ANALYSIS OF ADVANCED TECHNOLOGY IMPACT ON HSCT ENGINE CYCLE PERFORMANCE

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Abstract

The objective of this paper is to describe and apply methods that could assist the propulsion system designer in the evaluation and selection of propulsion technologies. The focus here is on the aerothermo-dynamic aspects of the problem, particularly estimation of engine internal losses. This is accomplished by leveraging developments in second law analysis methods that are able to quantify the theoretical work potential as well as the loss in work potential. Two basic methods, exergy and "gas horsepower," are described and the suitability of each for propulsion systems analysis is discussed. These are used to develop a simple approach to engine internal loss estimation, and are then demonstrated on several basic technology scenarios for a High Speed Civil Transport Propulsion System. The various sources of loss for each concept are examined in detail, and the results for the two methods are compared.

Nomenclature

Note: lower case letters for extrinsic state variables denote mass-specific quantities
P = Pressure (psia)
T = Temperature (R)
H = Enthalpy (BTU)
ε = Exergy (BTU)
S = Entropy (BTU/R)
GHP = Gas Horsepower (or Work Potential, HP)
$\dot{m}$ = Mass Flow Rate (lbm/s)
c_p = Constant Pressure Specific Heat (BTU/lbm-R)
γ = Ratio of Specific Heats
W = Work Input/Output
<u>Engine Station Designations</u>
Amb = Ambient Conditions
2 = Engine Front Face
25 = Core Stream Fan Discharge (Aft of Midframe)
3 = Compressor Discharge
4 = HPT Nozzle Inlet
42 = LPT Nozzle Inlet
56 = Core Stream at Mixing Plane
6 = Mixer Discharge Plane
7 = Nozzle Inlet Plane
9 = Nozzle Exit Plane
13 = Fan Discharge (Bypass Stream)
14 = Bypass Duct Discharge
16 = Bypass Stream at Mixing Plane
18 = Liner and Nozzle Cooling Flow Circuit

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Introduction

Technology has always been a major driver in shaping the gas turbine engine ever since its invention by Sir Frank Whittle in 1930. In fact, the design challenges presented by the development of the first Whittle engines were particularly daunting, and stretched the limits of what was then current technology. For instance, one of the most critical technological challenges was achieving the required combustor volumetric heat output, yielding combustion intensities that were an *order of magnitude* greater than had previously been achieved[1].

Additionally, the Whittle engine presented substantial technical challenges in turbomachinery design[2]. For instance, the compressor pressure ratio of the centrifugal impeller developed for the Whittle engines was far greater than had ever been achieved previously (and Whittle achieved this without sacrificing compressor efficiency). The turbine also presented challenges, particularly with respect to mechanical design, and it was only after lessons learned from several turbine failures that it matured to the point of being a flight-worthy design. In fact, throughout the early years of engine development, the only guide as to how to appropriately implement the new technologies was Whittle's technical ability, his faith in his invention, and his dogged determination to see his idea become a reality.

Although it is doubtful that Whittle ever consciously developed a formalized risk management plan in the form used today, he did manage development risk by deliberately picking a design that was as simple as possible (it had only one primary moving part). He selected a centrifugal compressor instead of an axial because there was already a great deal of development work on the centrifugal compressor in relation to engine supercharger design. He also pioneered the use of the reverse-flow combustor because it allowed the use of a short, stiff shaft to connect the impeller and turbine, thus reducing the risk of encountering rotor-dynamic problems during development. In short, it was Whittle's impeccable engineering intuition that guided him to select a blend of old and new technologies that placed the development emphasis in the right areas and gave a good balance between risk and performance[§].

In the more than six decades of progress that has since ensued, the fundamental nature of gas turbine design has changed little. Current technology limits are

[§]For more about the fascinating life and times of Sir Frank Whittle, the interested reader is referred to the biography of reference 2.

still primary drivers on engine configuration and architecture, and selection of the right technologies is still a critical ingredient required for success in the business. However, as the field has matured, the designs have increased in their complexity manifold. As a result, it now requires the skills of many technical experts working in concert to develop basic technologies and implement them in a sensible way such that the performance or cost benefits outweigh the risks incurred during the development program.

Furthermore, the cost of developing a new engine has risen dramatically. The original Whittle prototypes required on the order of 100,000 pounds sterling to develop, which one could imagine would now amount to several million dollars. The cost of developing a new centerline engine today is on the order of *one billion* dollars. As a result, cost is now considered to be a major driver on propulsion system design, even more so than performance. From the point of view of an engine manufacturer, the stakes are very high, and the issue of technology selection is critical.

