ISOMER HEAT EXCHANGER COMBUSTOR REPLACEMENT FOR A SUPERSONIC RAMJET POWERED VEHICLE

THESIS

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AFIT/GAE/ENY/04-J02

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Abstract

This study investigated the possibility of utilizing a Triggered Isomer Heat Exchanger (TIHE) within a ramjet engine to power a supersonic Conventional Air Launched Cruise Missile (CALCM). A computational ramjet engine was created using engine analysis software. This model was then run through a simulated cruise missile mission using AEDsys, a mission analysis and engine performance software suite. The replacement of the combustor with a TIHE led to an impressive range increase for the supersonic cruise missile.

Acknowledgements

I would first like to thank Dr. King for his direction and motivation in this project. From the beginning, he has helped to shape the course of this research. I appreciate his patience and help throughout this process.

Social support was essential to my ability to finish this research. My fellow Navy IGEP students provided camaraderie and helped me focus on finishing. My friends and family were always there to provide encouragement and gave me an outlet when it was necessary. I would have not been able to complete this project without all of their support.

Jonathan Cox

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List of Symbols

Symbol	Description
β	Weight Fraction
ρ	Density (lbm/ft^3)
η_b	Burner efficiency
$\prod_i$	Current Weight Fraction
A	Cross Sectional Area (ft^2)
AEC	Atomic Energy Commission
AGM	Air-Ground Missile
ANP	Aircraft Nuclear Propulsion program
CALCM	Conventional Air Launched Cruise Missile
CFD	Computational Flow Dynamics
C_D	Drag Coefficient
C_L	Lift Coefficient
D	Drag (lbf)
eV	Electron Volt
F	Uninstalled Thrust (lbf)
g_o	Acceleration of Gravity (ft/sec^2)
g_c	Newton's Constant
Hf	Hafnium
HTRE	Heat Transfer Reactor Experiment
K'	Induced drag
K''	Skin friction and Pressure Drag
L	Lift (lbf)
lbf	Pounds Force
lbm	Pounds Mass
Lu	Lutetium
M	Mach number
$\dot{m}_f$	Mass Flow Rate of Fuel (lbm/sec)
n	Load Factor
NEPA	Nuclear Energy for the Propulsion of Aircraft Program
nm	Nautical miles
NTA	Nuclear Test Aircraft
q	Dynamic Pressure (lbf/ft^2)
Q_{burner}	Heat power required by TIHE (BTU/sec) or (W)
r	Radius (ft)
R	Additional Drag due to Stores or Landing Gear
RPM	Rotations Per Minute
S_{wing}	Wing Area (ft^2)
S	Unistalled Thrust Specific Fuel Consumption (1/hr)
SI	International System of Units
T	Thrust (lbf)
t	Time (sec)

Ta	Tantalum
T_{T4}	Turbine Inlet Temperature (R)
TI	Triggered Isomer
TIHE	Triggered Isomer Heat Exchanger
TSFC	Thrust Specific Fuel Consumption (1/hr)
USAF	United States Air Force
v	Velocity (ft/s)
W	Weight (lbf)
W_{TO}	Takeoff Weight (lbf)
x	Shield Thickness (in)

ISOMER HEAT EXCHANGER COMBUSTOR REPLACEMENT FOR A SUPERSONIC RAMJET POWERED VEHICLE

1. Introduction

1.1 Motivation

Since 1942, with the creation of the first fission chain reactor, scientists and engineers have contemplated the use of atomic energy to power aerospace vehicles. These proposals originated with the idea of using a fission "pile" to heat a working fluid to high temperatures for use as a rocket propellant. In 1946, the United States Air Force established the Nuclear Energy for Propulsion of Aircraft (NEPA) project. NEPA was abandoned in 1951 in favor of a joint Air Force/Atomic Energy Commission (AEC) Aircraft Nuclear Propulsion (ANP) program to carry out engineering development of aircraft reactors and engine systems. Both these projects were cancelled due to a technical inability to produce a safe, controllable reactor that met the power requirements of flight at a sufficiently low weight, as well as safety concerns for the crew and civilians who might be affected (1:1-6).

Throughout the past 40 years, little research has been conducted specifically related to nuclear powered aircraft engines. Recent developments in the field of controlled or triggered nuclear decay (2, 3) as well as advances in airframe design, jet engine technology, and materials have increased the possibility of running an aircraft solely upon nuclear power. Nuclear power could conceivably provide aircraft with compact heat sources allowing larger thrust levels than conventional chemical combustion systems can provide, as well as practically eliminating endurance limitations based upon fuel requirements (4).

If such an engine could be utilized to provide heat energy to jet engines, it could revolutionize the modern aerospace industry. Aircraft would be able to fly for days at a time with no need for aerial refueling. Drag limitations could be compensated and thrust to weight ratios for these engines could allow for vertical or short runway takeoffs to become commonplace.

Research into this remarkable concept should begin with basic systems and suitable first step applications. Research into replacing a combustion section of a turbojet engine, with a triggered isomer heat exchanger (TIHE) represented a good start to the research process, by showing that the concept was feasible (5). The next step was to investigate the possibility of utilizing a TIHE to increase the endurance of a High Altitude Long Endurance Intelligence, Surveillance, and Reconnaissance aircraft (6).

1.2 Problem Statement

The goal of this study is to determine the feasibility of using a triggered isomer heat exchanger in a ramjet powered vehicle, such as a cruise missile. It was decided to use the Boeing Conventional Air Launched Cruise Missile (CALCM) as a baseline representative of a generic air launched cruise missile. The performance of the TIHE powered CALCM will be compared with the Boeing model. Shield weights will be determined using the point source gamma radiation shielding methods.

1.3 Thesis Overview and General Comments

This work is divided into 5 chapters. Chapter 2 contains information about the history of and results from research conducted on nuclear powered aircraft, as well as background information on the triggered isomer research program. Chapter 3 presents a discussion of the methods and theories used for this research. This chapter will also detail the two programs used to obtain results. Chapter 4 will present the results of the study. Chapter 5 includes the overall conclusions of the study and any recommendations for future study.

Due to the common practice of using English units in engine and aircraft design, the values used in calculations for this feasibility study are displayed primarily as English units. SI units are preferred for most other engineering applications and will be included where necessary.

2. Background/History

The purpose of this chapter is to present the history of the nuclear powered aircraft program, detail the triggered isomer physics program, discuss ramjet engines, and to introduce the Conventional Air Launched Cruise Missile.

2.1 History of Nuclear Powered Aircraft Development

The concept of using nuclear power in aircraft was studied extensively by both the United States and the Soviet Union throughout the mid 20th century. In fact, Enrico Fermi and his associates in the wartime Manhattan District Project proposed and discussed the use of nuclear fission energy for aircraft and rocket propulsion almost from the time of the achievement of the first fission chain reaction (1). Very little research has been done on using a triggered isomer reaction though. Other than Hartsfield's and Hamilton's works (5, 6) there is no published work in the field of TIHE powered aircraft.

2.1.1 USAF Nuclear Propulsion Aircraft Program

In 1946 the interest in atomic aircraft developed into a long-lived project known as NEPA, for Nuclear energy for the Propulsion of Aircraft. The NEPA project, which started in May, was controlled by the USAF and was therefore oriented towards developing both an atomic- powered long-range strategic bomber and high-performance aircraft (7). In 1948, the Atomic Energy Commission created a separate study at the

Massachusetts Institute of Technology to study the feasibility of nuclear propulsion for aircraft. This study resulted in the "Lexington Report" that concluded that nuclear propulsion was indeed feasible and that it could be achieved in 15 years with a price tag of over one billion dollars (4,8).

The findings of the Lexington Group resulted in a combination of the two research efforts into the Aircraft Nuclear Propulsion (ANP) program. The ANP program set forth the ambitious goal of full-scale development of aircraft reactor and engine systems within 3-5 years. Demonstration of nuclear flight was also a priority. Unfortunately, development occurred at a much slower pace than anticipated due to lack of funding.

