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J. Weglian, J. Olds, L. Marcus, J. McIntire, D. Nelson
Space Systems Design Laboratory
Georgia Institute of Technology
Atlanta, GA

J. Blevins
NASA MSFC
Huntsville, AL

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ASPEN Revisited: The Challenge of Nuclear Propulsion for ETO

John E. Weglian*, John R. Olds†, Leland Marcus*, James McIntire*, Douglas Nelson‡
Space Systems Design Lab
School of Aerospace Engineering
Georgia Institute of Technology, Atlanta, GA, 30332-0150
(404) 894-7783 Fax: (404) 894-2760
John Blevins§
NASA MSFC
Huntsville, AL 35812
(256) 544-3705 Fax: (256) 544-5926

ABSTRACT

ASPEN was a study conducted by Los Alamos National Labs in the early 1960's to examine the benefits of using a Nuclear Thermal Rocket (NTR) for Earth-to-Orbit (ETO) single-stage launch vehicle applications. Using the analysis methods and assumptions of the time, this formerly classified study showed that a significant performance potential might be derived from using NTR engines for the final acceleration phase to orbit (air-breathing engines were used to Mach 11). Given the increased NASA interest in low-cost reusable space transportation, the ASPEN concept has been revisited using contemporary design assumptions and conceptual analysis techniques.

The present analysis concludes with a more pessimistic view of NTR propulsion for ETO applications. Aerodynamic drag for the ASPEN configuration was found to be significantly more than that calculated in the original study. The resultant vehicle thrust-to-drag ratio is lower than necessary for high acceleration during the air-breathing acceleration phases. In addition, the NTR reactor power requirements are daunting. In most cases, reactor powers over 10 GW are required. Even with very aggressive assumptions (25% drag reduction and NTR thrust-to-weight ratio of 10 including shielding) a 500,000 lb gross weight ASPEN-like vehicle was found to only have a payload mass fraction of 1.6%. This is significantly less than the 6% to 15% payload mass fractions claimed in the original ASPEN study. Political issues aside, the engineering aspects for using NTR in an ETO system are extremely daunting and are not expected to be achievable in the foreseeable future.

* - *Graduate Research Assistant, School of Aerospace Engineering, Student member AIAA.*
†- *Assistant Professor, School of Aerospace Engineering, Senior member AIAA.*
‡ - *Graduate Research Assistant, School of Aerospace Engineering, member AIAA.*
§ - *Aerospace Engineer, member AIAA*

NOMENCLATURE

ε	Nozzle Expansion Ratio
A_e	Exit Area
APAS	Aerodynamic Preliminary Analysis System
c_P	Specific Heat at Constant Pressure
ETO	Earth to Orbit
GLOW	Gross Liftoff Weight
HABP	Hypersonic Arbitrary Body Program
$\overline{I_{SP}}$	Trajectory-Averaged Propulsive Specific impulse
I^*	Equivalent Effective Specific Impulse (sec)
I_{SP}	Specific Impulse (sec)
LH2	Liquid Hydrogen
LOX	Liquid Oxygen
$\dot{m}$	Mass Flow Rate
MR	Mass Ratio (initial mass/final mass)
NTR	Nuclear Thermal Rocket
P	Reactor Power
P_a	Ambient Pressure
P_e	Exit Pressure

POST	Program to Optimize Simulated Trajectories
RLV	Reusable Launch Vehicle
SCCREAM	Simulated Combined-Cycle Rocket Engine Analysis Module
SSME	Space Shuttle Main Engine
SSTO	Single Stage to Orbit
T	Thrust
T_{in}	Reactor Inlet Temperature
T_{out}	Reactor Outlet Temperature
TRF	Technology Reduction Factor
T/W	Thrust to Weight of Vehicle
T/W_e	Thrust to Weight of Engine (including shielding)
UDP	Unified Distributed Panel
v_e	Exit Velocity
WBS	Weight Breakdown Structure

INTRODUCTION

Single-stage-to-orbit (SSTO) reusable launch vehicles (RLVs) are desirable for their potential to achieve low cost operations, reduced turnaround times, and reduced procurement costs. However, SSTO vehicles require a very high propellant mass fraction to reach orbit. SSTO RLVs powered by LOX/LH2 chemical rockets might require propellant mass fractions on the order of 87% - 90% (hydrocarbon fueled rockets will be even higher). The remaining mass must be split between inert mass (structure, engines, thermal protections, subsystems, gear) and the payload. For an SSTO RLV powered by chemical rockets, payload mass fractions on the order of only 1% - 2% are typical.