In the present context, "cost" refers to all phases of the engine life cycle including research and development, manufacturing, acquisition, and operations & support costs. Any decision regarding the infusion of new technologies into an engine must necessarily be based on the cost impact of things such as maintainability, fuel burn, acquisition cost, etc. The result is an extremely complex analytical problem spanning engine aero-thermo and mechanical design, vehicle performance, engine manufacturer business practices, and airline business practices[3]. If one adds the additional dimension of risk, the problem becomes still more complex. The scope of this problem is such that an entire book could be written on the topic, far exceeding the bounds of a publishable paper length. Furthermore, it is a problem that the industry is struggling to overcome today.

The topic of this paper will be on one portion of the larger technology selection problem, this being the aero-thermo aspects of technology impact on engine design. In particular, the focus is on leveraging developments in second law analysis methods towards enabling the selection of technology options. Second law approaches are used here as a means of directly calculating losses in flow work potential and showing where opportunities for thermodynamic improvement lie. While it is acknowledged that any technology decision cannot be based on aero-thermo performance considerations alone, any aero-thermo performance insights gained through the use of methods described herein can only serve to engender a more complete understanding of the cost implications due to technology decisions.

This paper will first categorize basic technologies according to their impact on engine performance, and will also present a brief discussion on fundamental concepts of the second law methods used herein. Next, an approach applying these methods to the engine technology problem is described. Finally, these ideas are applied to the analysis of a High Speed Civil Transport (HSCT) propulsion system to investigate the impact of several basic technology scenarios on the propulsion system aerothermodynamic performance.

Technology in Engine Design

From a purely thermodynamic point of view, the impact of new technologies is manifested in two fundamental ways: a change in the internal losses of one or more components relative to a theoretically ideal process, and a change in the theoretical ideal cycle. Broadly speaking, these correspond to increases in component efficiency and capability, respectively.

The latter is characterized by technology improvements that have always been a cornerstone of the gas turbine propulsion industry, specifically those that allow higher compressor discharge (CDT) and turbine inlet (TIT) temperatures, as these enable better cycle efficiencies and increased core specific power output, respectively. These are technology improvements that allow an increase in machine capability through a change in the theoretical ideal cycle, usually via materials technologies that allow higher temperature operation (such as advanced alloys and cooling technologies).

The former consists of improvements that enable component capability approaching the theoretically ideal flow process. These are typically technologies that either increase the aerodynamic efficiency (such as turbomachinery designed using 3-D aerodynamics codes) or changes in architecture, secondary flows, etc. that reduce internal losses in the machine.

Typical system level figures of merit (FoMs) used in cycle analysis are specific fuel consumption (SFC), and specific thrust, or perhaps cycle efficiency and specific power output. All of these quantities can be accurately estimated using standard analysis methods, usually consisting of cycle, aeromechanical (flowpath), and installation effects analyses. These analysis methods and tools are now well developed and generally quite accurate. However, one thing that cannot be directly provided using standard analysis codes and techniques is the relative magnitudes of the sources of loss within an engine. Clearly, it is desirable to know this because it indicates where the largest losses are occurring and therefore, where technology improvements have the most potential for improving aero-thermo performance.

The reason that cycle analysis codes cannot directly give information as to the relative magnitudes of component loss is that coupling between components makes it difficult to determine the loss due to a particular component. For instance, given an engine cycle with some set of component efficiencies, assume that it is desired to know the loss due to compressor inefficiency. One approach to estimate this loss might be to re-set the compressor efficiency to one and re-balance the cycle, observing the impact on engine performance. However, by changing the compressor efficiency, the operating point of the engine will change also (either shaft power required to drive the compressor will decrease, or compressor speed will increase). The close-coupled nature of the components in an engine makes it difficult or impossible to estimate the loss contribution due to each individual component by simply changing the component performance and re-balancing the cycle. Since the relative magnitudes of losses from various sources are not readily available from first law analyses only, the standard analysis must be augmented with an analysis based on the second law of thermodynamics.

