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Beating the Rocket Equation: Air Launch with Advanced Chemical Propulsion

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Performance figures are presented for reusable, winged rocket stages launched from several large transport aircraft, including the Boeing 747, the Russian Anotonov An-226, a large supersonic aircraft comparable to the XB-70 aircraft, which achieved Mach 3.1 flight with conventional turbojet propulsion in 1964, and a Mach 6 conceptual aircraft based on the MA145-XAB ramjet demonstrated in 1968. Advantages of air launch include a reduced ascent-to-orbit delta-velocity, reduced drag, the capability to launch at any latitude, and simplified abort modes. Launch from the XB-70 flight condition would allow a 22% reduction in delta-velocity required of the rocket and a significant reduction in drag incurred by rocket flight beginning at an atmospheric density 1/20th that of sea level. Launch from a Mach 6, 85,000-ft altitude condition would allow a 33% reduction in delta-velocity; a reusable rocket of 262,000 lb launched at this condition could deliver a 23,500-lb payload to orbit utilizing a 523-s vacuum specific impulse advanced rocket engine. Fluorine/lithium–hydrogen engines achieved 523 s in U.S. Air Force and NASA development programs of the 1960s and 1970s. Comparisons to ground-launched, all-rocket vehicles delivering equivalent payloads to orbit are presented. Fluorine propellant reactivity and engine development history are also discussed.

Introduction

TOTALLY reusable spacecraft concepts that are air launched from the Boeing 747 and Russian Antonov An-225 subsonic transports, as well as from high-Mach, high-altitude carrier aircraft, for access to low Earth orbit (LEO) are presented. One of the goals of this design study is a reduction in recurring costs in comparison to the Space Shuttle, at a lower development and acquisition cost than that of a reusable, ground-launched, all-rocket two-stage-to-orbit (TSTO) rocket of the type presently under consideration by NASA. A variety of studies have examined air launch of rocket vehicles for Earth-to-orbit flight. Supersonic to low hypersonic aircraft first stages were analyzed as boosters for reusable launch vehicles as long ago as the 1950s.[1] Later it was recognized that even a subsonic aircraft could provide significant performance advantages. Today the small Pegasus launcher operates air launched and other air-launched concepts are under active study. Payload to altitude capability for a modified Boeing 747 is approximately 315,000 lb to a 30,000-ft altitude at a Mach number of 0.75. Subsonic air launch at this release condition reduces by 8% the total delta-V to orbit for the rocket portion of flight; at supersonic and high altitude release conditions the benefit increases measurably; at a 73,000-ft altitude/Mach 3.1 condition the delta-V is reduced by 22%; at 85,000 ft and Mach 6.0 the reduction is 33%.

The rocket equation, governing the performance of propulsive rocket flight, relates rocket initial-to-final-mass ratio (M_o/M_f) to an exponential function of delta-V; and even small reductions result in important savings in required rocket mass; a 22% reduction would result in dramatic savings. Other advantages of air launch at altitude include reduced drag incurred by rocket flight beginning at an atmospheric density much reduced from that of sea level (at 73,000 ft, density is 1/20th that of sea level), less stressing ascent loads, the ability to launch at any latitude, and simpler abort modes.

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A comparison of ascent delta-V is given in Table 1 and Fig. 1; values for several start altitudes and Mach numbers are compared to that of a ground-launched all rocket TSTO flying to the same orbital conditions. The data in Table 1 are valid for initial rocket thrust-to-weight ratios (T/W) of 1.24, initial flight-path angle at rocket ignition of 30 deg, and flight to 28-deg, 100-n mile circular orbits.

Benefits of Air Launch

Several benefits can be achieved because of launch at altitude. The first and primary benefit involves atmospheric density. The initial portion of a ground-launched rocket's ascent occurs within the denser portions of the atmosphere, with inherent inefficiencies related to aerodynamic drag; its flight path purposely begins vertically and continues on a near vertical path until past the denser portions of the atmosphere, where it tips over to fly a more nearly horizontal path. During its initial, vertical flight, severe aerodynamic loads are encountered, necessitating, as in the case of the shuttle, the throttle down of the orbiter engines to decrease velocity and reduce the maximum dynamic pressure (Max "Q") loads on the vehicle. Significant gravity and drag losses are incurred during the ascent, typically on the order of 5000 to 6000 ft/s—losses equivalent to about 20% of the actual final inertial velocity required to reach orbit (25,570 ft/s for a 100-n mile circular LEO). By starting at an altitude above the denser portion of the atmosphere, gravity and drag losses can be reduced to about 2500 ft/s. At a 30,000-ft altitude, about 5 n miles, the density of the atmosphere is about 1/3 of the air density at sea level; at 73,000-ft altitude, about 12 n miles, the density is 1/20. In addition, a more efficient engine nozzle design can be used as compared to ground launch because of the lower ambient pressure existing at ignition. Any ground-launched vehicle's first-stage nozzle design is typically a compromise as a result of the altitudes (and thus pressures) it experiences upon ascent. For air launch the nozzle need not be as compromised because it operates over a smaller range of pressures. Without the limitations imposed by fixed launch site latitude and the proximity of populated areas, a wide range of orbital inclinations can be entered as a result of the mobility of the launch platform. Ground-launched systems are restricted to only a relatively few inclinations unless wasteful "dog-leg" maneuvers are utilized. Launch from any airport with a 10,000-ft class runway and the requisite rocket-propellant loading facilities is possible. (Ground-launched rockets cannot do this because of noise and sonic boom.)

Table 1 Air and ground launched rocket ascent delta-velocity, ft/s

Parameter	Air launch				Ground
Takeoff Mach number	0.75	2.5	3.1	6.0	0.0
Takeoff altitude, ft	30,000	65,000	73,000	85,000	0.0
Takeoff flight path angle, deg	30	30	30	30	90
Final inertial velocity 100 × 100 n mile	25,570	25,570	25,570	25,570	25,570
Surface velocity at 28.5 deg due east	−1,340	−1,340	−1,340	−1,340	−1,340
Take-off relative velocity	−746	−2,420	−3,016	−5,886	−0
Delta-V required of rocket, no losses	23,484	21,810	21,214	18,344	24,230
Gravity/drag/steering/nozzle losses	+4,225	+2,771	+2,430	+1,838	+5,928
Delta-V required of rocket, with losses[b]	27,926	24,581	23,644	20,182	30,158[b]
Percentage of ground launch delta-V	92%	82%	78%	67%	100%

[a]Take-off $T/W = 1.24$, $I_{sp} = 523$ s. [b]Take-off T/W 1.49.