If a high specific impulse, high thrust engine like a Nuclear Thermal Rocket can be used instead of a traditional chemical propulsion system, the theoretical gains can be dramatic. Figure 1 shows the improvement in propellant mass fraction with increasing I_{sp} according to the well-known Rocket Equation for a constant V. For a total ΔV of 30,000 ft/sec, if the I_{SP} of the engine can be improved from 450 sec to 1000 sec, the propellant mass fraction improves from 0.874 to 0.606. This means that an additional 27% of the grow liftoff weight would be available for structural and payload mass.

While this simplified analysis highlights the potential of a high I_{sp} launch system and provides the motivation for the current study, a true multidisciplinary conceptual analysis is required to fully determine the overall effects of introducing an advanced propulsion system. In the case of an NTR system for example, the reactor and shielding mass might offset any reduction in propellant savings. In addition, the reduced propellant bulk density for a vehicle dominated by liquid hydrogen tends to increase the required tank volume and weight relative to traditional bi-propellant rocket vehicles. Modern conceptual design tools and methods are capable of resolving these complex interactions.

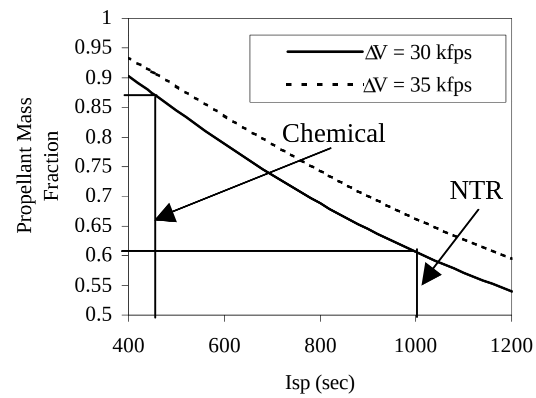

Figure 1. Propellant Mass Fraction vs. I_{SP}

ASPEN BACKGROUND

In the 1950s and 1960s, many potential uses of nuclear power were explored, such as NTRs. An NTR uses a nuclear reactor to heat a working fluid, such as hydrogen, and exhaust this fluid in a nozzle for thrust. Many studies of this technology were conducted, and prototypes were built and tested [1,2]. A companion paper to this paper, Reference 3, gives a detailed description of the history of NTR development.

ASPEN (Figure 2) was originally a classified study conducted at the Los Alamos Scientific Laboratory in 1961 by R. W. Bussard. It was an SSTO vehicle that used turbojets and ramjets followed by an NTR to provide the final boost into orbit. The study was declassified and approved for public release in 1995.

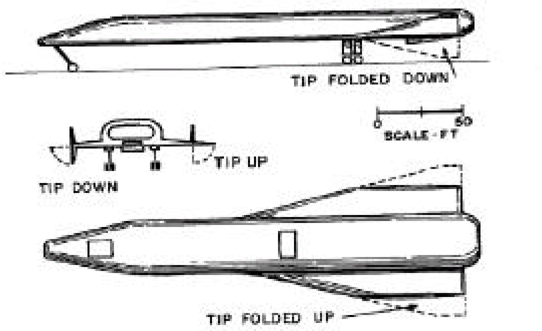

Figure 2. Original ASPEN vehicle [1]

ASPEN Reactor Shielding

Shielding is a major concern for nuclear reactors, and is of particular interest in aerospace applications due to the large weight. In a crewed vehicle, the radiation dose must be limited to certain levels for crew safety. Payloads may also have shielding requirements, but these are usually much less stringent and can usually be included in the payload design.

Terrestrial reactors typically completely surround the reactor with several inches of lead and other shielding materials. In order to minimize the shielding weight for space applications, it is desirable to use a shadow shield, which only shields radiation in one direction. This technique places a shield between the reactor and the crew, but does not completely surround the reactor. With this configuration, the portion of the vehicle aft of the shield would become radioactive due to the neutron flux while the reactor is at power. The effect of this on operations was not analyzed in this study.

ASPEN Configuration

ASPEN is a traditional wing-body vehicle with a high fineness ratio. Small turbojet and ramjet inlets are integrated onto the underside of the fuselage. The NTR engines are placed on the aft base area of the fuselage. An internal payload bay is provided between large internal hydrogen (LH2) tanks. A crew cabin is placed near the nose of the vehicle. ASPEN takes off and lands horizontally on retractable landing gear.