One approach to measure internal losses that has been successfully used in the past is to use the change in entropy across a component as a measure of inefficiency. This approach is based on the second law of thermodynamics and involves summing the entropy changes across a series of components to obtain the total loss. The disadvantage of this approach is that the losses are expressed in terms of entropy. This is not intuitively appealing because entropy is not a quantity that is directly measurable the way that temperature and pressure are, and therefore, it is difficult to relate to. This is where other second law methods such as exergy and gas horsepower become useful by allowing a direct estimate of the loss in each component knowing only the upstream and downstream flow conditions, and the power output of the component. The following section discusses the definitions, relative merits, and potential application of these two concepts to engine design.

Second Law Methods

The term "second law methods" refers to a class of thermodynamic measures of work potential based on the second law of thermodynamics. These methods enable one to directly track departure from a thermodynamically ideal process. The advantage of this approach is the ease with which one can identify sources of loss within the engine and also quantify the relative magnitude of these losses. They have the additional advantage that the loss is expressed in a physically intuitive way, that being the loss in work potential.

The best-known of these second law methods is exergy (or availability) which can be used as a means to determine the magnitude of the individual loss contributions of engine components relative to the thermodynamically ideal process[4,5]. This is the foundation of the exergy concept, which first appeared in the United States largely due to the work of Keenan[6]. Exergy is a thermodynamic state describing the maximum theoretical (Carnot) work that can be obtained from a substance in taking it from a given chemical composition, temperature, and pressure to a state of chemical, thermal, and mechanical equilibrium with the environment. Exergy is defined as:

$$\varepsilon \equiv (H - H_{amb}) - T_{amb}(S - S_{amb}) + (\text{other terms}) \qquad \textbf{(1)}$$

In this case, the "other terms" are used to denote exergy due to chemical potential, radiation, heat transfer, etc. By tracking changes in this quantity, one can determine the loss in work potential due to each flow process. Note that the definition of exergy depends on the ambient environment.

A good way to visualize this property is to plot lines of constant exergy on a Mollier diagram, as shown in Figure 1. The dashed line with slope equal to the ambient temperature is the zero exergy reference line. All points above this line have a positive potential to do work. Also shown are isobaric lines for 1 and 10 atmospheres. Exergy is depicted as the difference between the enthalpy delta-from-ambient and the Carnot losses (the T-S term). Since the exergy concept relates every state to the Carnot reference of work, *change in exergy is a measure of the change in work potential* at every station in an engine.

Although exergy has potential application in propulsion system design, it has not yet found widespread acceptance or application in the engine business beyond the publication of a few papers discussing the potential applications. The reason for this appears to be that there is no perceived need to apply it for cycle analysis, because all information that is needed for engine design can be obtained through classic cycle analysis. However, exergy methods have found application in the field of cryogenics[7] (primarily because of the inherently low cycle efficiencies of cryogenic equipment), and metallurgical processing[8] (because of the high energy requirements). Several of the more recent textbooks on the subject are those by Ahern[9], Li[10], Moran[11], and Szargut[8]. Additionally, significant work has been published in academia towards application of second law concepts to hypersonic propulsion, most recently by Riggins[12,13,14] et al, and Murthy & Czsyz[15].

One of the most comprehensive papers published to date on applications of second law concepts to propulsion is by Clarke and Horlock[16], in which they: 1) developed expressions for availability useful for propulsion systems, 2) illustrated exergy calculations for combustion of fuels, 3) developed general expressions for steady flow of air,

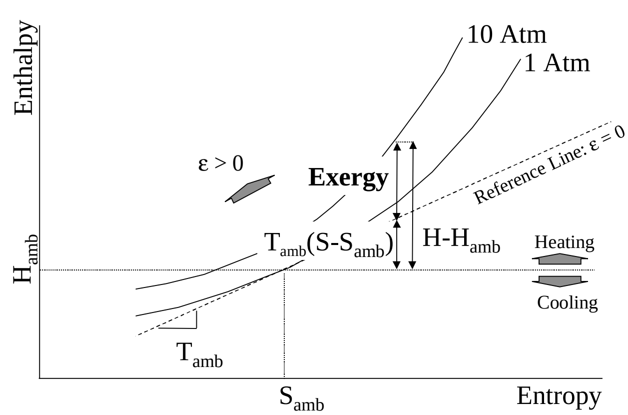

Figure 1: Constant Exergy Lines on a Mollier Diagram

and 4) applied the method to a simple turbojet example. More recently, Brilliant[17] published a paper in which he applied exergy to the analysis of several turbojet and turbofan configurations at cruise flight conditions to show the relative magnitudes of the losses within the engine configurations studied.