Several projects were developed within the ANP program including the Project Rover nuclear rocket, the Project Pluto nuclear ramjet, and the Snap nuclear auxiliary power systems programs (4). Progress was achieved within important research projects such as aircraft development, jet engine and reactor design, and radiation shielding.

2.1.2 Nuclear Airframe Development

Part of the ANP program was the X-6 program. Beginning in 1952, the designated goal of the X-6 program was to produce two flying aircraft powered by atomic energy. A B-36 was converted for this purpose. This aircraft was referred to as the Nuclear Test Aircraft (NTA). It was modified to carry a small air cooled reactor in the aft bomb bay and to provide shielding for the crew. The NTA incorporated shielding around the reactor itself and a totally new nose section which housed a twelve ton lead and rubber shielded compartment for the crew. There were also water jackets in the fuselage and behind the crew compartment to absorb radiation. The reactor was made critical in flight on several occasions and the aircraft was used for many radiation and shielding experiments (7).

A separate contract was awarded to Lockheed Aircraft Corporation to investigate the feasibility of a transonic bomber that would fly below 5,000 feet (9). The study conducted by Lockheed pointed out several important concepts that a nuclear powered aircraft designer would have to take into account (10). The first was that since reactors of the time were immense (around 10-20 thousand pounds), it represented a highly concentrated weight in the aircraft. In a conventional aircraft, the weight of the fuel is normally carried in the wings, spreading the total weight of the aircraft throughout the structure. This could not be accomplished with an all-nuclear aircraft, where the reactor and majority of the shield weight would be located near the engines. This would require intensive structural consideration in the design (6).

The second concept was that powerful radiation emanating from the reactor must be attenuated to acceptable levels. This resulted in the concept of divided shielding-the shield is divided into a section around the reactor and a section around the crew compartment. This helped reduce the total weight of the shielding while providing necessary radiation protection for the crew. A disadvantage to this is that it allows for high radiation rates everywhere else in the airframe and into the environment (6).

Since this propulsion system resulted in virtually unlimited flight endurance, several design challenges were introduced. Traditional aircraft performance calculations factor in fuel weight loss over the duration of the mission. A lack of change in weight actually would simplify the calculations; however this must be kept in mind during design work (6). Landing gear would have to be reinforced due to the fact the landing weight would be equal to the takeoff weight.

2.1.3 Reactor and Engine Development

Two separate contracts were issued by the ANP were made for the development of the reactors and jet engines. General Electric Co. was issued a contract to develop a direct-cycle turbojet while Pratt & Whitney Aircraft Division of United Aircraft Corp. was authorized to study an indirect cycle. The indirect cycle involves using a liquid metal heated by the nuclear reactor. The liquid metal would then be used to heat the air flow in a turbojet engine. Gains were made in reactor and heat exchanger designs, but a test reactor was never produced (7). General Electric's direct cycle program was extremely successful. In a direct cycle jet engine, the airflow in the engine is diverted after it leaves the compressor. It then is heated directly by a reactor then ducted back into the turbine section of the engine.

Several experiments were conducted were the direct cycle concept. These were referred to as the Heat Transfer Reactor Experiment (HTRE) series. The series involved three reactors, HTRE-1 through HTRE-3. HTRE-1 became HTRE-2 at the conclusion of its test program. HTRE-1 (and therefore HTRE-2) successfully ran one X-39 (modified J-47) solely under nuclear power. HTRE-3 was the closest to a flight article the program came. It was solid moderated, as opposed to the earlier reactors which were water moderated, and it powered two X-39s at higher power levels. HTRE-3 was limited by the two turbojets, but it could have powered larger jets at even higher power levels. HTRE-1 was principally a proof of concept reactor. (7) These experiments proved that the nuclear powered aircraft is feasible. Although these tests were highly successful, a flight test model was never constructed.

2.1.4 Shield Development

Early research for radiation shielding was conducted both on the HTRE tests, as well as on board the Convair B-36 that was cut from the X-6 program. The aircraft was fitted with a one-megawatt reactor weighing 36,000 lbf (160.1 kN), for shield research. The aircraft completed 47 successful flights during the remainder of the ANP program. Unfortunately, shielding requirements for the envisioned manned bomber were prohibitive (9). These shield weights were immense for the first aircraft pursued. A low, fast, manned bomber was the goal throughout the NEPA/ANP program. Radiation levels would have been huge for such an aircraft due to the immense reactor power needed.

The ANP program was cancelled in 1961 due to the political pressures stemming from the planned nuclear test ban treaty and the technical hurdles detailed above.

2.2 Triggered Isomer Program

While research in the field of radioisotope decay is not new, the ability to trigger a large release of this energy on demand is a recent discovery. The Directed Energy Directorate of the Air Force Research Laboratory has been working in this field for the past several years and a joint Department of Defense and Department of Energy effort has been created to pursue this technology (3).

In 1998, University of Texas researchers led by Dr. Carl Collins were able to trigger significantly increased energy decay in a hafnium isomer sample using a dental X-Ray unit (2). The decay of the hafnium in this case was a cascade of gamma rays and X-rays of varying energy levels. Some of the X-rays in the cascade were similar in power and wavelength to the triggering X-rays from the dental device. If a means of reflecting the X-rays can be incorporated into a reactor, a chain reaction might be possible. This would allow for a near instantaneous decay and the creation of a controllable power source. (3:1)

This compact power source would be capable of providing large amounts of heat. If made part of a high thrust-to-weight ratio heat exchanger propulsion system, aircraft and spacecraft would be able to utilize this new power source (6). Another advantage of this power source is that the only significant radiation product for TI reactions is gamma radiation. While still dangerous, this radiation would require much less shielding.

One of the studies commissioned by the triggered isomer program was a feasibility study of replacing outright a combustion section of an off-the-shelf turbojet engine with a solid-state heat exchanger (5). This study, utilizing current computational fluid dynamics and heat transfer methods, was able to show that a J-57 turbojet engine could provide equal thrust with a combustor or a heat exchanger at sea level static conditions. Several conclusions were made in this study.

The first is that if the heat generation rate could be controlled and that the heat exchanger material itself was made from the isomer, several different configurations could be utilized to be suitable replacements for the combustor.

The second conclusion was that the ability of this heat exchanger to supply sufficient heating to flow increases with higher altitudes. Heating requirements would decrease due to the thermodynamics involved in engine performance at these heights. This results in lower heat generation rates and reduced radiation output, thereby increasing component lifetime.

The final conclusion was that this heating source would greatly increase aircraft performance and "drastically change the operating paradigms for many missions." (6:5-2) Heat exchanger geometry and construction could be optimized for a specific aircraft or platform.

2.3 The Ramjet Engine

The ramjet engine is the simplest of all air-breathing engines. It consists of a diffuser (inlet), a combustion chamber, and an exhaust nozzle (see Figure 1 below). Air enters the diffuser, where it is compressed before it is mixed with the fuel and burned in the combustion chamber. The hot gases are then expelled through the nozzle because of the pressure rise in the diffuser as the incoming air is decelerated from flight speed to a relatively low velocity within the combustion chamber. The disadvantage of the ramjet is that the pressure ratio is strictly limited by flight speed and diffuser performance. This results in the inability of the ramjet to produce to static thrust and therefore cannot accelerate a vehicle from a standing start. (11:155-157) In this study, the Triggered Isomer Heat Exchanger would be placed within the combustion chamber to heat the air exiting the diffuser.