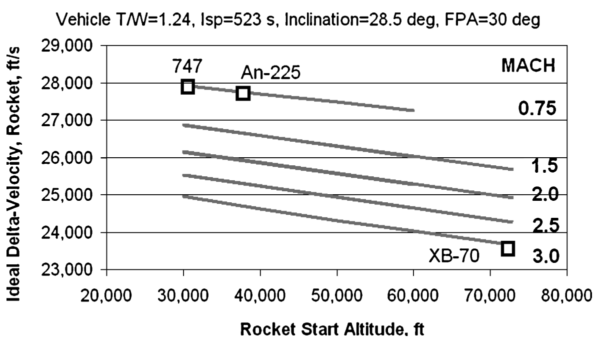

Fig. 1 Air-launch rocket ascent delta-V vs start altitude and Mach number.

Fig. 2 747 with Enterprise orbiter: separation maneuver.

Fig. 3 747 Shuttle ferry aircraft and orbiter in loading tower.

The benefits associated with reduced delta-V, launch positioning, reduced structural loads, and more efficient nozzles would allow air-launched rockets to achieve significantly higher payload fractions then their ground-launched counterparts. For example, Pegasus has a payload fraction almost twice that of a ground-launched vehicle of similar size.[2] Finally, air launch inherently implies ready-made transportation for the rocket to and from the manufacturing site, test site, and recovery site. The additional cost of the transport carrier aircraft is not a disadvantage, as a ground-launched TSTO could also require a large transport aircraft as a ferry vehicle for the orbiter element (such as the 747 ferry for the shuttle orbiter). The air-launched solution offers a smaller system, with less to inspect and refurbish when compared to a TSTO ground-launched rival having an equivalent payload capability.

Disadvantages of Air Launch

If a unique, one-of-a-kind carrier aircraft must be designed and developed, this would represent a significant expense; however, so to would the first stage of a ground-launched, all-rocket TSTO vehicle—especially if the first-stage rocket is a reusable, winged, turbojet-powered flyback booster. With air launch there are integration constraints; subsonic air launch makes integration of the rocket and aircraft rather simple compared to supersonic air launch, mainly because the supersonic system must be carefully tailored to enable acceleration through transonic drag rise. Maintenance of cryogenic propellants during aircraft climb-out might require a Dewar and top-off system built into the airplane. Separation is a difficult maneuver that must be carefully planned. The Shuttle Ferry 747 and the Enterprise orbiter test vehicle demonstrated separation at altitude in 1980 (Fig. 2). (Orbiter loading facilities for the Shuttle Ferry 747 are shown in Fig. 3.) Maintenance of cryogenic propellants during aircraft climb-out may require a Dewar and top-off system built into the airplane. For supersonic aircraft, the rocket stage can be externally mounted and immersed (slotted) into the back of the carrier for transonic drag reduction.

Air-Launch Flight Profile

After takeoff, the aircraft/rocket combination climbs to altitude, by which time preparation of the rocket systems is completed. At design altitude the carrier enters a very shallow dive and then pulls up to release the rocket so that a flight-path angle for launch of 30 deg is reached. At that time the aircraft pushes over to zero gs for rocket release. Rocket T/W at liftoff is 1.24, large enough to limit gravity losses but not excessive enough to require substantial engine throttling (below 50%) during flight to avoid exceeding maximum acceleration limits (3.0 g). After completing its mission, the vehicle conducts a deorbit burn via its orbital maneuvering system (OMS), reenters, and lands. The carrier/rocket separation latitude and longitude can be selected so that if a difficulty occurs during or immediately after main engine start, the rocket can shut down and glide to a landing strip unpowered. If the aircraft experiences an engine failure before rocket takeoff, rocket propellants are dumped and the aircraft returns to land with the vehicle still attached.

Rocket Design

A representative single-stage, winged, fully reusable air-launched rocket is shown in Fig. 4 and 5. It incorporates forward and aft cylindrical propellant tanks, both of which are integral and load

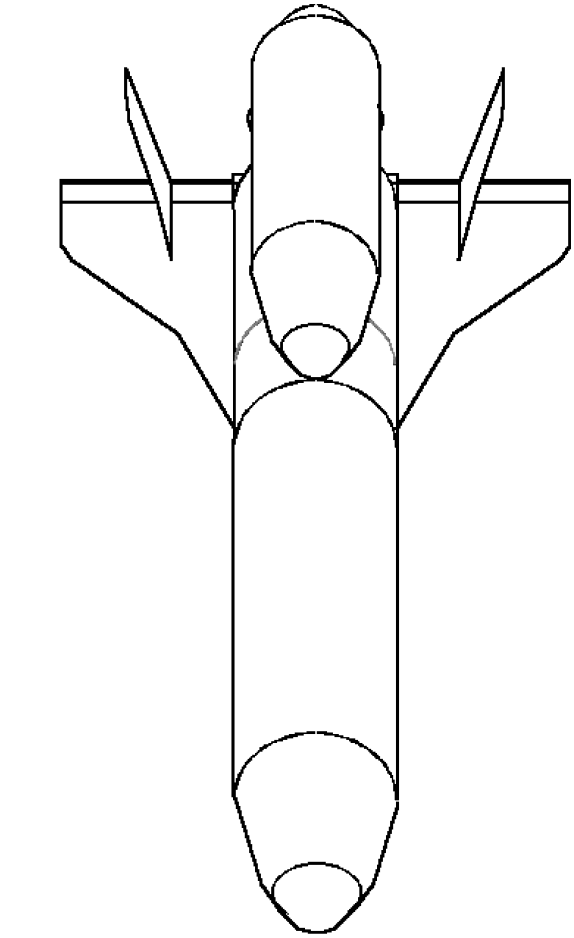

Fig. 4 Air-launched rocket.