Although exergy gives the theoretical maximum work that can be obtained from a substance in a given state, it is sometimes inconvenient for direct application to propulsive cycles. This is due to the fact that exergy bookkeeps heat rejected in the exhaust as being a loss in work potential. To understand this, recall that it was earlier mentioned that exergy is a measure of work potential relative to the *Carnot* ideal. However, the gas turbine engine operates on the *Brayton* cycle, and therefore, even if all component efficiencies are perfect and there are no pressure losses, a gas turbine engine will show losses due to non-equilibrium combustion & heat transfer, as well as exhaust heat rejection. As a result, the Brayton cycle has an exhaust heat exergy loss, even though exhaust exergy is unusable within the confines of the Brayton cycle.

This concept is illustrated in Figure 2, which gives a schematic representation of the Brayton (dotted) and Carnot (solid) cycles superimposed on a Mollier diagram, where both cycles are operating between the same high and low temperature reservoirs. The shaded areas represent the energy available to do work in the Carnot cycle but unavailable in the Brayton cycle. Since the exhaust from the gas turbine engine emerges at an elevated temperature with respect to the environment, it has a positive potential to do work. This appears as an exergy loss in the exhaust stream, one whose magnitude can appear to overshadow losses from other sources (such as component losses and pressure drops).

This difficulty can be circumvented by using alternative second law formulations such as the "gas horsepower" figure of merit (FoM) mentioned previously, which is commonly used to measure the power output of gas generators. The idea behind this concept is that the work potential of a high enthalpy flow can be measured by simply subjecting the gas stream to an imaginary isentropic expansion from the given pressure to ambient pressure. The work produced by this expansion is the maximum work that can be obtained from this flow in the specified environment without the use of some means of heat exchange. Thus, just as exergy is the thermodynamic loss FoM relative to the Carnot ideal, the gas horsepower can be thought of as the loss FoM relative to the Brayton ideal.

The difference between this FoM and exergy for propulsion applications is that gas horsepower does not count exhaust heat as being available to do work. Instead, the work potential is dependent only on the gas conditions and the ambient pressure (ambient temperature does not enter into the picture, as it does for exergy). Only losses due to component inefficiencies and pressure losses will appear in the gas horsepower calculations.

The differences between the two FoMs are illustrated in Figure 3. For open cycles without heat exchangers (such as Brayton), the process start and end points are fixed at atmospheric pressure. The lack of a heat exchange mechanism means the exhaust gas can only be brought to mechanical (pressure) equilibrium, not thermal equilibrium. As a result, the cycle incurs "Carnot losses" and "thermal unavailability" due to inability to recover latent exergy in exhaust stream. The difference between the gas horsepower and the exergy is labeled "thermal exergy" in the figure. Clearly, the difference between the two increases as the entropy increases above ambient entropy. The difference between these two FoMs is discussed in further detail in appendix A.

In general, gas horsepower is a function of the temperature, pressure, mass flow, and composition (fuel to air ratio) of the flow stream. The most general expression for gas horsepower of a fluid stream at a given temperature, T, Pressure, P, and ambient pressure is given by:

$$GHP = \left[\sum_i m_i h_i\right]_{T,P} - \left[\sum_i m_i h_i\right]_{\substack{ambient\\pressure}} \quad \textbf{(2)}$$