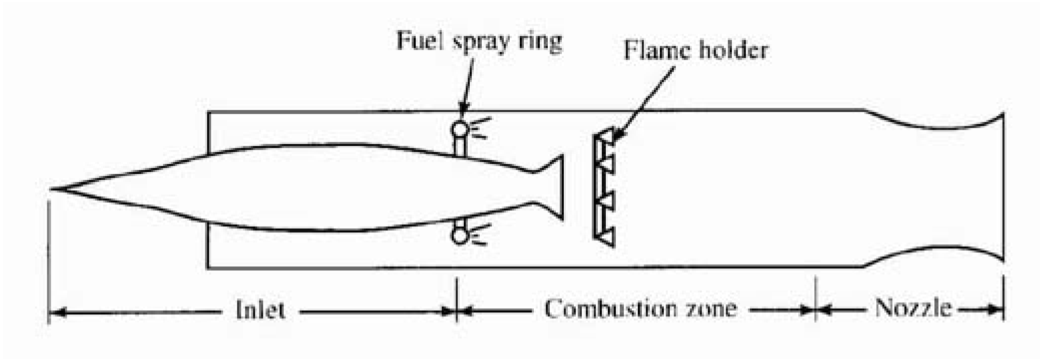


Figure 1. Schematic of a Ramjet Engine (12)

2.4 The AGM 86 Conventional Air Launched Cruise Missile

The AGM-86C CALCM will be the test vehicle used to model a cruise missile in this study. It was originally developed to increase the mission effectiveness of the B-52 Stratofortress bomber and carried nuclear weapons payloads. The Block O CALCM is now equipped with a 2000 lb class high explosive blast/fragmentation warhead. When carried under, or in, aircraft the flying surfaces are folded around the missile body (Figure 2) and are unfolded during launch (Figure 3). It has a 20.73 ft (6.32 m) long by 2.26 ft (.69 m) wide fuselage, an extended wing span of 12 feet (3.65 m) and a launch weight of 3858 lbf (1750 kg). The vehicle is powered by single Williams F-107-WR-101 turbofan engine that provides 600 lbf (2.67 kN) of uninstalled sea level static thrust with an engine inlet of approximately 1.5 ft in diameter (13). The current range of the AGM-86 C is 712.75 nautical miles (1320 km).

Figure 2. AGM-86 Being Loaded Onto a B-52 (13)

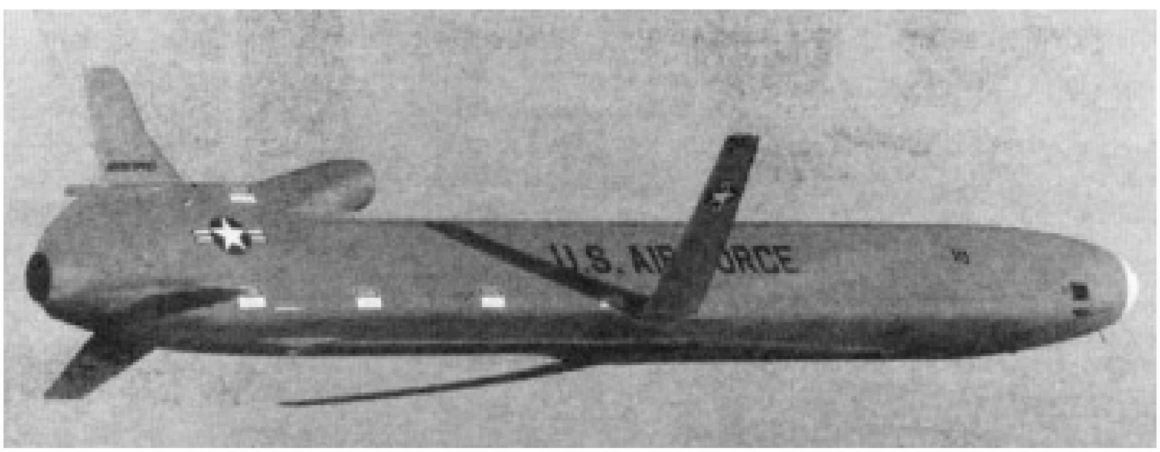


Figure 3. CALCM With Control Surfaces Deployed (13)

3. Methods and Theory

This chapter details the methods use the select mission and engine parameters as well as the shield weight. Included in each section are important theoretical issues that were associated with the engine design process.

3.1 Basic Flight Dynamics

The engine design process begins by considering the forces that act on any aircraft. These forces are: lift (L), drag (D), thrust (T) and weight (W). Lift is the force that overcomes the weight of the aircraft and keeps the plane aloft. Thrust must overcome or be equal to the drag the aircraft encounters while flying to keep it moving forward.

A traditional equation for lift is:

$$L = nW = \frac{1}{2}\rho v^2 C_L S \qquad (1)$$

where

n = load factor
ρ = density
v = velocity
C_L = lift coefficient
S = wing area (ft^2)

All values needed to find lift are known during each flight phase except for C_L. This value is determined using Equation 2.

$$C_L = \frac{nW}{qS} = \frac{n\beta}{\frac{1}{2}\rho v^2}\left(\frac{W_{TO}}{S}\right) \qquad (2)$$

where

W_{TO}=takeoff/launch weight
β= current weight fraction (W/ W_{TO})

The equation for aerodynamic drag is similar to Equation 1:

$$D = \frac{1}{2}\rho v^2 C_D S \qquad (3)$$

where C_D is the drag coefficient. C_D values are normally found using wind tunnel testing, CFD, or flight test data and are based on C_L numbers. An equation for calculating C_D, as a function of C_L, is called the drag polar and represents the sum of induced drag, skin friction, and pressure drag components.

$$C_D = K_1 C_L{}^2 + K_2 C_L + C_{Do} \qquad (4)$$

$$K_1 = K' + K''$$

$$K_2 = -2K'' C_{L\min}$$

$$C_{Do} = C_{D\min} + K'' C_{L\min}{}^2$$

where

K’ = inviscid drag due to lift (induced drag)
K’’ = skin friction and pressure drag
C_{Lmin} = minimum value of C_L
C_{Dmin} = minimum value of C_D

Thrust required can be readily determined now that lift and drag at each point in the mission can be calculated. Two different types of thrust maneuvers exist: one where thrust is equal to drag for steady flight or steady turns and the second where the thrust is either larger or smaller than the drag leading to acceleration or deceleration of the aircraft. The methods used to determine thrust required will be explained in a later section.

3.2 Cruise Missile Drag Profile

Drag polar data can be roughly estimated for aircraft that fit into several categories (fighters, large passenger/cargo planes, small private planes) due to the abundance of information available and the similarities within each type. It was decided to model the drag profile for the hybrid cruise missile after that of a current fighter. The high speeds and mission characteristics that will be associated with the hybrid cruise missile match those of a current jet fighter. The lift-drag polar is then estimated using Figures 4 and 5, Equation 4, and K_2=0.

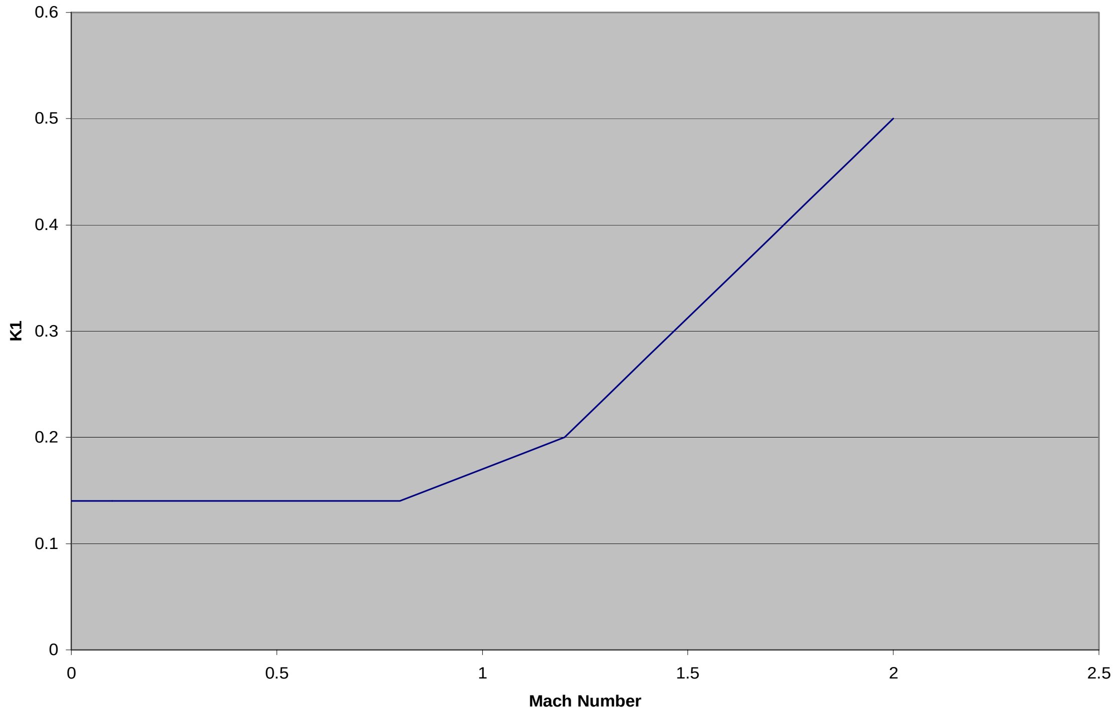


Figure 4. K1 for Fighter Aircraft (14)