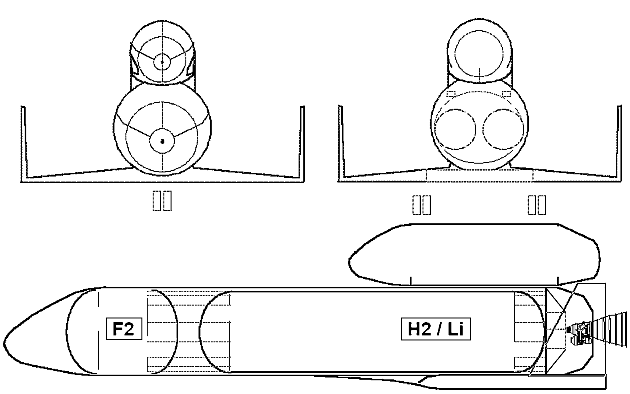

Fig. 5 Air-launched rocket views.

bearing. Aluminum or aluminum–lithium (Al-Li) is used for the fuel tank, and titanium (Ti) alloy is used for the oxidizer tank. Tank insulation is provided by 1.1-in.-thick advanced polymide foam (APF), at a density of 3.2 lb/ft^3. The aft fuselage contains the thrust structure, two main engines, OMS, aft reaction control system (RCS) thrusters, engine feed lines, and the body flap. OMS and RCS utilize oxygen (O_2)/ethanol propellants. Much of the internal and non-tankage load bearing (interstage) structure is composite material. The area ahead of the forward tank houses the nose landing gear. The reentry thermal protection system (TPS) consists of advanced carbon–carbon material for leading edges, toughened unipiece fibrous insulation (TUFI) or high-temperature reusable surface insulation (HRSI) tiles of 2.0 lb/ft^2 areal density for high heat load areas, and advanced flexible reusable surface insulation (AFRSI) blankets of 1.0 lb/ft^2 for topside, lower heat load areas. The wing is a trapezoid planform with trailing edge ailerons and flaperons. Midwing or tip-mounted vertical fins with rudders provide directional stability and control. The main landing gear is attached to the wing box near the body join. Aerosurfaces were designed for minimum area consistent with the wing loading factor, i.e., the total mass of the space plane at reentry divided by the effective wing surface. As the wing load increases, the thermal flux during reentry increases also and the thermal protection must resist higher temperatures. A further consideration is that the landing speed will also increase. A landing wing loading of 85 lb/ft^2 was selected for the vehicles presented in this analysis. A constant chord body flap for pitch control and trim is attached at the base of the closeout bulkhead.

An external detachable payload pod is attached to structural hard points on the vehicle above the aft tank and above the vehicle center of gravity. The aerodynamic complexity associated with an external payload bay is less severe for air-launched than for ground-launched configurations, as a significant portion of the low-level, high-dynamic-pressure region of flight is avoided. The external position more readily accommodates a crew escape system (as an ejectable cabin or ejection seats) for missions carrying passengers. External payload pod packaging efficiency might approximate that of a typical shroud of an expendable vehicle in some cases; in others, such as personnel delivery, the pod may be a single dedicated passenger-cabin-type structure. In the analyses described in the following sections, reported payload pod weights are inclusive of the additional pod structure, payload door weight (if any), and pod thermal protection required for any payload. Actual enclosed payload would be less than the quoted pod value. Reusable vehicles described herein all have a return (down) payload capability equal to their payload to orbit capability.

An alternative drop-tank configuration of the rocket was also evaluated; the fuel is housed in an expendable tank attached via specialty interstage structure to the rocket's forward structure (behind the vehicle nose). After main engine cutoff (MECO) this tank and interstage is jettisoned, reducing the rocket's landing weight at the cost of partial reusability. For the rocket vehicles reported here, diameter was set to 15 ft and payload pod dimensions to 15 × 45 ft.

Mission Design Options

The space shuttle main engine (SSME) represents the product of a significant technology advancement program spanning two decades of design, testing, flight, and component improvement programs. Without the need for a compromise, reduced expansion ratio nozzle for initial sea-level liftoff thrust and low-altitude operation, a single position, large-expansion ratio (150:1) nozzle equipped Block II SSME would be able to deliver a vacuum specific impulse I_{sp} of about 462 s. However, an entirely reusable, all-rocket vehicle air launched from a 747-400F with separation at 30,000 ft and Mach 0.75, equipped with a SSME, would not achieve LEO orbit with any significant payload. However, several options are available to the designer to achieve some significant payload (thousands of pounds) to LEO via air launch. The first option would entail increasing the payload carrying capability of the transport aircraft to that well above the present capability of a modified 747-400F. A second option would entail the use of a complex liquid air cycle engine concept to collect oxidizer from the atmosphere during flight to reduce the amount of rocket propellant and thus the weight that the air transport aircraft would have to take off with. Concepts of this type are under active study.[3–5] A third option entails adding a rocket engine to the tail of the 747 to achieve a higher altitude, increased velocity departure point for the rocket.[6–8] A fourth option, chosen for this study, selects an engine for the rocket with a demonstrated I_{sp} significantly higher than that of an SSME.

Advanced Chemical Engine

An engine design was chosen that utilizes liquid fluorine (F_2) oxidizer with a combination lithium (Li)/hydrogen (H_2) fuel. This combination offers very high I_{sp} with a reasonably high density. The performance potential of the fluorine/lithium hydrogen (F_2/Li + H_2) combination is among the highest available from chemical propellants[9] and was tested both at Pratt and Whitney[10] in

Table 2 Fluorine-lithium-hydrogen propellant characteristics

Parameter	Fluorine oxidizer	Lithium fuel	Hydrogen fuel
Mixture ratio	53.1%	19.4%	27.6%
Molecular weight	38.0	6.941	2.016
Stored density, lb/ft^3	93.86	33.1	4.42
Freezing point, °R	96.4	813.9	24.9
Normal boiling point, °R	153.1	286.2	36.7

1963 and at Rocketdyne in 1967–70 under contract to the U.S. Air Force[11–13] and NASA.[14–18] Both ground and altitude tests were performed at Rocketdyne. For this application the lithium exists as a metallic gel or slurry in the hydrogen. At an engine chamber pressure (Pc) of 1000 psia and a nozzle expansion area ratio (AR) of 100, vacuum specific impulse is 523 s, which is about 65 s higher than the best I_{sp} attainable from O_2/H_2 engines. $F_2/Li + H_2$ propellant properties are given in Table 2.