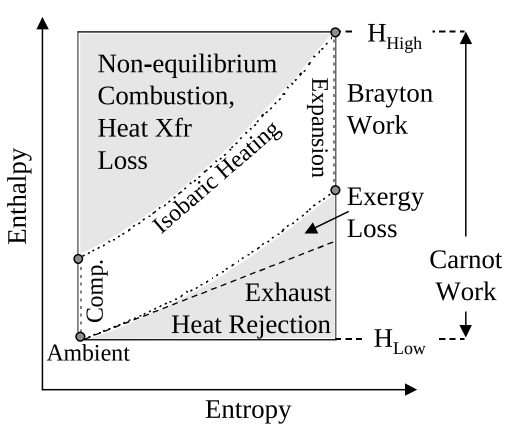

Figure 2: Comparison of Carnot and Brayton Cycles

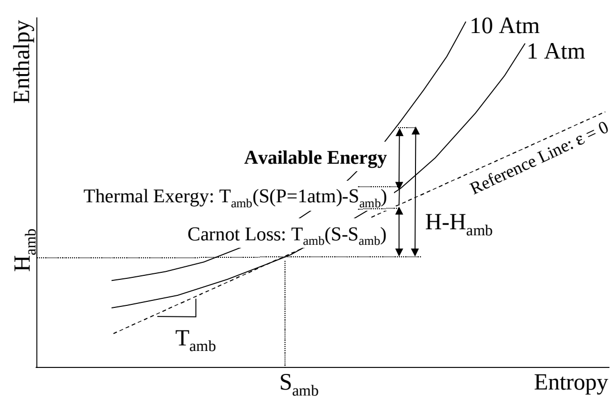

Figure 3: Differences in Work Availability for Exergy and Gas Horsepower Concepts

If the mixture can be assumed to have a constant composition, the above expression can be reduced to:

$$GHP = \dot{m}\left([h]_{T,P} - [h]_{\substack{ambient \\ pressure}} \right) \quad \textbf{(3)}$$

Finally, if the gas is calorically perfect, then the gas horsepower can be expressed as[18]:

$$\text{GHP} = f(T,P) \equiv c_p T \left(1 - {P \over P_{amb}}^{1-\gamma \over \gamma} \right) \quad \textbf{(4)}$$

If the gas can be assumed to be ideal, then the temperature after expansion is easily calculated using the equation of state. The approach given in equations 1 and 3 are the primary tools with which the impact of technology concepts will be evaluated in this paper¶.

Application to Engine Design

The application of either the exergy or the gas horsepower concepts to evaluate engine internal losses is conceptually a very simple process. As shown in Figure 4, one can think of each component in the engine as being a "black box" into which flow enters at state 1 and exits at state 2, and onto which some work is done while some work potential is lost. Thus, this approach requires only knowledge of the state of the gas entering and exiting from each component. The exact nature of the flow process inside the component is immaterial.

Typically, one can assume that the temperature, pressure, and composition at every state inside the engine are known from cycle analysis. Thus, the calculation of exergy and gas horsepower is merely a matter of applying the appropriate equation (eqs 1-4). The difference between the input and output values of work potential is equal to the loss plus the work input (by the first law of thermodynamics). This is illustrated in Figure 4, where in general, one can write:

$$\varepsilon_{in} - \varepsilon_{out} = \dot{W}_{out} + \text{Exergy Loss} \quad \textbf{(5)}$$

or, in the case of gas horsepower:

$$GHP_{in} - GHP_{out} = \dot{W}_{out} + \text{GHP Loss} \quad \textbf{(6)}$$

Note that in general, the exergy loss of a process will not be equal to the gas horsepower loss. Using the equations from the previous section, it is possible to directly calculate the loss in each component *without* the coupling influence between components obscuring the answer.

Although the basic method is very simple, there are a few finer points that deserve mention. First, the treatment of customer bleed and horsepower extraction for accessories drive can be treated either as losses and charged against cycle efficiency or they can be bookkept as a useful output and credited towards cycle work output. The determination of which approach to use is a matter of one's point of view and intentions. Second, one must be careful to use a consistent reference frame when calculating thrust power output. This point is discussed in further detail in Appendix B.

Third, note that it is advantageous in some situations to treat the power output as being the gross thrust times velocity, and then subtract individual installation and ram drags to arrive at net power output. The advantage is that the various components of inlet and nozzle drag can be accounted for individually, and the overall loss attributable to each installation effect can be examined separately rather than being lumped into a total installation drag. This allows a direct comparison of individual installation losses to other loss sources such as internal pressure drops, component efficiencies, etc.

Next, it is important to use the instantaneous ambient conditions that prevail around the vehicle at each instant in its mission trajectory as a reference point (dead state) in the work potential calculations. For instance, for a vehicle cruising at 50,000 ft altitude, the correct reference pressure is clearly not 14.7 psia, but is rather 1.68 psia of the local atmosphere at 50,000 ft. The same applies to the choice of temperature reference, though at high Mach number flight conditions this may require careful consideration since it may not be possible for any wetted surface of the vehicle to experience freestream static temperatures, due to viscous heating of the surface.

Finally, it is worth stressing one last point regarding second law methods, that being the fact that all cycle losses appear as a decrease in work potential. This is very useful if one desires to make an "apples to apples" comparison of all loss mechanisms in an engine, and is one of the primary strengths of lost work as a figure of merit. It enables pressure drops, machine efficiencies, combustion losses, viscous drags, etc. to be directly compared using an intuitively appealing FoM. The next section illustrates these points for an HSCT example and demonstrates how gas horsepower and exergy can be used for examining and weighing the relative merits of technology alternatives.