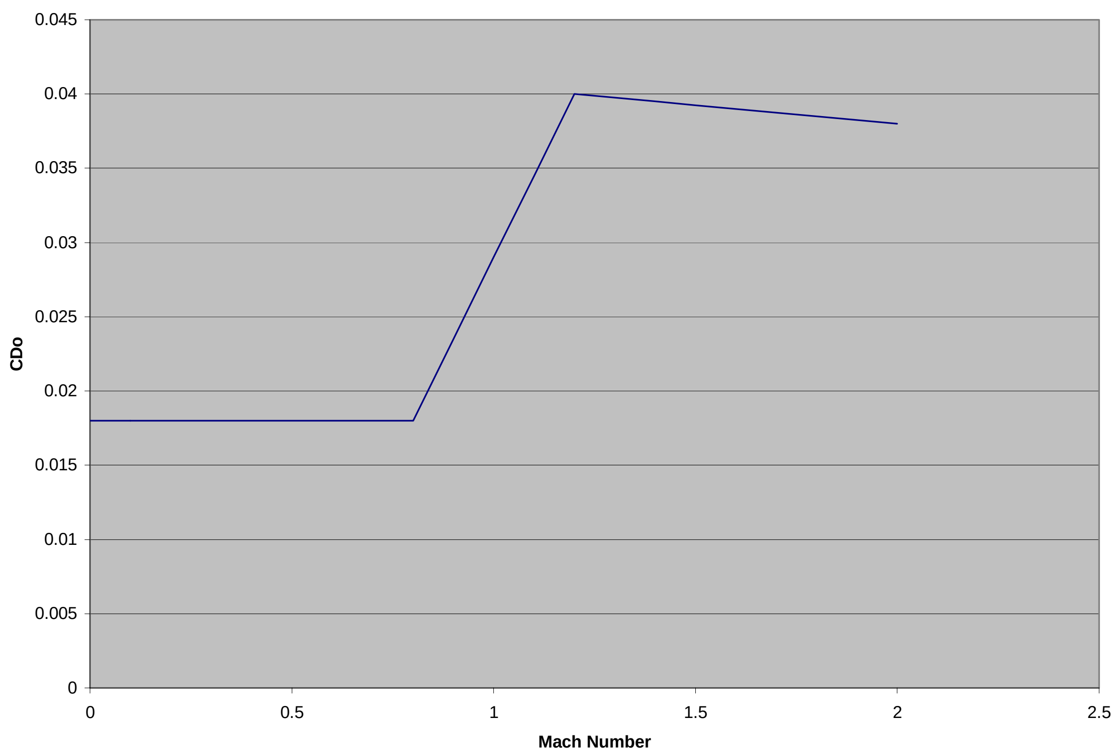


Figure 5. C_{Do} for Fighter Aircraft (14)

3.3 Mission Description

In engine design, three things must be determined prior to starting the design process: aircraft configuration, mission description, and material tolerances. Since the AGM-86 is the vehicle that is being used as a design reference, the aircraft configuration is, for the most part, predetermined. In this section we will examine the reference mission to be used for the TIHE powered air launched cruise missile.

The notional cruise missile mission calls for 4 main legs as seen in Figure 6. The first is the launch from a B-52 Stratofortress at an altitude that is to be determined. A launch speed of Mach 0.8 will be used and is typical for this type of weapon platform (13). This will be followed by a descent to the ideal cruising altitude, also to be determined in this study. The missile will then cruise for a specified distance then descend to the specified target.

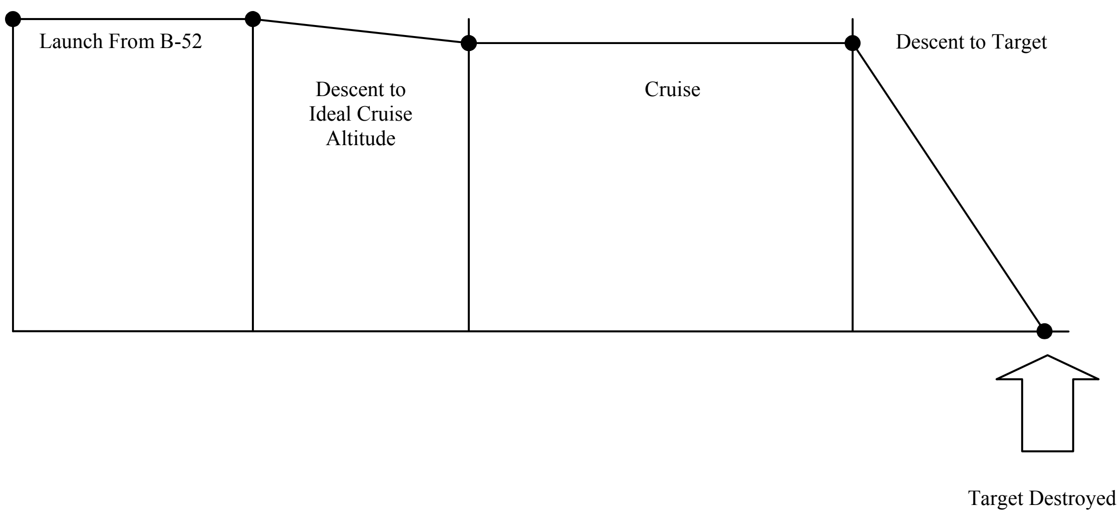
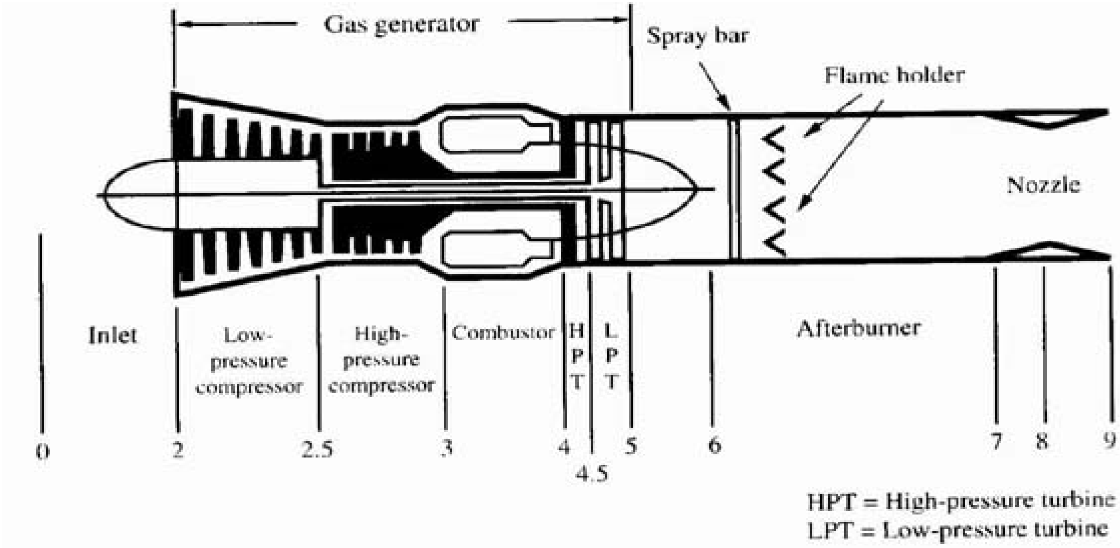