ASPEN Mission

The original ASPEN vehicle used LH2 turbojets for take-off and to accelerate to Mach 2.5 at 60,000 ft. At this point, the subsonic combustion ramjet engines would accelerate it to Mach 11 at 120,000 ft. (Note: contemporary wisdom would include a switch to supersonic combustion scramjets above Mach 5 or 6. Conceptual design tools now available were not able to duplicate the ASPEN ramjet performance to Mach 11.). From Mach 11, the NTR would provide the final acceleration to put the vehicle in an orbit with perigee of 80 nmi and apogee of 300 nmi. Radiation can scatter off the atmosphere and reach the crew even if no direct path from the reactor to the crew is left unshielded. For this reason, an NTR powered vehicle should not bring the reactor up to full power until it is high in the atmosphere in order to reduce shielding requirements.

Figure 3 shows the proposed trajectory for the original ASPEN vehicle. The flight corridor shown diverging below the main corridor above 12,000 ft/sec flight speed in Figure 3 was a proposed flight corridor for a NASP-like vehicle that accelerated to orbital velocity using only air-breathing engines. As the NASP program later showed, acceleration to Mach 25 using only air-breathing propulsion is now considered highly unlikely.

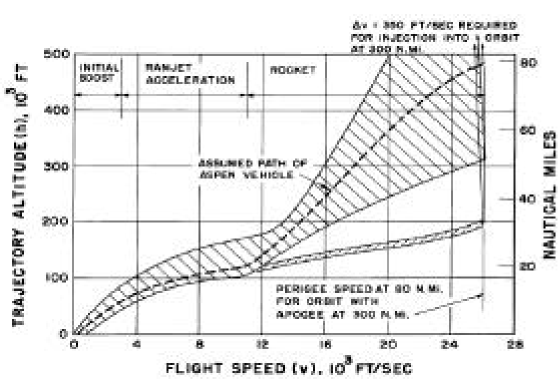

Figure 3. ASPEN Flight Trajectory [1]

The I_{SP} assumed for the original ASPEN vehicle is shown in Figure 4. The supersonic burning region shows the projected performance of a scramjet engine, but the ASPEN concept utilized subsonic combustion ramjets to Mach 11, as previously noted.

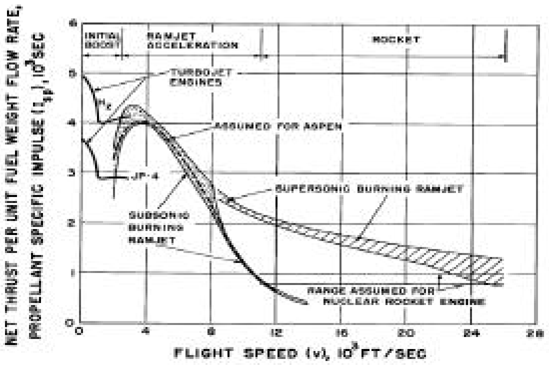

Figure 4. ASPEN I_{SP} vs. Flight Speed [1]

The thrust-to-weight and drag-to-thrust ratios from the original analysis are shown in Figure 5. The vehicle thrust-to-weight (T/W) is 0.3 at takeoff, which equates to 150,000 lbs. of turbojet thrust. At rocket takeover, the T/W is 0.6 and the vehicle is about 82% of GLOW, which equates to about 245,000 lbs. of NTR thrust. Note that the predicted D/F of only 0.2 between Mach 6 and Mach 11 (thrust-to-drag ratio of 5) must be viewed with some degree of skepticism. Even well designed contemporary air-breathing RLV configurations are unable to achieve this level of acceleration at higher Mach numbers.

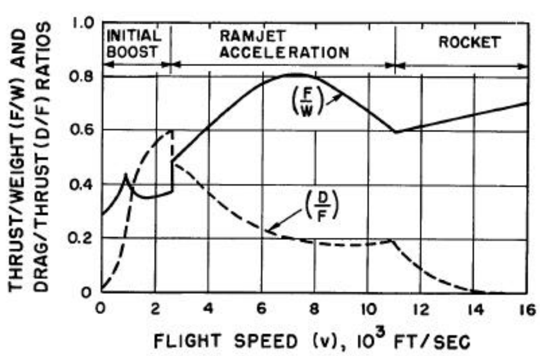

Figure 5. ASPEN Thrust-to-Weight and Drag-to-Thrust Ratios [1]

ASPEN Sizing Results

The ASPEN vehicle was designed to carry two passengers into orbit and remain in orbit for two days. The vehicle design parameters are listed in Table 1. The original payloads for the ASPEN vehicle are presented in Table 2. The vehicle was sized for a constant gross weight of 500,000 lb. The payload was allowed to vary in order to close the vehicle. As a baseline case, an NTR I_{SP} of 800 sec was used. The report also considers an improvement in NTR I_{SP} to 1000 sec. If the air-breathing engines could be replaced with "advanced chemical engines", the orbital payload with an NTR I_{SP} of 1000 sec increases to 80,000 lbs to a 300 nmi equatorial orbit. Payload mass fractions for the original ASPEN vehicle range from 6% - 16% for an equatorial orbit. By comparison, contemporary SSTO designs using traditional chemical rockets have propellant mass fractions of only 1% - 2%.

