a modified baseline missile which was lengthened 0.3 m to carry extra fuel. The missile/booster system was assumed launched at $M = 0.8$ at an altitude of 12 km, and staging was assumed 8 s later at Mach 6 with no increase in altitude. The missile was calculated to accelerate on air-breathing propulsion from Mach 6 to Mach 6.9 in 5 s at 0-deg angle of attack. The angle of attack was then increased to 10 deg for the climb. During the initial stages of the climb, the missile would experience a maximum of 40 g's vertical acceleration with a fuel flow rate of 9 kg/s, reaching a maximum velocity of 2600 m/s at an altitude of 24 km and a flight path angle of 65 deg, 13 s after launch. The engine power was assumed terminated at 43 km altitude because of imposed minimum dynamic pressure restraints on combustion. The missile then coasted to an altitude of 360 km, 273 s after launch. A second pull-up trajectory in which the angle of attack was increased to 7.5 deg at Mach 7.3 resulted in a more shallow climb and slightly higher velocities.

Pull-outs from power-off dives with operational empty weight from a Mach number of 6 and 31 km altitude assuming initial flight path angles of -30 and -60 deg were also studied (power-off engine drag included). The initial angle of attack for each of these trajectory calculations was set at 0 deg. After 5 s into each dive, the angle of attack was changed to 2.5 deg to initiate pull-out. For the -30 deg flight path angle dive, pull-out (level flight) was calculated after 32 s at a Mach number of 4 and an altitude of 9 km. The maximum normal load factor was 7.7 g's. For the -60 deg flight path angle dive, pull-out was calculated after 25 s at a Mach number of 2.8 and an altitude of 2.4 km, and inducing a 14 g normal load factor. Oscillatory-type climbs and dives, in which the airbreathing missiles uses both power-on and power-off flight segments, are discussed in Ref. 15.

Range/Weight/Size Relationships

Although a detailed sizing excercise was beyond the scope of this study, a first-order scaling effort was conducted to more fully assess the potential of this concept. Missile lengths of 7.6, 9.1, and 12.2 m with a constant l/d of 11.6 and payloads of 45, 136, and 227 kg were considered. Multiwall radiation shields were added to protect the Lockalloy fuselage structure, and the beryllium plate forming the wing surface was thickened to maintain a heat-sink wing. Anticipating a possible loss of control effectiveness because of shadowing by the lengthened fuselage, two side fins were used instead of a center vertical fin. The engines were sized for cruise ($M = 6$, $q = 24$ kPa) at an equivalence ratio of 0.58. Staging weight as a function of range is shown in Fig. 13. Range in this case is powered range at cruise conditions, thus neglecting ascent and glide components. An estimate of booster plus staging weight is shown as carry weight on the alternate weight scale. The boosters were sized to accelerate the missile from $M = 0.8$ to $M = 6$. The results for this particular configuration show that 3000 n.mi. ranges are easily obtainable and not strongly affected by payload variation; however, doubling the carry weight may not provide 4000 n.mi. capability, and the range advantage of lengths over 9 m diminishes rapidly.

Liquid hydrogen fuel was also considered during this sizing study. For the 136 kg payload and the 12.2 m missile length, the Mach 6 cruise range was less than one-third that of the missile fueled with HiCal 3-D. (The thrust per pound of inlet air is comparable for the two fuels.) The range performance of this limited volume vehicle and the logistics of maintaining a missile at the ready with cryogenic fuel would thus seem to preclude the use of hydrogen fuel for many cruise missile applications for Mach numbers below 8.

Sizing of missiles smaller than the baseline was not compatible with the packaging assumptions of this study; however, there is no basic reason why the concepts discussed here could not be applied to smaller missiles. Actually, the heat-sink structural concept should be increasingly attractive for smaller, shorter range missiles with a shorter heat pulse.

Potential Areas for Further Research

This study by no means addresses all of the technical solutions for hypersonic airbreathing missiles. Certain assumptions are inherent and certain problem areas such as guidance and control are omitted. Nevertheless, attractive potential performance appears achievable if the concepts addressed in this study can be further developed. The following discussion outlines research areas in propulsion, propulsion integration, configuration synthesis, and structures/materials which seem warranted to expand a technology base for hypersonic, airbreathing, lifting missiles.