Figure 6. Notional Air Launched Cruise Missile Mission Profile

3.4 AEDsys and ONX

Software that is based upon the engine design process detailed by Mattingly, Heiser, and Pratt's *Aircraft Engine Design* was used in the engine optimization, design, and test processes (14). Aircraft Engine Design System Analysis Software (AEDsys) version 3.1 (15) was used in conjunction with an embedded version of On-Design Analysis of Gas Turbine Engines (ONX) version 5.1 (16). ONX is a design point and parametric cycle analysis program and is used to develop a particular on-design flight. Once proper output is attained, a data file is then saved and inputted into AEDsys. AEDsys is then used to calculate fuel and aircraft performance for a given mission profile from the ONX data file.

Both programs utilize a station numbering system that works for each type of jet engine. Figure 7 displays a diagram and a station numbering system of a dual spool turbojet with afterburner.



Station	Location	Station	Location
0	Freestream	4	Burner Exit
1	Inlet or Diffuser Entry	5	Turbine Exit
2	Low Pressure Compressor Entry	6	Afterburner Entry
3	High Pressure Compressor (HPC) Exit	9	Exhaust Nozzle Exit

Figure 7. Engine Station Numbering (12)

3.4.1 ONX

ONX estimates engine performance using a parametric cycle analysis. The object of the parametric cycle analysis is to obtain estimates of the performance parameters (mainly the specific thrust and thrust specific fuel consumption) in terms of design limitations (such as maximum allowable turbine temperature and attainable component efficiencies), the flight conditions (the ambient pressure, temperature, and Mach number), and design choices. The simplicity of the aerothermodynamic analysis is achieved by treating each stream in the engine as a one dimensional flow of a perfect gas. Non-ideal component behavior is accounted for by including realistic component efficiencies.

(14:95) The process used by ONX is a fairly straightforward series of equations that calculate pressures and temperatures at each station utilizing each inputted efficiency and ratio values. Equations that are not included in this document may be found in both Aircraft Engine Design and Elements of Gas Turbine Propulsion (14, 17).

The primary measure of an engine's overall performance is given by the uninstalled engine thrust (F) and the uninstalled thrust specific fuel consumption. These values can be determined by equations 5 and 6, respectively.

$$F = \frac{1}{g_c}\left(\dot{m}_9 V_9 - \dot{m}_0 V_0\right) + A_0 (P_9 - P_0) \qquad (5)$$

where

F=uninstalled thrust (lbf)
$\dot{m}_o$ = mass flow rate at station (lbm/sec)
V = velocity at station (ft/s)
A = cross sectional area (ft^2)
g_c = Newton's constant (32.174 lbm ft/(lbf sec^2))

$$S = \frac{\dot{m}_f}{F} \qquad (6)$$

where

$\dot{m}_f$ = fuel flow rate to the burner of the engine

Since mass flow rate is not an an input to ONX, the program first tabulates its results as specific values of such as specific thrust (F/$\dot{m}_f$) and uninstalled thrust specific fuel consumption (S). These values, along with the fuel-to-air ratio, ONX input values, component pressure ratios, and component temperature ratios are saved to the output reference file. The user can then enter in a mass flow rate and conduct a design point iteration and final values for uninstalled thrust and specific fuel consumption are calculated.

ONX makes creating various engines a very easy process, once all of the input values have been determined. Each engine type utilizes slight different input parameter. For this study, the challenge is to manipulate ONX, which is meant to be used for conventional turbojet, turbofan, and turboprop engines, to analyze the cycle of a ramjet engine. This was achieved by setting the appropriate efficiencies and ratios to the values shown below in Table 1. The pressure changes normally experienced by the air while in the turbine and compressor were eliminated by using a turbine and compressor efficiency of 1. Variable specific heats were used throughout the ONX design iteration process.

This now leaves only a few important design choices to be studied for the engine optimization process.

Table 1: Engine Design Parameters

Description	Design Value	Description	Design Value
Polytropic Efficiency		*Total Pressure Ratio*	
Compressor	1	Diffuser Max	0.97
Turbine	1	Burner	0.97
		Nozzle	0.98
Component Efficiency		*Miscellaneous*	
Burner	0.98	Fuel (JP-8) Heating Value	18000 BTU/lbm
Mechanical		Bleed Air Flow	0%
Spool	1	Cooling Air Flow #1	0%
Power Takeoff	1	Cooling Air Flow #2	0%
P0/P9	1	Power Takeoff	0

3.4.2 Engine Controls

Several maximum parametric values are important in both the ONX and AEDsys programs. AEDsys will use these inputs as engine controls and restricts the engine from operating above these values. The most important of these controls is the turbine inlet temperature (T_{T4}). It is limited by the thermal strength of the materials used to construct the moving fan blades of the turbine. In the case of the ramjet, this will be the maximum temperature of the air exiting the combustor. Since the ramjet has no moving parts this temperature has the possibility of being much higher. Most modern turbojet/turbofan engines have a maximum T_{T4} of around 3200°R (22). Tests will be run for a maximum T_{T4} of 3200, 3500, and 3800 °R to help optimize the engine. Values for maximum speed for the spools were set at 100% of reference RPMs.

3.4.3 Off-Design Analysis

Within the mission analysis of AEDsys is a sub-algorithm that estimates an engine's performance over a specified operating envelope. This performance analysis differs significantly from the parametric analysis (on-design) calculations. In parametric analysis all of the design choices (including the flight conditions) are free to be selected by the designer, and the engine performance characteristics per unit mass flow are determined for each selected set of choices. The off-design subroutine uses an iterative method using the values found from running ONX for the inputted specific design point engine. It solves a set of independent equations in order to solve for an equal number of dependent variables. Two important concepts are mentioned here to help explain this off-design analysis method (14).

The first is called referencing, in which mass, energy, momentum, and entropy are applied to the one-dimensional steady flow of a perfect gas at either an on-design or off-design steady state operating point. This leads to relations between temperature and pressure ratios that apply to both operating points and can be applied to utilize the reference values from the on-design analysis to calculate the off-design parameters.

The second concept employs a mass flow parameter, where the one dimensional area specific mass flow property is written in terms of total pressure, total temperature or Mach number. The technique is useful in applying the conservation of mass equation and in calculating flow areas.

Another important aspect of the off-design subroutine is that it not only calculates full thrust values at off-design conditions, but it allows throttling the thrust through fuel control to any thrust level within its operational limits. These-off design analysis methods and related equations are thoroughly explained in Mattingly's text (14) and are not included in this document.

3.4.4 Flight Performance Analysis

The most important aspect of the AEDsys software is its ability to calculate required thrust and the corresponding thrust specific fuel consumption for each mission leg by taking into account the off-design performance of a chosen engine, the aerodynamic forces applied to the aircraft, and the changing weight due to the fuel consumption.

Weight change in an aircraft during flight is mainly caused by consumption of the fuel and the release of stores or disposable items. For studying the cruise missile engine with a conventional fuel powered ramjet, this weight change rate is equal to the fuel consumption rate:

$$\frac{dW}{dt} = TSFC \bullet T \qquad (7)$$

where

W = weight (lbf)
TSFC = installed thrust specific fuel consumption (lbm/(lbf sec))
T=installed thrust (lbf)

The T and TSFC values will not be the same as the uninstalled F and S values calculated by ONX and the off-design subroutine. The installed thrust value be smaller and the TSFC larger, due to losses incurred by the inlet and exit nozzle for an installed engine. TSFC can be calculated by incorporating a loss model that can determine the uninstalled thrust needed to attain the required installed thrust. This issue will be detailed later in the document.