Table 1. ASPEN Design Parameters [1]

GLOW (lbs)	500,000
Orbit Altitude (nmi)	300
Reactor Power (MW)	4.9

Table 2. ASPEN Payload Capabilities (lbs) [1]

Destination Orbit	NTR I_{SP} 800 sec	NTR I_{SP} 1000 sec	Advanced Air-breathing
Polar	20,000	50,000	70,000
Equatorial	30,000	60,000	80,000

CURRENT DESIGN PROCESS

Given the original ASPEN configuration, mission, and payload results as a starting point, the goal of this study was to reexamine this configuration using contemporary methods and assumptions. It is recognized that the application of NTR propulsion to ETO is still not a "near-term" possibility. Therefore the technology assumptions made for this revisit are appropriate for a vehicle that would be deployed in the 2025 – 2040 timeframe (Gen3 or Gen4 RLV). Advanced metal matrix composites are used for the airframe and undercarriage structure. Advanced metal and ceramic materials are used for the turbojet and high-speed air-breathing engines. Liquid hydrogen propellant tanks are assumed to be constructed of lightweight graphite composites. Where needed, thermal protection materials are assumed to be highly durable ceramic tiles and blankets. Electromechanical actuators rather than hydraulics are baselined for surface and engine actuators. Lightweight power and other supporting subsystems are also assumed.

One of the most critical assumptions for a nuclear-based propulsion system is the installed weight of the reactor and its associated shielding. Counting reactor weight, shielding weight, nozzle skirt weight, and powerhead weight, the installed thrust-to-engine weight ratio (T/W_e) of a high power NTR is typically around 3 – 5. In this revisit of ASPEN, the T/W_e of the NTR engine was allowed to vary between 5 and 30 in order to quantify the potential gains from NTR weight reduction. While the original ASPEN was sized for a 500,000 lb. gross weight, the current study considers vehicle gross weights of 500,000, 1,000,000, and 2,000,000 lbs.

Specific Impulse

In order to determine a reasonable vacuum I_{sp} for the NTR, some initial baseline calculations were made. The thrust and specific impulse from an NTR is directly related to the hydrogen working fluid's total temperature after leaving the reactor/heat exchanger. For this simplified analysis, an ideal nozzle (nozzle efficiency equal to 1.0) was used. The expansion ratio (77.5) and total pressure (3100 psi) of the Space Shuttle Main Engine (SSME) were used as representative values to determine the I_{sp} for a range of total temperatures from 1000 K to 3500 K. The nozzle's exit temperature, pressure, velocity, and Mach number are easily determined using isentropic equations and assuming a calorically perfect gas [4]. The NTR vacuum thrust and I_{sp} are calculated from these values and the throat area [5]. For direct comparison to the NERVA reactors, the throat area was set to provide a vacuum thrust 200,000 lb. in each case.

The resultant curve is shown in Figure 6. Two reactor designs from the 1960s are included for comparison. The Pewee reactor operated at 2555 K and generated 514 MW, and the NERVA engine for the Rover program operated at 2360 K and generated 1556 MW [2]. An additional reactor type, the particle bed reactor, was investigated by the Timberwind (unclassified name) project, which ended in 1993. This type of reactor used ceramic pellets containing the nuclear fuel, and was designed to operate at a much higher temperature than previous NTR designs. The particle-bed reactor operated at a temperature of 3200 K and produced 1945 MW of power [5]. In order to achieve a vacuum I_{sp} of 1000 sec (equivalent to Bussard's high NTR I_{sp} setting), a reactor outlet temperature of about 3500 K is required. This is certainly challenging, even with advanced technology assumptions, but is believed to be technologically feasible in the timeframe considered. 1000 sec is therefore used for I_{SP} in the rest of the analyses. Higher nozzle expansion ratios could also be used to help achieve this goal since the NTR is operated only above 100,000 ft.