Fluorine reacts very vigorously and is toxic. It is sensitive to moisture. If a suitable containment vessel material is not chosen, fluorine is so reactive it will ignite and burn the material. However, suitable tank and feed line materials can be treated such that they are nonreactive with fluorine. This is done by pacivating the inner surface of the containment vessel; in this process diluted fluorine gas is first introduced into the tank and reacts with it to form a fluorinated surface layer (coating) that acts to protect the tank inner surface when pure liquid fluorine is loaded later. The procedure is repeated several times, each with a less-diluted gas mixture, until a series of protective coatings are built up. A variety of engine test programs from the 1940s through the 1970s demonstrated pacivated containment materials, i.e., materials were made completely nonreactive to fluorine. The least reactive metal, fortunately, turns out to be a material that is an excellent choice for propellant tanks and feed lines, titanium. It is a lightweight, strong, and durable material that has excellent high-temperture properties. The procedure to pacivate the exposed surfaces of the titanium tank and propellant lines has to be done only once; thereafter they are nonreactive for the life of the system. Because of fluorine's sensitivity to moisture, care must be exercised when loading it; the tank must be dry as the fluorine would react with residual water condensate. Nitrogen purging systems are commonplace in rocket systems and would be utilized in this case for purging the tank before loading. After main engine cutoff the F_2 tank and its feed lines are valved open through the engine to vacuum; onboard nitrogen purge systems are activated removing completely all residual fluorine from the vehicle. Fluorine is also highly toxic, and care also must be taken with its vapors, as with other hypergolic propellants (such as nitrogen tetraoxide and hydrazine used on the shuttle OMS). More work needs to be done to quantify the extent of operational impacts caused by toxicity and reactivity concerns.

Another challenge posed by fluorine occurs in the engine combustion process; with lithium/hydrogen fuels fluorine preferentially reacts more quickly with the hydrogen; the lithium reaction takes slightly longer. (Lithium exists as suspended particles in the hydrogen.) This challenge was identified and overcome in the engine development programs cited. Propellant mixing and lithium vaporization approaches were developed and tested, demonstrating complete lithium combustion.[11] Maximum I_{sp} occurs at the stoichiometric F_2/Li mixture ratio (2.74) and approximately 28% H_2. (Overall oxidizer/fuel mixture ratio is 1.15.) The density of $Li + H_2$ fuel mixture is 17.8 lb/ft^3. Theoretical vacuum I_{sp} is plotted vs H_2 percentage in Fig. 6; maximum I_{sp} is 540 s (Pc = 1000, AR = 100). Because the $F_2/Li + H_2$ combination is composed of a 28% rather than the 11% (stoichiometric) mixture of H_2 usual for O_2/H_2 systems, overall $F_2/Li + H_2$ density is slightly lower than O_2/H_2. Actual attitude simulation tests with a 60:1 expansion ratio nozzle showed that theoretical performance can be approached in experimental firings. Reference 11 states, "Measured, uncorrected I_{sp} efficiencies were near 95%, corresponding to a vacuum I_{sp} of about 523 s deliverable by a regeneratively cooled engine at nominal operating conditions (Pc = 1000 psia, F_2/Li MR = 2.74, H_2 = 25%, AR = 100)." This I_{sp}

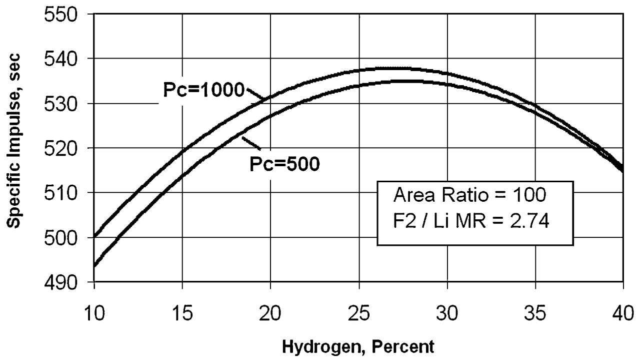

Fig. 6 Advanced chemical engine I_{sp} vs hydrogen percentage.

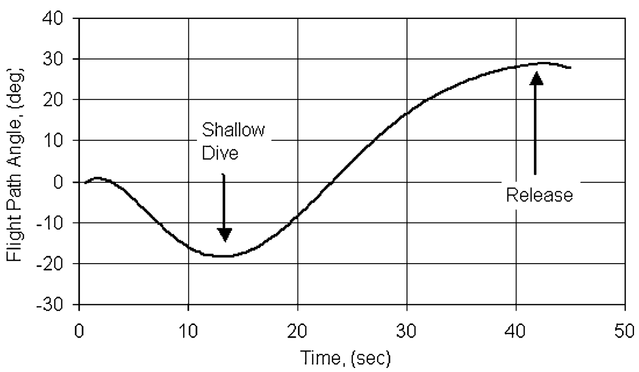

Fig. 7 Separation maneuver: 747 flight-path angle at pull up.

is attainable at a Pc of 1000 psia, a value three times lower than the SSME, which operates at chamber pressures above 3000 psia and thus requires a highly stressing, staged combustion cycle. The lower Pc of the fluorine engine would allow for a simpler, more reliable engine cycle. Reference 19, a textbook on rocket propulsion development history, devotes half a chapter to the advantages of fluorine oxidizers for nonbooster applications.

Payload Capability to Orbit with 747

Payload to altitude capability for a modified Boeing 747 aircraft is approximately 315,000 lb to a 30,000 ft/Mach 0.75 condition. This capability is representative of a modified 747-400F with a top-mounted payload, passenger accommodations removed, and structural stiffing added, as is presently the case for the Shuttle ferry aircraft. The 747-400F presently utilizes four GE CF6 turbofan engines of approximately 62,000 lb of thrust per engine. By uprating the 747 with more powerful GE90-115 engines (115,000-lb thrust each), and adding a larger wing and additional structural stiffening, the 747 air-lift capability might be further increased above the nominal 315,000 lb. A testbed 747 aircraft has already flown with a single GE90-115 engine and three standard CF6 engines. Rocket data presented herein will be for the present GE CF6 engine configuration, however. After takeoff, the 747/rocket combination climbs to above 30,000 ft, by which time preparation of the rocket systems is completed. The 747 then enters a shallow dive and then pulls up to release the rocket. The preseparation maneuver (Fig. 7) begins with level horizontal flight at Mach 0.70. Based on 747 aerodynamic data, the thrust required to maintain this cruise condition is about 51,000 lb (Ref. 20), which is less than the maximum all-engine climb thrust available of about 58,000 lb, giving a marginal rate-of-climb capability. The 747 dives at about 0.25 g, beginning to pull up when the Mach number reaches 0.75, so that the Mach number at the bottom of the dive will be about 0.80. The pull up continues at buffet onset conditions until the desired positive flight-path angle for launch is reached (30 deg), at which time the 747 pushes over to zero (or slightly negative) g for rocket release.[20] The 747 continues to dive until it is well clear of the rocket. The release occurs at Mach 0.75. Separation is done with the main engines at reduced throttle.