This study suggests that a "hydrocarbon-fueled," modular, airframe-integrated, dual-mode scramjet with a variable second minimum would be a promising propulsion system for a hypersonic missile. Noncryogenic fuel selection, method of fuel injection (strut and/or wall), techniques for igniting and sustaining efficient supersonic combustion[16,19] and for converting from subsonic to supersonic combustion are areas in which basic research programs are needed. Cycle analyses, inlet option evaluations, and inlet/combustor decoupling techniques,[14,17] combustor contour, combustor cross-section[25] and combustor length requirements,[16] as well as techniques for providing a variable-area second minimum or effective thermal choking[17] are needed to better assess the viability of this concept.

The performance and size of the engine and the aerodynamic efficiency and maneuverability of the missile depend strongly on the mutual interaction between the propulsion system and airframe. The uniformity of the inlet flow[26] including the boundary layer and ability to nest the engine within the shock layer at high angles of attack and/or Mach numbers are features crucial to the performance of the missile. Not only should the external nozzle be designed to extract the maximum possible thrust force from the high energy flow at the combustor exit, but also the forces produced by the nozzle must be aligned such that they do not unduly affect aerodynamic balance.[10] The strong coupling between the propulsion system and the aerodynamics of the missile requires at least a partial experimental simulation of the inlet, exhaust, and external flows to define the actual interactions.[27] Inlet and aftbody nozzle flowfields must be determined, precompression surfaces and aftbody nozzles optimized, and aerodynamic-force data acquisition techniques developed. In addition, alternate propulsion system locations (e.g., on top of the missile) should be considered and performance implications determined.

Aerodynamic/volumetric efficiency is a critical design goal for any cruise missile. This, coupled with stability/control and bank-to-turn maneuverability requirements at hypersonic speeds as well as the need to keep observables and aerodynamic heating at a minimum, provides a significant challenge for the researcher in configuration synthesis.

Adequate materials[15] seem to be available for hypersonic airbreathing missiles (at least up to Mach 6). The operational flight regimes of these missiles impose thermal stress constraints as well as severe stiffness constraints, especially for the more slender configurations. Hybrid systems using combinations of external TPS and heat-sink structure appear to offer potential weight savings for long-range missiles, but the true usefulness of these concepts remains to be proven and comparisons with hot structure options are yet to be made. These hybrid systems also will require optimization and experimental verification. In addition, the compatibility of structural materials and thermal protection coatings on control surfaces and inlet, combustor, and aftbody nozzle walls is a concern, as well as the thermostructural design of these items. The use of storable heat-absorbing materials[15] may offer a high payoff, but further research is needed. Radomes that can operate effectively in the severe thermal environment as well as resist rain erosion[23] and withstand up to 40 g's, or more, normal load factors need further research.

Concluding Remarks

This study indicates that many of the technologies NASA is developing for hypersonic aircraft are, with the appropriate modifications, applicable to and beneficial for hypersonic, airbreathing, lifting missiles. For example, careful integration of the modular propulsion system with the airframe in which the forebody is used as a precompression surface and the aftbody as a nozzle yields large benefits for missiles as well as aircraft. These benefits include smaller inlets, lower drag and greatly improved maneuverability. The use of heat-sink materials and multiwall thermal protection systems also appears to offer potential benefits for missiles.

These concepts have been used in the formulation of a hypersonic, staged missile concept characterized by aerodynamic efficiency, structural integrity, and a fuselage-integrated, airbreathing, "hydrocarbon-fueled" dual-mode scramjet propulsion system. The flat-bottom, low-wing design provides an efficient lifting and precompression surface that allows ample space between the bow shock and fuselage for integration of engine modules sized for cruise, for acceleration, or for both, at Mach numbers from 4 to at least 7. The lightweight Lockalloy heat-sink structure is a viable concept for Mach 6 missiles with ranges of 200 n.mi. or less. A titanium and Inconel multiwall heat-shield system used in conjunction with the heat-sink concept provides high strength and stiffness with good thermal protection at low weight for extended cruise ranges.

It is recognized that the modular, dual-mode scramjet suggested in this paper would require a major development effort and that a nearer-term propulsion system, an airframe-integrated ramjet, could possibly provide similar performance characteristics at speeds up to near Mach 6. In either case, substantial research progress must be made in propulsion, propulsion integration, configuration, synthesis, and structures/materials if a viable technology base is to be developed.

Overall, the concept of a hypersonic, airbreathing, lifting missile appears to offer attractive performance and warrants further exploration.

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