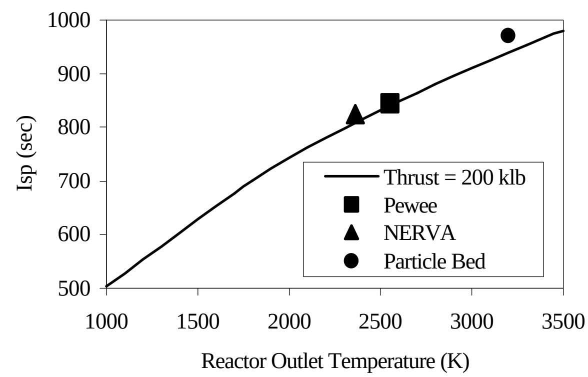

Figure 6. Specific Impulse vs. Reactor Outlet Temperature

Aerodynamics

The original ASPEN configuration was modeled in the Aerodynamic Preliminary Analysis System (APAS) to determine the lift and drag characteristics of the vehicle. In the original design, the wing tips could rotate down to a vertical position for additional stability in the hypersonic regime. This capability was not modeled in the APAS model, and the wingtips were kept horizontal for maximum lift throughout the mission.

The subsonic and low supersonic flight regimes were calculated using the Unified Distributed Panel (UDP) component of APAS. The hypersonic flight regime was calculated using the Hypersonic Arbitrary Body Program (HABP) component of APAS. The lift and drag coefficients, non-dimensionalized by the wing planform area, were tabulated as a function of altitude and Mach number, and this table was used in the subsequent trajectory analysis.

Propulsion

Several difficulties arose in trying to duplicate the air-breathing propulsion assumptions of the original ASPEN study. Using current design tools, it was not possible to get a ramjet to produce positive thrust to Mach 11 with subsonic burning as ASPEN assumed. Another difficulty was that the initial T/W of the vehicle needed to be higher than the 0.3 value used by Bussard in order for the trajectory optimization tool to find a viable solution to get to orbit. That is, the turbojet thrust needed to be increased to get through the transonic regime and on to Mach 2.5. As a result, the takeoff T/W was increased to 0.42.

In order to effectively optimize the trajectory, the engine performance as a function of altitude and Mach number was required. The Simulated Combined-Cycle Rocket Engine Analysis Module (SCCREAM) [6], an RBCC/ramjet/scramjet conceptual design tool developed at Georgia Tech, was used to generate the high-speed air-breathing performance data used in the study. The engine setup parameters used in SCCREAM were chosen to try to closely match the ASPEN performance. Turbojet performance was estimated separately.

For the current analysis, a scramjet mode was required to reach the NTR takeover condition. Thus the engine used in the current study is a dual-mode ramjet/scramjet. The ramjet to scramjet transition occurred between Mach 6.0 and 7.0. Obvious issues related to the airframe integration of the scramjet inlets were not addressed. Thus the scramjet performance for this forebody shape is optimistic. In order to ensure the NTR did not take over below 100,000 ft. altitude, the scramjet was run to Mach 11.5 rather than Bussard's Mach 11, and the scramjet to rocket transition occurs between Mach 11.5 and 12.5.

The thrust produced by the NTR was reduced as much as possible to limit the reactor power and weight required. The trajectory analysis determined that 60% of the GLOW was the minimum value of vacuum thrust at rocket takeover that would still let the vehicle achieve orbit.

Trajectory Analysis

The trajectory analysis consisted of using the Program to Optimize Simulated Trajectories (POST) [7] to calculate the trajectory that minimizes the fuel used to reach orbit. The basic trajectory consisted of a horizontal take-off followed by acceleration onto a dynamic pressure boundary of 1000 psf during the air-breathing modes. Once the vehicle had accelerated to Mach 11.5, the air-breathing propulsion system was shut down and the NTR took over. The notional launch site was assumed to be on the equator, and the vehicle flew due east, the same as the original ASPEN equatorial orbit analysis. Throughout the flight, the trajectory must conform to the following constraints:

1. The maximum dynamic pressure on the vehicle must not exceed 1200 psf.
2. The maximum wing normal force must not exceed two times the initial weight of the vehicle
3. The flight path angle, γ, at orbit insertion must equal zero.
4. The perigee altitude at orbit insertion must equal 50 nmi.
5. The apogee altitude at orbit insertion must equal 300 nmi.

The original ASPEN configuration was found to have a significantly lower thrust-to-drag ratio than that calculated by Bussard. This was due to the combined effect of a larger than expected vehicle drag and a smaller than expected air-breathing thrust. The former is the result of a relatively blunt nose and thick wings. The latter is the result of an insufficient capture area for the air-breathing engines. As a result, the performance of the original configuration was marginal at best.

To gauge the effects of drag on the vehicle, three different aerodynamic configurations were simulated. The first was the baseline case (100% drag). Two additional cases were run simply assuming some unnamed configuration modification could be used to reduce the drag of the vehicle by 15% or 25% of the original drag. For each case, the mass ratio (final mass/initial mass) was calculated.