Table 3 Air-launch vehicle data for 28-deg, 100-n mile LEO mission, lb

	Subsonic		Supersonic		All rocket TSTO	
Carrier aircraft	Boeing 747	Antonov 225	XB-70 type	New design	Ground launched	
Engine	Turbofan GE CF-6	Turbofan Lotarev D18	Turbojet YJ93-GE	Ramjet MA145-XA		
Separation	Mach 0.75, 30,000 ft	Mach 0.75, 38,000 ft	Mach 3.1, 73,000 ft	Mach 6.0, 85,000 ft	Mach 6.3, 192,000 ft	
Max take off weight	875,000 lb	1,320,000 lb	Undefined	Undefined		
Lift capability	315,000 lb	510,000 lb	343,000 lb	263,000 lb		
					Booster	Orbiter
Rocket						
Delta-velocity, ft/s	27,926	27,649	23,644	20,182	9,383	20,775
Propellant type	F_2/Li-H_2	F_2/Li-H_2	F_2/Li-H_2	F_2/Li-H_2	O_2/RP-1	O_2/H_2
Engine type	Adv chem	Adv chem	Adv chem	Adv chem	RS-84	StagedC
I_{sp} vac/traj avg, s	523/511	523/514	523/521	523/522	326/315	446/431
Chamber press, psia	1,000	1,000	1,000	1,000	2,650	2,800
Mixture ratio	1.15	1.15	1.15	1.15	2.70	6.9 b/6.0
Thrust vac, lbf	246k × 2	265k × 3	236k × 2	163k × 2	1,113k × 4	350k × 3
Rocket total weight	314,704	509,870	343,005	262,477	1,914,657	591,402
Propellant (onboard)	256,000	411,450	259,000	183,000	1,603,171	439,814
usable (crossfeed)	——	——	——	——	−207,553	+207,553
usable (through engs)	——	——	——	——	1,375,165	647,368
Main engine cut off wt	59,704	98,420	84,005	79,477	311,486	151,587
Payload pod	6,500	22,000	23,500	23,500	——	23,500
Reserves, residuals	3,607	5,744	3,649	2,940	21,173	7,791
OMS propellant	1,615	2,666	2,273	2,151	——	2,043
RCS propellant	708	1,169	996	778	——	2,222
Fluids, APU propel	944	1,089	1,027	867	1,622	1,297
Booster flyback prop	——	——	——	——	74,750	——
Dry weight	46,330	65,752	52,560	49,241	213,941	114,735
Tank oxidizer, insul.	3,186 Ti	4,342 Ti	3,215 Ti	2,646 Ti	19,854 Al	4,876 Al
Tank fuel, insul.	5,315 Al	7,843 Al	5,383 Al	4,227 Al	8,157 Al	16,012 Al
Propel management	1,251	1,625	1,174	999	7,183	3,551
Structure-forward	1,649	1,582	3,181	5,936	6,851	6,891
Structure-mid	1,436	2,178	2,210	2,215	9,478	8,188
Structure-aft	2,010	3,577	3,053	2,750	15,653	8,094
Main engines	9,107	13,563	7,984	6,086	47,930	19,516
OMS, RCS hardware	2,355	3,000	2,754	2,676	——	3,746
Power, APU	1,333	1,561	1,371	1,279	1,717	1,594
Distribution, actuation	1,972	2,464	1,983	1,758	2,399	2,349
Avionics, controllers	2,239	2,239	2,239	2,239	1,947	1,947
Wing, aero surfaces	5,492	8,335	7,278	6,952	22,547	13,246
Thermal protection	6,838	9,877	7,656	6,572	6,580	17,081
Landing gear	2,146	3,565	3,080	2,905	9,404	5,660
Truss-booster-orbiter	——	——	——	——	5,342	——
Flyback engs, hardw.	——	——	——	——	46,000	——
Crossfeed hardware	——	——	——	——	1,921	420
Vehicle dia/length, ft	15/93	15/122	15/97	15/79	25/121	25/114
P/L pod dia/length, ft	10/30	15/45	15/45	15/45	——	15/45
Propellant mass fraction	83.1	84.3	81.1	76.6	83.7	77.4

Weights and performance data generated by the vehicle synthesis model are presented in Table 3. The 747/F_2/Li + H_2 rocket system can deliver a payload pod of 6500 lb to the reference 28-deg, 100-n mile circular LEO (column 1 of Table 3). Usable main propulsion ascent propellant is 256,000 lb; reserves, residuals, RCS, OMS, and APU propellants are 6874 lb. Rocket stage dry mass is 46,330 lb. Vehicle length and diameter are 93 and 15 ft, respectively. Two 246,000-lbf-thrust (vacuum) engines produce a vehicle T/W of 1.24 at liftoff (engine T/W is 55). Propellant mass fraction (PMF), defined as usable, main propellant mass divided by total mass (excluding payload), is 83% and payload pod-to-total-weight (P/Wt) ratio is 3%. Rocket delta-V from separation to LEO is 27,926 ft/s. Fully loaded the rocket is 314,704 lb, the maximum lift capability of the CF6 turbofan equipped 747. For an expendable version (no wings, landing gear, or TPS and reduced OMS and RCS), a 20,976-lb payload pod can be delivered (rocket dry weight drops by 14,476 lb). PMF in this case is 87%, and P/Wt is 7%.

Air Launch from a Russian Antonov An-225

A similarly configured rocket with a total weight equal to the lift capability of the world's largest airplane, the Antonov An-225 transport (Fig. 8), was also evaluated. Data for this combination are given in column 2 of Table 3. The An-225 can lift an external payload of 510,000 lb to 38,000 ft/Mach 0.75. Given these parameters, a payload pod of 22,000 lb could be delivered to the reference LEO. Usable propellant is 411,450 lb, rocket dry weight is 65,752 lb, length is 122 ft, and diameter is 15 ft. Three engines of 265,000-lbf vacuum thrust are utilized. The P/Wt ratio is 5%, PMF is 84%, and delta-V to orbit is 27,649 ft/s. A payload of 42,000 lb is achievable if an expendable rocket is utilized. Usable propellant remains about

Fig. 8 Russian Antonov An-225 with Buran orbiter.

the same, and dry mass drops to 46,228 lb. The P/Wt ratio is 8%, and PMF is 88%.