Thrust values are necessary for both weight change calculations and the off-design subroutine. The basic forces involved in finding the thrust were detailed in section 3.1. Two other important terms must also be found to calculate the aircraft's flight performance. These are the takeoff thrust loading and the wing loading. The takeoff thrust loading (T_{SL}/W_{TO}) is simply the ratio of takeoff thrust (sea level static) to takeoff weight. Wing loading (W_{TO}/S_{wing}) is the ratio of takeoff weight to wing surface area. For this study, the values were set constant as shown in Table 2 (13). The value used for T_{SL} was set constant at 600 lbf (value given for standard AGM-86C) since a ramjet cannot produce static sea-level thrust. These values were used in various versions of the flight performance equation that AEDsys uses for its calculations.

Table 2: Wing Loading and Thrust Loading Values

T_{SL}	600 lbf
W_{TO}	3858 lbf
S_{wing}	18.73 sq ft
T_{SL}/W_{TO}	0.155
W_{TO}/S_{wing}	205.98 lbf/sq ft

This flight performance equation is constructed by using the principle of conservation of energy around the aircraft. It sets the rate of mechanical energy input equal to the rate of change of potential energy and kinetic energy. The full flight performance equation used in AEDsys (Equation 8) assumes that thrust and aerodynamic drag act in the same direction as the velocity vector.

$$\frac{T-(D+R)}{W}=\frac{d}{dt}\left(h+\frac{V^2}{2g_o}\right) \qquad (8)$$

where:

D+R = total air resistance (lbf)
V = velocity (ft/sec)
h = altitude (ft)
g_o = acceleration of gravity (ft/sec^2)
t = time (sec)

The left hand side of this equation represents the mechanical power delivered into the system. The right hand side is also known as the weight specific excess power or P_S and is the time derivative of the sum of the aircraft's kinetic and potential energies. AEDsys breaks each mission leg into two distinct categories based upon its P_S value. The first case is when there is increasing P_S and is characterized by known values of altitude and velocity changes and usually requires full thrust from the engine for the specified flight conditions. Combining Equations 7 and 8 and performing an integration results in the P_S>0 weight fraction equation:

$$\prod_i = e^{\left(\frac{-TSFC}{V\left(1-\frac{D+R}{T}\right)}\Delta\left(h+\frac{V^2}{2g_o}\right)\right)} \tag{9}$$

where

$\prod_i$ = weight Fraction ($W_{Final}/W_{initial}$)
Δ = change in value form initial point to final point

This value of $\prod_i$ represents the percentage change in weight that occurs during a mission leg. In order to calculate the total aircraft weight after a mission leg, all previous weight fractions, including the current mission leg's value, are multiplied together with the takeoff weight of the aircraft.

The second case is when P_S=0, and represents mission legs where velocity, altitude, and duration or distance traveled are given. These legs usually utilize less than full thrust from the engine and require the throttling method used in the off-design subroutine. In these cases, thrust will equal to the total drag of the aircraft. The following flight performance equation is used by AEDYS for these mission legs:

$$\prod_i = e^{-TSFC\left(\frac{D+R}{W}\right)\Delta t} \qquad (10)$$

The integration involved in finding both Equations 9 and 10 can sometimes lead to erroneous results over an extended mission leg (long range). The most common method of combating this problem is to divide the mission segment into several smaller legs. This results in a better value for fuel consumption than a single averaged equation, but takes less computing time than a full blown numerical integration. This method will be used when comparing TIHE ramjet to a conventional ramjet's range performance.

3.5 TIHE Engine Design Process

Engine design criteria for the TIHE powered ramjet engine are much different than for a conventional powered ramjet engine, since fuel consumption of an isomer would be negligible. This virtually eliminates any range limitations put upon the vehicle.

The disadvantage of using a TIHE powered engine is the radiation emitted from the heat exchanger throughout the flight. This radiation could easily disturb the instruments needed to navigate the cruise missile as well as damage the required launching vehicle and its crew. Since the vehicle is a ramjet powered cruise missile, a high speed flight is ideal. High speeds dictate that the engine must output incredible amounts of power. Since radiation-shielding weights are directly affected by power output from the heat source, the engine that requires the smallest heat power at reasonably high speed would be considered the optimum choice.

The design process for the TIHE powered ramjet engine was a straightforward process. Due to AEDsys' ability to calculate important engine and flight parameters quickly through a mission, it was decided to use this software package for the design progression. A ramjet engine was first designed using the design point and parametric cycle analysis that is offered in ONX using the values in Table 1. A reference altitude of 30000 ft and a Mach number of 1.5 were entered as reference point values. This represents the goal cruising speed and altitude for a ramjet powered cruise missile of this size. With these values and using the inlet diameter of 1.5 ft, a mass flow rate may be calculated (Equation 11).

$$\dot{m}_{air} = \rho V A \qquad (11)$$

where

ρ = Density at Altitude (lbm/ft^3, found using Standard Atmosphere tables)
V = Velocity of freestream air (ft/s)
$A = \pi\left(\frac{d}{2}\right)^2$ (ft^2)

The mass flow rate of air through the inlet of the engine was then inputted into the program and a reference engine was then created. This file was then brought up into the specific switchover mission in AEDsys. The engine control values discussed within section 3.4.2 were entered and a 5% loss model was selected. This loss is incurred by installation of the engine into the aircraft and the drag associated with it The mission analysis routine was then run to calculate engine performance at each leg of the mission.

A means for calculating the heat power required was developed from the outputs provided by AEDsys as follows. Since the conventional fuel provides all of the heat into the conventional system, it is possible to calculate the heat power to run the TIHE system as though it were a conventional engine at the specific flight conditions. AEDsys provides T and TSFC values at each mission leg, multiplying these values together results in a fuel consumption rate.

$$\dot{m}_f = T \bullet TSFC \qquad (12)$$

where $\dot{m}_f$ is the fuel consumption rate (lbm/sec). Fuel consumption rate, along with the burner efficiency value, which represents the ratio of heat energy rise actually supplied to the system to the maximum heat power possible, and the heating value of the fuel, can be used to calculate the heat power required by the TIHE.

$$Q_{burner} = \eta_b h_{pr} \dot{m}_f \qquad (13)$$

where

Q_{burner} = heat power required by TIHE (Btu/sec)
η_b = burner efficiency
h_{pr} = heating value of fuel (BTU/lbm)

Equations 12 and 13, along with the outputs of AEDsys, allow the calculation of required power at all points along the mission profile, thus creating valid and important criteria for the design process. In order to run the TIHE engine in AEDsys, the program had to be manipulated into not burning any fuel while still calculating fuel consumption rates. This was accomplished by setting distance and time lengths for each mission point to zero. This in turn kept the weight of aircraft constant throughout the entire mission.

3.6 Triggered Isomer Decay

The USAF Air Force Office of Scientific Research has been sponsoring an international group of physicists to research an exciting new process for extracting energy from isomers of lutetium (Lu), hafnium (Hf), and tantalum (Ta). These 4 and 5-quasiparticle isomers of Lu, Hf, and Ta are being examined because they are hindered from spontaneous reactive decay due to their specific structural composition. This causes the 2 to 3 MeV excited states of these isomers to have relatively long lifetimes. The process of extracting energy consists of bombarding the isomer with X-rays to excite the material to a higher energy state that would release the nucleus from its structural prohibitions. A rapid decay of the excited isomer could release the total energy of the isomer plus that of the absorbed trigger photon. (2:695; 3:2-13)

The researchers have focused on the 31-year half life, 4-quasiprarticle isomer ^{178}Hf with a 2.446 MeV excitation energy. Using 10 to 90 keV X-ray pulses from a dental quality device, they were able to cause the absorption of X-ray photons on the order of 40 keV of energy to induce the prompt release of the 2.446 MeV stored in the hafnium isomer. This research resulted in a source of power that returned 60 times the energy inputted from the X-ray. In particular, the ^{178}Hf isomer stores approximately 1.3 GJ/g of energy and with this X-ray triggered accelerated decay and volumetric energy release rates of up to 50 GW/m^3 are possible (3). Since this reaction releases only photons, no particles, no change to the material properties of the hafnium is expected.