Table 3. Weight Breakdown Structure in lbs for 1,000,000 lb Vehicle with NTR T/W_e of 10 and 15% Drag Reduction.

Group	Group Name	Weight (lbs)
1.0	Wing Group	39,900
2.0	Tail Group	6,100
3.0	Body Group	73,600
4.0	Thermal Protection	21,000
5.0	Landing/Takeoff Gear	23,900
6.0	Propulsion (All)	120,500
7.0	RCS Propulsion	5,300
8.0	OMS Propulsion	0
9.0	Primary Power	800
10.0	Electrical Conversion & Dist.	4,600
11.0	Hydraulic Systems	0
12.0	Surface Control Actuation	1,900
13.0	Avionics	1,600
14.0	Environmental Control	2,900
15.0	Personnel Equipment	800
16.0	Dry Weight Margin	45,400
	Dry Weight	**348,300**
17.0	Crew and Gear	1,900
18.0	Payload Provisions	0
19.0	Cargo (up and down)	39,200
20.0	Residual Propellants	3,000
21.0	RCS Reserve Propellants	200
	Landed Weight	**392,600**
22.0	Entry/Landing Propellants	800
	Entry Weight	**393,400**
23.0	RCS Propellants (on-orbit)	1,500
24.0	Cargo Discharged	0
25.0	Ascent Reserve and Unusable Propellants	3,000
26.0	In-flight Losses and Vents	3,900
	Insertion Weight	**401,800**
27.0	Ascent Propellants	598,200
	Gross Liftoff Weight	**1,000,000**

Weights and Sizing

The weights and sizing analysis was conducted using Mass Estimating Relationships (MERs) based upon work done at the NASA Langley Research Center in the 1980's and 1990's on air-breathing and rocket RLVs. This MER database includes historical data from the space shuttle and high-speed aircraft as well as numerical analysis of RLV structures using finite element analysis. This database was adjusted as necessary using Technology Reduction Factors (TRFs) to reflect the time frame for the current ASPEN study. The WBS includes a 15% weight margin to account for growth in the mass budget during detailed design of the vehicle.

The NTR thrust-to-weight (T/W_e) was treated parametrically in order to determine the required reactor size without actually designing the reactor itself. The T/W_e would then drive the required reactor design. The T/W_e was set to 5, 10, 20, and 30 in this analysis. This value includes all components associated with the operation of the engine, including the weight of the required shielding and a radiator system to reject decay heat (heat generated from the decay of fission products) once in orbit. No detailed analysis of the shield mass was conducted, but it is worth noting that ASPEN (a 500,000 lb vehicle) assumed 40,000 lbs for its shadow shield.

The required mass ratio for the vehicle was determined by the trajectory optimization, and this analysis assumed that the mass ratio would remain constant as the vehicle was photographically scaled. Three vehicle sizes, 500,000 lbs, 1,000,000 lbs and 2,000,000 lbs GLOW, were investigated. The mass of the payload was allowed to be negative, if required to converge the vehicle. The WBS extended to two levels for all items, and to three levels for some. An example top-level statement for a 1,000,000 lb vehicle with a T/W_e of 10 and 15% drag reduction is given in Table 3.

Reactor Power

A separate trade study was performed to determine the NTR thrust required at NTR takeover. A thrust value of 60% of GLOW was found to give reasonable performance results while minimizing the thrust required. Therefore, the NTR vacuum thrust was set to 60% of GLOW for all three vehicle sizes analyzed in this study. Given the required thrust, the mass flow rate through the nozzle is determined by Equation (1). The ambient pressure at 100,000 ft is very small, and the mass flow rate does not change appreciably from vacuum conditions. This study assumed that the mass flow rate through the reactor was the same as that through the nozzle. The required reactor power is based on the temperature differential across the reactor and the mass flow rate as given in Equation (2). For this calculation, the specific heat was set to the average temperature in the reactor, 33.74 J/mol-K at 1850 K. The reactor inlet temperature was 200 K and the reactor outlet temperature was 3500 K.

$$\dot{m} = \frac{T - (P_e - P_a)A_e}{v_e} \tag{1}$$

$$P = \dot{m}c_P(T_{out} - T_{in}) \tag{2}$$

RESULTS

High drag was a major factor in the performance of the ASPEN vehicle. In order to account for the possibility of significant improvement in the design of the vehicle, drag reduction was treated parametrically. Drag values of 100%, 85%, and 75% of the original drag coefficients were used. In other words, the nominal drag case was compared to a 15% and 25% drag reduction. NTR T/W_e's of 5, 10, 20 and 30 were used, although with current technology, a T/W_e above 10 is probably not feasible.