Case Study: High Speed Civil Transport

The case study used in this paper is the HSCT propulsion system. The HSCT was selected for

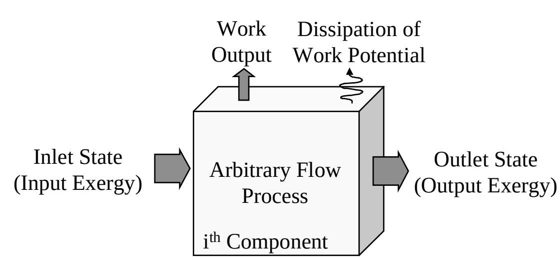

Figure 4: Conceptual Model for Second Law Engine Component Analysis

¶Note that exergy and gas horsepower are not the only second law concepts with potential application to gas turbine engines. Specifically, the concepts of stream thrust analysis[19], thrust work potential[14], and the lost thrust method[14] have the potential to be useful metrics of propulsion system performance.

application of second law methods because: 1) the HSCT propulsion system is highly complex with many disparate sources of loss; 2) the HSCT propulsion system has the potential for the development of an improved cycle due to the large losses incurred during supersonic flight; and 3) the HSCT propulsion system is a prime candidate for infusion of new technologies.

The primary technologies of interest in this paper are those that allow increased compressor discharge & turbine inlet temperature capability, and technologies specific to the HSCT environmental requirements. The former is of interest because these are typical technology development goals that will inevitably continue to improve as basic materials technologies are developed and refined. Therefore, it is desirable to know just how much benefit can reasonably be expected from increased CDT and TIT capability over the next few years. The latter is of interest because the emissions and noise requirements place additional demands on an already difficult design so it is of interest to have a clear picture as to the impact of proposed technologies on overall cycle performance and internal loss.

The Need for HSCT Propulsive Technologies

It was mentioned earlier that the HSCT propulsion system is a good candidate for new technologies. There are several reasons for this, one of which is the environmental requirements placed on the aircraft. First, the HSCT will fly at an altitude where its emissions are thought to have the potential to adversely impact atmospheric composition, particularly ozone. Therefore, the propulsion system must emit far fewer emissions than current engines. Several low emissions technologies have demonstrated the ability to meet these goals and will likely find their way not only onto the HSCT, but also onto high-bypass commercial engines of the future. However, these technologies are not expected to have a dramatic impact on engine aerothermodynamic performance. Instead, the primary impact will likely be on combustion stability (particularly lean blowout), cooling technology (due to the high dome flow rates necessary for lean combustion), and maintainability. Therefore, a detailed analysis of these technologies is foregone in this paper in favor of a more in-depth treatment of other technologies.

A more important environmental requirement from a cycle performance point of view is the need to meet Federal Aviation Administration (FAA) Part 36 Stage 3 noise limits. It is well known that noise is driven primarily by jet velocity, which is in turn a function of specific thrust. Low noise implies low jet velocity and therefore, low specific thrust. This is in direct conflict with the requirements for the supercruise leg of the mission, which demands a high specific thrust engine for efficient supersonic cruise flight. The solution most often proposed is to use some type of mixer-ejector nozzle that acts to entrain low momentum air and mix it with high momentum exhaust during takeoff, thereby lowering the jet velocity. During cruise flight, the ejector doors are closed, and the nozzle acts as a conventional nozzle. Although this configuration may be moderately effective at reducing jet noise, it also causes significant penalties from a cycle point of view.

The nozzle penalties at cruise come from several sources. First, in order to make the nozzle function effectively in the ejector mode, lobed mixers (or “chutes”) must be placed in the primary flow stream to enhance entrainment of outside flow. However, at cruise flight conditions, the ejector doors are closed, and these chutes merely act to cause a pressure drop in the tailpipe, resulting in a loss in thrust and work potential. Second, the nozzle usually considered for use on the HSCT is of the 2-D rectangular type. This geometry acts to enhance noise suppression, but is aerodynamically less efficient than axisymmetric nozzles due to the corner flows which develop in these configurations. As a result, the 2-D ejector nozzle would likely suffer a penalty in cruise thrust coefficient relative to the axisymmetric convergent-divergent nozzle due to this reason (not to mention a significant weight penalty).