Mach 3.1 Supersonic Carrier for Air Launch

As described previously, a high-Mach, high-altitude condition, previously demonstrated by a large aircraft using conventional turbojet propulsion, was also evaluated. The XB-70 Valkyrie aircraft demonstrated Mach 3.1 flight at 73,000 ft (12 n miles) without supersonic combustion ramjet (scramjet), rocket-based combined cycle (RBCC), or turbine based combined cycle (TBCC) airbreathing propulsion elements. The XB-70, first flown in 1964, is shown in Figs. 9 and 10. The large Russian Sukhoi T-4 achieved similar performance levels in the same time frame. The XB-70 utilized six YJ93-GE-3 turbojet engines designed to cruise at Mach 3 (Fig. 11); a single engine could be changed out by technicians in less than 30 min. Representative illustrations of a conceptual, high-Mach aircraft with a rocket payload are given in Fig. 12 and 13. Large Mach 2+ capable turbojet military aircraft have been operational for four decades, and this aircraft would represent a size increase over these systems; no new technology development would be required for the aircraft. The supersonic carrier concept has not been defined beyond a requirement for a 343,000-lb lift capability to the Mach 3.1/73,000-ft release condition; actual design analyses were not done for the aircraft. The supersonic carrier would be configured so that the rocket would be partially immersed (slotted) into its back to reduce transonic drag. The high-altitude, low-density, low-Q (539-lb/ft^2) release-point simplifies separation aerodynamics.

Performance data are given in Table 3, column 3; a payload pod of 23,500 lb is delivered to the reference orbit by a 343,000-lb winged, reusable rocket of the type described earlier. Rocket usable propellant is 259,000 lb, the sum of reserves; OMS, RCS, and APU propellants is 8400 lb; and dry weight is 52,560 lb. Rocket length is 97 ft, and diameter is 15 ft. Two engines of 236,000-lbf vacuum thrust are utilized. Of the 23,644-ft/s delta-V, gravity, drag, thrust alignment, and nozzle losses account for 2430 ft/s, or about 10%. Internal tank pressures are 32 and 48 psia for the oxidizer and fuel, respectively. PMF is 81%, and P/Wt is 7%. Expendable rocket payload increases to 37,900 lb for the same 343,000-lb initial weight. PMF and P/Wt ratios increase to 85% and 11%, respectively.

A similarly configured, 462-s I_{sp} O_2/H_2 rocket, launched from the same condition, with an equivalent payload was generated to compare against the fluorine system. Its weight is 490,200 lb, 147,000 lb more than the fluorine rocket. Usable propellant for this O_2/H_2 alternative is 390,000 lb, the sum of reserves; OMS, RCS, and APU propellant is 10,300 lb and dry weight is 66,400 lb. Two staged-combustion engines of 275,000-lbf vacuum thrust, T/W = 55, and Pc of 2800 psia are utilized. If a forward-placed, H_2 drop tank partially reusable configuration is utilized, rocket mass is reduced to 460,000 lb. In this case the H_2 tank would be jettisoned after MECO, allowing reductions in TPS, wing, and landing gear weight. Rocket gross weight at ignition is plotted against PMF and I_{sp} in Fig. 14 for both the F_2 and O_2/H_2 variants.

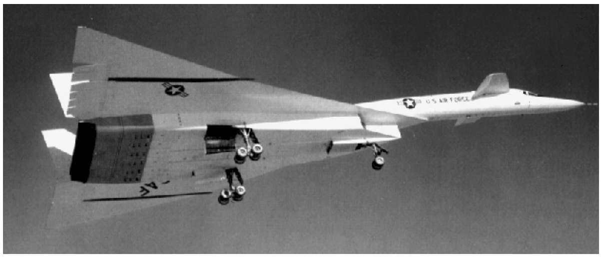

Fig. 9 High-Mach, high-altitude XB-70 (1964).

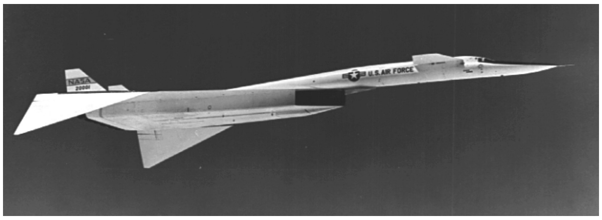

Fig. 10 XB-70: Mach 3.1 with conventional turbojet propulsion.

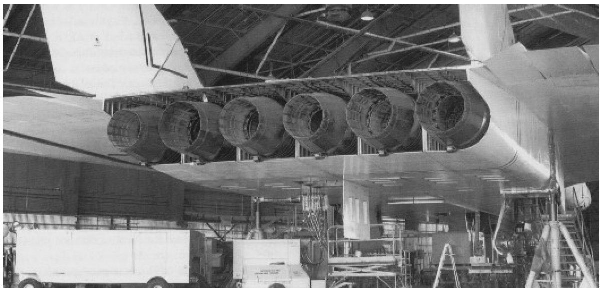

Fig. 11 YJ93-GE engines of XB-70.

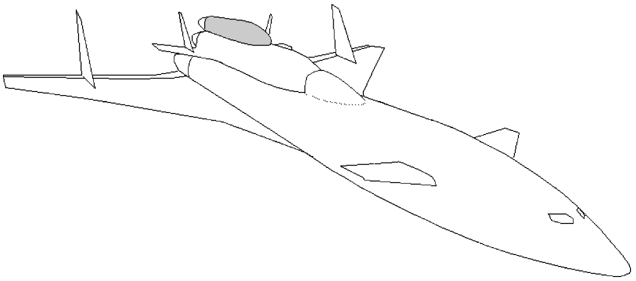

Fig. 12 Conceptual Mach 3+ air-launch system.