Table 4 lists the mass ratios determined from the trajectory analysis as well as the ΔV_{total}, trajectory-averaged propulsive specific impulse ($\overline{I_{SP}}$) and equivalent effective specific impulse (I^*). I^* accumulates trajectory losses into an overall effective I_{SP}, and I^* is defined in Reference 8.

Table 4. Mass Ratio for Various Drag Configurations

	Nominal Drag	15% Drag Reduction	25% Drag Reduction
Mass Ratio	3.115	2.489	2.374
ΔV_{total} (ft/sec)	57,600	43,600	40,200
$\overline{I_{SP}}$ (sec)	1508	1480	1440
I^* (sec)	670	840	890

Based on the weights and sizing analysis, the payload required to achieve the proper orbit was determined and is presented in Tables 5, 6, and 7 for the 500,000 lb, 1,000,000 lb, and 2,000,000 lb GLOW vehicles, respectively. Note that in many cases the payload is a negative number. This implies that it is not possible to achieve orbit with that vehicle configuration unless the weight of the structure can be reduced beyond the assumptions made in this study.

Table 5. 500,000 lb Vehicle Payloads

Drag	NTR T/W_e			
Reduction	5	10	20	30
Nominal	-100400	-55000	-32200	-24600
15 %	-48200	-2700	15900	21900
25 %	-35600	7900	25900	32000

Table 6. 1,000,000 lb Vehicle Payloads

Drag	NTR T/W_e			
Reduction	5	10	20	30
Nominal	-143700	-52700	-7200	-6400
15 %	-41600	39200	75300	87400
25 %	-16900	58900	95000	99100

Table 7. 2,000,000 lb Vehicle Payloads

Drag	NTR T/W_e			
Reduction	5	10	20	30
Nominal	-223200	-41400	39300	63400
15 %	-23200	126100	198300	222400
25 %	20100	143500	220600	243000

Figure 7 shows these results for the 500,000 lb GLOW vehicle. For this vehicle, unless drag is reduced, it is not possible to develop a viable vehicle (i.e. a non-negative payload), even with the most ambitious engine thrust to weights. Vehicles with reduced drag become feasible with a T/W_e of 11 for the 15% drag reduction case and 9 for the 25% drag reduction case. ASPEN assumed a T/W_e of 25 for the engine without any shielding, plus an additional 40,000 lbs of shielding. Factoring this weight into the T/W_e for comparison to the results in this paper yields a T/W_e of 4.8. With an NTR I_{SP} of 1000 sec, the ASPEN vehicle had a payload of 60,000 lb in an equatorial orbit. This point is included for reference on Figure 7.

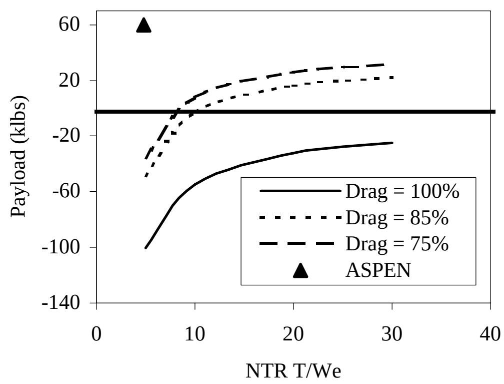

Figure 7. Payload vs. NTR Thrust to Weight for 500,000 lb Vehicle

Figure 8 shows the results for the 1,000,000 lb GLOW vehicle. With the baseline drag, a T/W_e above 24 allows for a viable vehicle. The reduced drag cases allow for a viable vehicle with a T/W_e of 7 for the 15% drag reduction case, and 6 for the 25% drag reduction case.

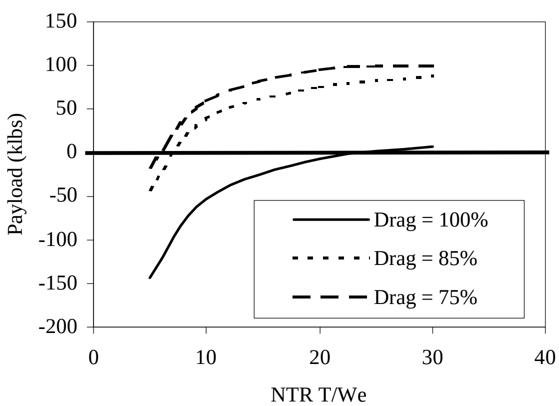

Figure 8. Payload vs. NTR Thrust to Weight for 1,000,000 lb Vehicle

Figure 9 shows the results for the 2,000,000 lb GLOW vehicle. The baseline drag case becomes viable with a T/W_e above 14. With reduced drag, this value drops to 6 for the 15% drag reduction case, and 5 for the 25% drag reduction case.