The other technologies of interest are those that enable higher cycle temperatures at cruise flight conditions. The need for high compressor discharge temperature (CDT) capability is driven by two main factors: the desire for fuel-efficient cruise flight, and the inherently high temperatures and stringent duty cycle of the HSCT engine. The first point is self-explanatory, as cruise fuel efficiency is always a goal of engine design. The second point is due to the high cruise Mach number of the HSCT. Since the cruise flight condition is supersonic, the HSCT engine will be running near full power for extended periods of time, giving it a very stringent duty cycle. Also, the cruise flight condition will force the engine to operate near the CDT limits of the machine for the majority of the mission.

Finally, the need for high turbine inlet temperatures is driven by the need for minimal engine size, frontal area, and weight. Increased turbine inlet temperatures allow higher core specific power output, thereby allowing the use of a smaller core to drive the same LP system. Alternatively, for the same core size, the fan pressure ratio can be increased to give a smaller engine capable of producing the same thrust, assuming the subsonic cruise fuel penalty and the increased takeoff jet noise can be tolerated.

Baseline Engine

The baseline engine cycle used in this paper is a mixed flow dual-spool turbofan with the cycle parameters given in Table 1. This engine cycle was selected to be representative of the those that have been considered for the HSCT in the past and is not intended to mimic any particular cycle used by either government or industry. The inlet used for this propulsion system is a Mach 2.4 translating centerbody conical inlet, while the baseline nozzle is axisymmetric convergent-divergent similar to

Table 1: Baseline Engine Size and Cycle

Parameter	Value
Fan Pressure Ratio	3.7
Overall Pressure Ratio	19
Throttle Ratio	1.11
Turbine Inlet Temperature	3,400 R
Machine Size (SLS Flow Rate)	650 lbm/s

the type typically used on military engines (non-noise suppressing).

Note that the flow size given in Table 1 is smaller than that typically deemed necessary for a 5,000 nmi range, 300 passenger, Mach 2.4 HSCT. However, all the results presented herein scale directly with flow size (and thrust). Therefore, changes in engine scale will change the absolute magnitudes of the results, but will not change the relative proportions. Thus, the reader can scale these results to any flow size desired by simply multiplying by an engine scale factor.

In this analysis, the sizing condition for dry engine operation is taken to be the top-of-climb flight condition at Mach 2.4 and 50,000 ft, and all engines studied here are sized for the same dry thrust at this condition. Top of climb is used as a sizing condition because the engine corrected thrust is highest here due to the FAA-imposed requirement for a 500 ft/min minimum rate of climb in combination with a nearly full fuel load. Note that the use of 50,000 ft as the critical flight condition is mostly a matter of convenience in this case, as the actual top-of-climb for the HSCT will likely be closer to 55,000 ft. It is a simple matter to obtain engine thrust at any altitude (above 36,089 ft) by simply multiplying the results at 50,000 ft by the atmospheric density ratio between the altitude of interest and that at 50,000 ft.

The results for the gas horsepower analysis of the baseline HSCT engine at the top of climb flight condition are given in Figure 5. These results were generated assuming variable specific heats and variable gas composition in the hot section. All results for cold section gas horsepower calculations are accurate to within 1%, but it should be pointed out that the calculation of gas horsepower in the combustor and turbine flow stations carries a lower accuracy due to the difficulty of accurately accounting for gas composition and internal energy distribution. Therefore, the calculation of gas horsepower loss across the HPT and LPT are a rough estimate only. Finally, the gas inside the engine is assumed to have negligible kinetic energy relative to the vehicle, an assumption that is reasonable for most engine stations.

The schematic diagram of Figure 5 also shows the details of the flow stations and various secondary flow circuits inside the engine flowpath. These include a total of 5 cooling flow circuits for HPT (chargeable and non-chargeable), LPT, turbine rear frame, and liner/nozzle cooling. Also, pressure drops and shaft friction losses are depicted as being analogous to the resistors used in electronics schematics. Finally, power losses are depicted as being analogous to the "ground" symbol used in electronics, and the power loss is noted next to each source of loss.