3.7 Radiation Shielding

Radiation shielding must be present on this aircraft so that the dosage upon the navigational instruments guiding the cruise missile can be kept safe. Since the highest level of radiation will be in the form of 600 keV gamma rays, this leads to a straightforward method to determine shielding weights. The method utilized for this study was developed by Hamilton (6:44-51) for the study of the TIHE powered high altitude long endurance unmanned aerial vehicle. This method was derived from a procedure formed in Turner's textbook *Atoms, Radiation, and Radiation Protection* (18: 452-456).

To begin, we must treat the hafnium reactor as a point source of gamma rays so an unshielded weight can be found. Prior to ONX and AEDsys runs, an initial estimate for the maximum amount of heat power necessary for the TIHE powered ramjet engine of 10 MW was used. Efficiency requirements of the heat exchanger will require that most of the radiation will be contained, leaving approximately 5% of the heat power as escaping radiation. (6:2-13)

Utilizing the method detailed by Hamilton (6:44-51), a shield weight of approximately 1700 lbf was calculated. This estimate seems high for the cruise missile study as it represents around 45% of the initial launch weight. It is believed that this weight may be reduced significantly. Hamilton's method was used to design a shield that would have to protect the sensitive intelligence gathering equipment that is employed by a reconnaissance drone. The navigational instruments that would be necessary for a cruise missile will be less sensitive. The cruise missile will also be a one-time use vehicle so long time material wear due to radiation would not be a salient design factor.

3.8 TIHE Weight and Fuel Calculations

While radiation shielding will be the critical factor in the feasibility of using a TIHE source for the ramjet powered cruise missile, other weights must be taken into account for completeness. The two topics covered in this section will be the weight of isomer fuel that is required for the mission and the actual weight of the heat exchanger itself.

Weight calculations for the isomer fuel were a simple calculation. With values of required power, duration of use, efficiency, and energy stored an equation of fuel mass can be derived:

$$m_{hafnium} = \frac{Q_{required}\Delta t}{e_{stored}\varepsilon_{heat}} \qquad (14)$$

where

$m_{hafnium}$ = mass of hafnium required (kg)
$Q_{required}$ = heat required power (W)
e_{stored} = mass specific energy stored (W/kg)
ε_{heat} = heat conversion efficiency

A sample calculation using the values determined from previous sections, of $Q_{required}$ of 10 MW and e_{stored} of 1.3 GJ/g, a mission time of 5 hours, along with a value of ε_{heat} of 10% estimated from efficiency values of fission reactors tested in the NEPA/ANP programs and Soviet nuclear programs (19), results in a mass of about 1.5 kg. This mass will not form a significant part of the engine design process. If the hafnium fuel is made part of the heat exchanger as proposed by Hartsfield (5), the fuel weight will be absorbed by the heat exchanger weight.

Calculating the actual heat exchanger weight is a complex and time-consuming process. Fluid flow factors such as turbulence, viscosity, and heat transfer must included in any attempt at designing a heat exchange and would require extensive computational models or experimental work. Preliminary research on TIHE was discussed in Chapter 2 and will provide the background for the weight values used in this study. Hartsfield states that for a J-57 turbojet engine, the heat exchanger could add a significant amount of mass of 800-1200 lbf (365-560 kg) (5:4-16). This assumption is based upon a jet engine that provides 8-10,000 lbf. Much less thrust will be necessary for the cruise missile application so the heat exchanger weight will be less. It would be possible to create thrust-to-weight correlations for TIHEs, however with only one study to consult, the decision was made to scale down the values from the previous research (5).

4. Results

4.1 Assumptions

In order to proceed with the task of optimizing the TIHE powered ramjet engine several assumptions were made.

1. A controlled triggered decay of hafnium isomers can be developed to the point that a compact heat source can be produced.

2. Triggered isomer heat exchangers can produce equal heating rates to chemical combustors and be of similar size.

3. Flight of the TI source will be limited to any altitude below 60,000 ft, and to any possible Mach number less than 3.9 due to computational limits.

4. Power output of the TIHE will be limited to 10 MW in order to keep radiation levels low and to reduce the amount of hafnium needed to fuel the vehicle.

5. The increase in weight of the heat exchanger and necessary radiation shielding would be offset by the loss of the fuel and associated system's weight.

4.2 ONX Results

ONX was run at M=1.5, $\dot{m}_{air}$ =77.56 lbm/sec, and an altitude of 30,000 ft as describe in section 3.5. A Mollier Diagram displaying the process the air goes through in an ideal ramjet is displayed in Figure 8. The compression process takes the air from its condition at station A (the freestream) to its stagnation state, B. The combustion process is represented by a constant-pressure heat addition process to the maximum temperature at C. The exit nozzle then expands the combustion products isentropically to the ambient pressure (D).

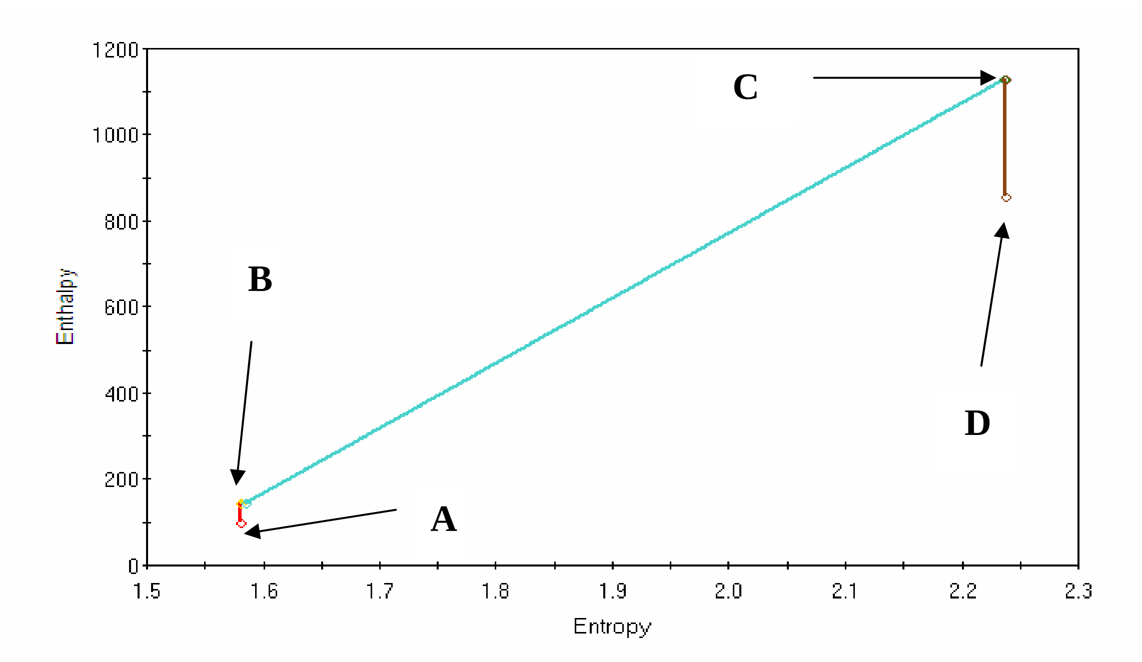


Figure 8. Mollier Diagram of Ramjet Process

4.3 Turbine Exit Temperature

It was stated in section 3.2.4 that tests will be run on the uninstalled engine using three different maximum turbine exit temperatures: 3200, 3500, and 3800 °R. This was accomplished by using the off-design Engine Test program embedded into AEDsys. Tests were run for various Mach number/Altitude combinations and the amount of thrust available from the engine at these off-design points was noted. Table 3 displays an example run at an altitude of 35,000 ft.