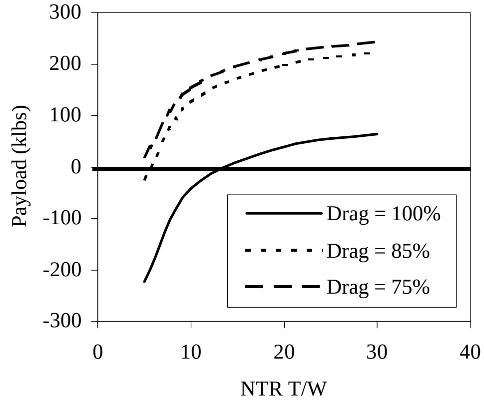

Figure 9. Payload vs. NTR Thrust to Weight for 2,000,000 lb Vehicle

The payload mass fractions (payload weight/GLOW) for each vehicle at a T/W_e of 10 are presented in Table 8. Only the viable solutions (payload mass > 0) are used. As the vehicle gets larger, a larger percentage of the vehicle can be payload for a given drag profile.

Table 8. Payload Mass Fractions at $T/W_e = 10$

GLOW (lbs)	Nominal Drag	15% Drag Reduction	25% Drag Reduction
500,0000	-	-	0.016
1,000,000	-	0.039	0.059
2,000,000	-	0.063	0.077

The reactor power required to achieve the required thrust is shown in Table 9. It is worth noting that the highest power reactor ever operated was the Phoebus 2A, an NTR in the NERVA program tested in 1969. This reactor operated up to 4 GW, although it was designed for 5 GW [2]. Most commercial nuclear power plants operate between 2-3 GW, thermal power. The high power requirements lead to the requirement to reject a large amount of heat once in orbit due to the natural decay of fission products produced during reactor operations. The weight of the heat rejection system was not considered separately in this analysis and must be included in the T/W_e.

Table 9. Reactor Power Requirements

Vehicle Size (lbs)	Reactor Power (GW)
500,000	7.5
1,000,000	15.1
2,000,000	30.2

Shielding calculations were not conducted for this study. The mass of the shielding must be included in the T/W_e to determine the available payload. In the original ASPEN paper, the shield for the 4.9 GW reactor was 40,000 lbs for an engine that produced 245,000 lbs of thrust. Without including the mass of the rest of the engine (estimated at 9800 lb by Bussard), the T/W_e including shielding is at most 6.1. In order to get the T/W_e larger than this value, the shielding weight per GW must be less than that assumed in the original ASPEN study. To further complicate matters, the use of a shadow shield helps to shield the crew and payload, but provides no protection for other vehicles or the space station for rendezvous operations. In order to supply this protection, the entire core must be surrounded with shielding, although shielding requirements after shutdown are much less than while at power. Based on this, it is not likely that an engine with a T/W_e greater than 5 could be developed without a revolution in shielding materials.

CONCLUSIONS

The high T/W requirement for ETO missions leads to the difficulty in using an NTR for ETO applications. In order to reach orbit, the thrust must be a significant fraction of the GLOW of the vehicle. The power required from the reactor is set by the required thrust and temperature differential across the reactor. For a small vehicle, the payloads are too small to make the vehicle useful. For a large vehicle, the power requirement from the reactor is extremely large compared to any existing nuclear reactors.

Very aggressive assumptions were made in this analysis, such as an I_{SP} of 1000 sec, in order to see what could be accomplished at the limits of current technology. However, the technical challenges associated with developing a very high power and yet lightweight NTR system that is viable for an ETO vehicle are beyond today's capabilities. The power required to accelerate a large launch vehicle into orbit is very high: over seven times larger than the highest powered nuclear reactor ever built for a 2,000,000 lb vehicle. Until the technology exists to eliminate or mitigate these difficulties, an NTR system does not make an attractive candidate for ETO applications.

An air-breathing vehicle that minimizes drag is very important to maximize the payload a given vehicle can get to orbit. Also, larger vehicles can bear the weight of the NTR system much better than a smaller vehicle, but this results in a higher-powered nuclear reactor.

ETO missions require a high T/W in rocket mode to reach orbit. This leads to the high reactor powers and high T/W_e requirements for vehicles that can get any payloads into orbit. An in-space vehicle would not suffer from the high T/W requirement; so much lower thrust values could be used, which would lead to much lower reactor powers as well. For this reason, an NTR may provide a very attractive propulsion system for in-space vehicles.

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