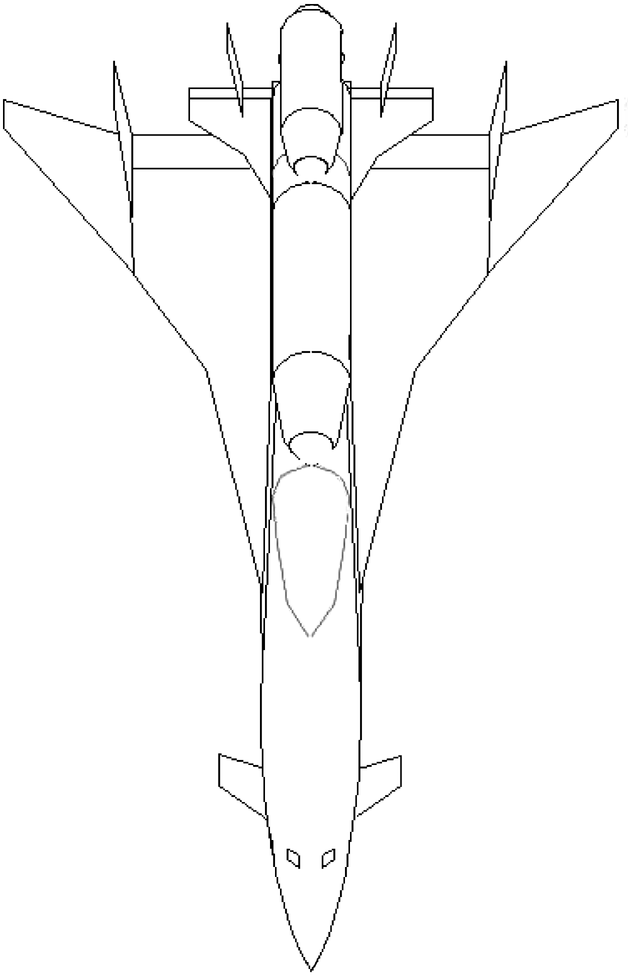

Fig. 13 Mach 3+ air-launch system: top view.

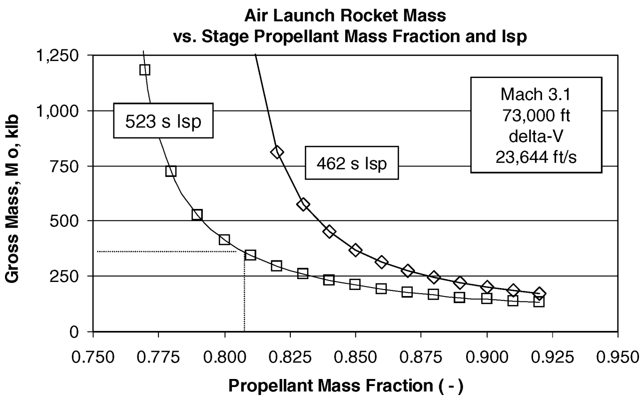

Fig. 14 Rocket mass as a function of I_{sp} and PMF: Mach 3.1.

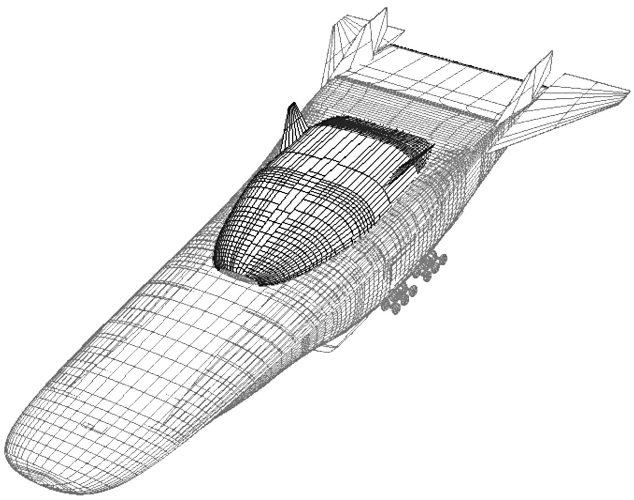

Fig. 15 Conceptual Mach 8+ hypersonic vehicle.

Fully integrated scramjet/rocket combined cycle systems have yet to be demonstrated in flight; Mach 8–12 hypersonic vehicles typically require large, variable geometry inlets and complex, structurally inefficient noncylindrical airframes that require conformal cryogenic tankage. A conceptual Mach 8+ hypersonic TSTO vehicle configuration, typical of the noncylindrical airframe archetype, currently under study by NASA, is represented in Fig. 15. In place of this, the less complex and less costly Mach 3 turbojet carrier aircraft, described herein, could be designed for subsequent refitting to accommodate ramjet propulsion to raise its Mach capability without changing its cylindrical configuration. The aircraft could expand its flight envelope gradually, facilitating the lower risk, test-as-you-go philosophy that prevailed at NASA in the 1960s.

Mach 6 Supersonic Carrier

Full utilization of ramjet propulsion would allow for a doubling of flight velocity to Mach 6, reducing by a further 3500 ft/s the delta-V required of the rocket (from 23,644 to 20,182 ft/s), allowing the entire system to be downsized while retaining equivalent payload capability. The hydrogen-fueled, Marquardt MA145-XAB ramjet engine underwent a series of ground tests in 1968. A planned uprated version of this engine (60-in.-diam) was rated for 161,000 lb of thrust at an engine T/W ratio of about 50 installed. Government funding in support of the technology development and demonstration project totaled \$5.8 million (\$40 million when converted to today's dollars). This effort established the current state of the art in hydrogen-fueled high-Mach-number subsonic combustion ramjet engines. The engine was actively fuel cooled. A total on-test duration for the engine series was about 3 h. Performance for a reusable rocket with a separation condition uprated to Mach 6.0/85,000 ft is given in Table 3, column 4; a 23,500-lb payload pod is delivered to the reference orbit with a F_2 rocket of only 262,477 lb. Propellant

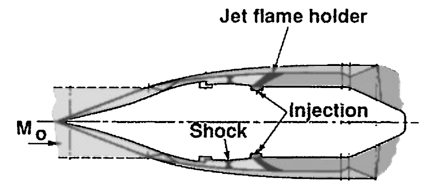

Fig. 16 Ramjet illustration.

and dry mass are 183,000 and 49,241 lb, respectively. Two 163,000-lbf vacuum thrust engines are utilized. PMF is 77%, and P/Wt is 9%. Payload increases to 42,168 lb for the same 262,477-lb initial weight for an expendable version. PMF and P/Wt ratios increase to 83% and 16%, respectively. A representative ramjet diagram is given in Fig. 16.