Table 3 : Maximum Uninstalled Thrust Available (lbf) at 35,000 ft

Mach Number	3200	3500	3800
0.4	115	125	133
0.5	333	345	355
0.6	526	540	552
0.7	726	744	759
0.8	949	972	990
0.9	1204	1233	1256
1	1544	1573	1570
1.1	1888	1956	1943
1.2	2282	2365	2442
1.3	2742	2845	2940
1.4	3277	3404	3520
1.5	3900	4055	4197
1.6	4626	4817	4991
1.7	5471	5704	5917
1.8	6449	6735	6995
1.9	7580	7928	8246
2	8881	9306	9693
2.1	10373	10889	11359
2.2	12075	12700	13271
2.3	14009	14765	15455
2.4	16197	17110	17942
2.5	18661	19759	20760
2.6	21424	22741	23940
2.7	24506	26081	27515
2.8	27928	29806	31517
2.9	31709	33942	35975
3	35865	38512	40922
3.1	40412	43540	46386
3.2	45357	49044	52396
3.3	50708	55038	58975
3.4	56463	61534	66144
3.5	62617	68538	73920
3.6	69154	76048	82313
3.7	76053	84053	91321
3.8	83275	92536	100944

Table 3 shows that the highest amount of maximum uninstalled thrust available would be for a maximum T_{T4} of 3800 °R. This trend is also displayed for all altitudes tested (0-50,000 ft). This was as expected due to the fact that higher combustion temperatures lead to higher peak thrust. This higher temperature limit would allow the ramjet to operate at higher Mach numbers than tested.

4.4 Cruise Missile Mission Testing

The purpose of this section is to detail the mission legs for the TIHE powered cruise missile.

4.4.1 Leg 1 - Launch

Launch of the cruise missile will occur at Mach 0.8 and may occur at altitudes of up to 50,000 ft (the ceiling for a B-52). Using the "mission" flight performance analysis program imbedded into AEDsys a proper launch altitude could be found. With Mach number set constant at 0.8, the cruise missile was tested at all altitudes within the possible range at increments of 2500 feet. Values for thrust and thrust specific fuel consumption were calculated and entered into an offline spreadsheet. The heat required from the TIHE could then be found using Equations 12 and 13. Table 4 displays the calculated values for each specific altitude. At altitudes above 40,000 ft the engine could not produce the required amount of thrust to operate at this Mach level. This is due to the low density of air at these altitudes and the small size of the inlet for this particular engine.

Table 4 displays that the lowest amount of heat power required occurs at an altitude of 25,000 feet. This would be the ideal launch altitude; however it is below the goal cruise altitude of 30,000 feet. Although, the cruise missile would be able to climb to the cruise altitude it would not be as efficient. For this reason, an altitude of 32,500 feet will be used for launch. The heat power required is still below 10 MW and this allows for a descent to cruise.

Table 4 : Launch Altitude Calculations

Mach=0.8	
Altitude (ft)	**Heat Power Required (MW)**
15000	9.658782651
17500	9.205818116
20000	9.19031695
22500	8.623148361
25000	8.514539705
27500	8.535967229
30000	8.74206039
32500	9.159559063
35000	10.11402984
37500	11.41217197
40000	14.46060175
42500	Engine Too Small to Meet Required Thrust
45000	Engine Too Small to Meet Required Thrust
47500	Engine Too Small to Meet Required Thrust
50000	Engine Too Small to Meet Required Thrust

4.4.2 Leg 2 – Descent to Cruise

The next mission leg will be a descent from the launch altitude of 32,500 feet at Mach 0.8 to the cruising altitude of 30,000 feet and a final speed of Mach 1.5. The angle of descent for this leg will be 30 degrees. This leg will take approximately 0.7 nautical miles to complete and will only require 1.8 MW of heat power to complete. Due to the low power required for this stage, radiation effects will be minimal. The acceleration of the missile from Mach 0.8 to Mach 1.5 will be caused by the gravitational forces acting upon it.

4.4.3 Leg 3 - Cruise

Leg 3 will be the cruise to the vicinity of the target. It will occur at an altitude of 30,000 feet at a speed of Mach 1.5. It will require 9.66 MW of heat power from the heat exchanger. The range of the cruise missile is only limited by the amount of hafnium fuel. Using the values found from the example in section 3.8, a range of 4400 nautical miles (nm) is possible using only the 1.5 kg of hafnium fuel. This represents an extraordinary improvement over the range of the original AGM-86C of 713 nm.

4.4.4 Leg 4 – Descent to Target

Once the cruise missile has come within 5 nm of the target it will begin its final descent. It will begin at 30,000 feet at Mach 1.5 and end at 1 foot at a speed of Mach 2. The descent angle will be 45 degrees and 6.55 MW of heat power. This low power setting will keep the radiation levels down. This is especially important as the missile approaches its target at ground level.

4.4.5 Comments on Mission

This mission was based upon an engine designed with a 1.5 ft diameter inlet. The missile's cruise altitude and speed could be higher and faster had the inlet been a larger size. This aircraft could also be able to loiter around the target for some time due to the tremendous amount of stored energy associated with the isomer reactions.

4.5 Conventional Ramjet vs. TIHE Powered Ramjet

The cruise range for a fuel powered ramjet can be easily calculated using the mission software embedded into AEDsys. The empty weight of the standard AGM-86C was estimated to be 3086 lbf utilizing the fighter aircraft empty weight model. This results in a fuel weight of 775 lbf and a range of around 400 nm compared with 4400 nm for the TIHE powered ramjet.

5. Conclusions and Recommendations

This chapter summarizes the research conducted and contains conclusions based upon the current research and recommendations for continued research.

5.1 Summary of Research

This study consisted of the selection of a air launched cruise missile as a vehicle that could benefit from the advantages of using an Triggered Isomer Heat Exchanger as a replacement for the combustor in a ramjet engine. The AGM-86, the most commonly used Air Launched Cruise Missile in US inventories, was selected as the baseline cruise missile for this study. The use of engine design (ONX) and mission analysis (AEDsys) software allowed for the concept to be analyzed. Additionally, radiation shield weight was calculated based on the TIHE maximum power.

5.2 Conclusions

The following conclusions are shown here based on the work done during this study.

1. Replacing the combustor of a conventional ramjet with a TIHE would be an excellent way to improve the range of an air launched cruise missile. The launch aircraft would be able to stay further from harm's way while still having the same amount of firepower delivered to the target.

2. Shielding requirements, that hindered the fission powered aircraft program, are significantly reduced due to the lack of neutron and radioactive product release in a triggered isomer reaction. This reduction in radiation results in tremendous drops in shield weight. Also, the use of the TI source in a one time use cruise missile reduces the shield weight as well.

5.3 Recommendations

The results from this research lead to several interesting directions for future study.

1. Utilizing the heating requirements found in this study, the design of the actual heat exchanger to be used in the ramjet engine could be accomplished using methods from Hartsfield's research. An examination of both a direct and in-direct cycle could be examined as was done during the NEPA/ANP programs.

2. A more intensive optimization of the cruise missile could be done. Design variables such as wing area, inlet area, and drag could be iterated upon to determine an optimal design for the TIHE ramjet.

3. While this study focused on an air launched cruise missile, other vehicles such as rockets powered by isomer reactions could be studied as well. Spacecraft powered by TI reactions could lead to an exciting new era of spacelift and interplanetary propulsion.

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14 ABSTRACT

This study investigated the possibility of utilizing a Triggered Isomer Heat Exchanger (TIHE) within a ramjet engine to power a supersonic Conventional Air Launched Cruise Missile (CALCM). A computational ramjet engine was created using engine analysis software. This model was then run through a simulated cruise missile mission using AEDsys, a mission analysis and engine performance software suite. The replacement of the combustor with a TIHE led to an impressive range increase for the supersonic cruise missile.

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