Comparative Ground-Launched TSTO

A ground-launched, all-rocket TSTO vehicle was generated for comparison purposes: total gross liftoff weight is 2,506,000 lb to accomplish the reference 23,500 lb to LEO mission. Data are given in the last two columns of Table 3. The vehicle's first, or boost, stage weighs nearly two million pounds (1,914,000 lb). Four 1,113,000-lbf-thrust, O_2/hydrocarbon (RP) RS-84 engines, characterized by an I_{sp} of 326 s, a T/W of 89, and a Pc of 2650 psia, are used. (Just the RS-84 engines alone weigh as much as the entire dry weight of the Mach 6 air-launched rocket detailed previously.) The RS-84 is presently in an early stage of development. Booster dry mass is 214,000 lb, flyback propellant is 74,750 lb, and reserves and APU propellants are 24,000 lb. Booster length is 121 ft and diameter 25 ft. TPS tile density for the booster is 0.5 lb/ft^2.

The second stage (orbiter) weighs 591,400 lb, and three 350,000-lbf-thrust, 446-s I_{sp}, staged combustion O_2/H_2 engines are utilized. Engine T/W is 55 and Pc = 2800 psia. Orbiter dry weight is 114,800 lb exclusive of the payload pod, which is externally mounted; MECO weight is 151,600 lb; length and diameter are 113 and 25 ft, respectively. Single engine out is carried on both stages. Al–Li tanks are utilized, as are composite materials for much of the interstage structure. Internal tank pressures are 28, 32, and 48 psia for the RP, O_2 and H_2 tanks, respectively; likewise, tank insulation thicknesses are 0.5, 1.1 and 1.1 in. TPS tile density for the orbiter is 1.0 and 2.0 lb/ft^2 for top side and bottom side locations, respectively; 207,600 lb of O_2, held in the booster, is crossfed to the orbiter during the boost phase so that the latter's O_2 tank is full at separation. Second stage T/W at separation is 1.11 and stack T/W at liftoff is 1.48. The flyback booster returns using subsonic turbojets burning hydrocarbon fuel. The O_2/H_2 second stage operates at a mixture ratio of 6.9 for boost phase flight and 6.0 thereafter. (Vacuum I_{sp} at 6.9 mixture ratio is 440 s.) Design case delta-V is 30,158 ft/s; losses account for 5928 ft/s: large, ground-launched systems inherently incur significant gravity, drag, and nozzle losses ascending vertically through the high density portion of early flight. The trajectory simulation throttles booster engines early to remain under a prescribed max Q; later throtting limits maximum acceleration to 3.0 gs. Booster/orbiter delta-V split is 9383/20,770 ft/s; the booster stages at Mach 6.3 and 192,000 ft.

Simulation Tool

The Boeing Launch Vehicle Design Code (LVDC) was used to determine the payload to orbit capability for the air-launched and ground-launched rocket presented here. LVDC is a multidisciplinary, modularly structured software tool for conceptual design, analysis, and evaluation of a broad range of future launch systems. LVDC uses specialized conceptual design programs; these engineering software tools are linked in a modular fashion to form an iterative computational chain for designing and optimizing a variety of vehicle types. Program sizing and configuration algorithms are coupled with flight routines to simulate propulsive and aerodynamic flight to orbit. The combination of the vehicle's thrust profile, aerodynamic behavior, structural content, and configuration elements forms an internally consistent blueprint for a launch vehicle concept. The aerodynamics module relies on a blend of simplified aerodynamic theory and empirical relationships, which result in acceptable agreement with wind-tunnel test data. The subprogram generates a table of axial and normal aerodynamic force coefficients as a function of Mach number and angle of attack based on airframe geometry determined in the airframe/subcomponents system module. The propulsion module computes composite I_{sp} as a function of throttle, altitude, mixture ratio, nozzle exit area, and chamber pressure. The trajectory is integrated, using the actual I_{sp}, thrust, flow rate, and mass at every altitude to get the correct velocity gained. Lift, drag, dynamic pressure, axial forces on the vehicle, and a number of other parameters are calculated for each point in the trajectory. [The integrated trajectory module set is a simplified version of the industry standard Optimal Trajectory via Implicit Simulation (OTIS) code developed by Boeing in the 1980s.] Flight axial acceleration, thrust, and drag loads are computed in the trajectory simulation for each load-bearing element and passed to the structural routines where required cylindrical tank and interstage wall thicknesses are determined. Tank weights are determined not by a generic structural weight-to-volume ratio but by the actual maximum flight induced structural loading seen in worst-case operation, including propellant hydrodynamic loading on the tank domes. LVDC was developed on a Digital VAX starting in the late 1980s by teams of technologists with expertise in structures, aerodynamics, trajectory simulation, propulsion, materials, power, and other disciplines. The tool was subsequently calibrated, refined, and updated in the course of numerous advanced launch vehicle studies throughout the 1990s and has now been ported to the PC where it runs in the Visual FORTRAN Development Environment. LVDC integrates into one package all of the elements necessary for comprehensive launch vehicle conceptual design.

Conclusions

Any new development of a large carrier aircraft exclusively for air launch would be expensive; nevertheless, subsonic aircraft with lift capabilities comparable to the An-225 could enable 20,000+-lb-class payload missions to LEO with reusable $F_2/Li + H_2$ rockets of on the order of 500,000 lb. Adequately sized turbojet, and turbojet/ramjet supersonic aircraft, could deliver equivalent payloads with reusable rockets of 340,000 and 262,000 lb, respectively. Comparable reusable, ground-launched, all-rocket O_2/RP-1 booster, O_2/H_2 orbiter TSTO vehicles would weigh almost an order of magnitude as much, on the order of 2,500,000 lb, while requiring high Pc (>2000 psia) staged-combustion engines. The 10-fold reduction in total rocket mass is primarily due to the exponential effects of a 10,000-ft/s reduction in delta-V and 70-s increase in I_{sp}. Though a high-Mach aircraft would be more expensive to operate than an airliner, compared to a 2-million-lb flyback rocket booster it could be operated relatively inexpensively and could be turned around for its next flight in a matter of hours, rather than weeks. A conventional Mach 3+ turbojet aircraft, once operational, could be fitted to accommodate ramjets, expanding its flight envelope in a gradual fashion, following a lower risk, test-as-you-go philosophy. This evolutionary approach to building capability has much historical precedent. Eventually, the aircraft could be fitted with experimental SCRAM and combined cycle propulsion elements to serve as a full-size technology test bed; these complex propulsion systems are very difficult if not impossible to ground test.

Building an interim, conventional cylindrical airframe Mach 3+ capable air-launcher and test bed aircraft would allow technology advancement to be carried out in a gradual, evolutionary way: flying, testing, and progressing in measured steps suited to a limited fiscal environment.

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