Several interesting results are apparent from the schematic diagram of Figure 5. First, the inlet incurs a gas horsepower loss of only 1,040 HP, even though the inlet pressure recovery at this flight condition is 0.93. This result is somewhat surprising given the importance of inlet recovery to achieving efficient high Mach flight. A loss of 1,040 HP on a total of 64,857 HP amounts to 1.6%, which is reasonably small given the pressure recovery at this flight condition. The nozzle, on the other hand, has a loss of 3,065 HP on 125,970 HP, or 2.4% of total power available. This is considerably more loss than the inlet, and this effect is largely due to the elevated gas temperature in the nozzle relative to the inlet.

Second, it is clear that the losses in the hot section turbomachinery are greater than their corresponding compression component when compared on a lost work basis. This is in spite of the fact that the turbine adiabatic efficiencies are generally higher than those in the compression system are. Also, note that the high spool has more loss than the low spool. This is due to the fact that the low spool is at a relatively low corrected speed for this flight condition due to CDT limitations, and therefore has reduced shaft work relative to the core spool. Obviously, the reduced corrected speed of the LP spool at cruise causes a significant penalty on cruise

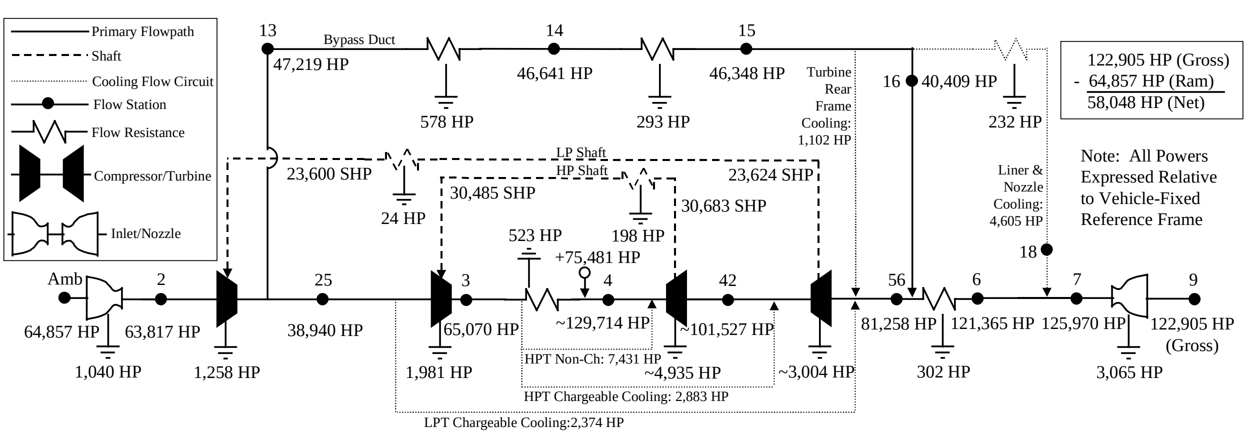

Figure 5: Baseline Engine Performance Schematic (Top of Climb Flight Condition)

thrust lapse, and the baseline engine used here could be considerably improved with some "tweaking" of the cycle, particularly throttle ratio and/or extraction ratio.

Finally, Figure 5 shows that the combustor raises the gas horsepower of the core stream by roughly 75,481 HP. Of this, 58,048 HP appears as net work on the environment. Thus, 76.9% of the work potential added to the system appears as useful work at the nozzle, which is not a bad figure if one is accustomed to the classical cycle efficiencies that are the typically quoted at ~40% for a modern engine.

Technology Concept Examples

An example gas horsepower performance schematic for the improved compressor discharge temperature capability engine scenario is given in Figure 6. This schematic represents an engine with an additional 240 degrees of CDT capability relative to the baseline, with all else remaining the same. The engine is sized to have the same thrust at cruise as the baseline engine, and as a result, the high CDT engine is a much larger machine than the baseline (by a factor of 1.66). This is due to two reasons: first, the higher overall pressure ratio tends to reduce the dry specific thrust of the machine, and second, the cruise corrected speed of the LP spool is decreased from the baseline case. Both of these effects tend to force an increased engine size to compensate for lost flow rate and fan pressure ratio.

The relative magnitudes of the component losses for the advanced CDT engine are roughly the same as for the baseline engine, with the exception of the nozzle. The nozzle loss is reduced due to the lower nozzle pressure ratio available at this design condition (a consequence of low fan corrected speed), and better matching between nozzle area ratio mechanical limitations and nozzle pressure ratio. Also, this engine has a net power output of 63,669 HP on an input of 86,122 HP, or 73.9%. The reduction relative to the baseline engine is due to the reduced core specific power output of this design, and this obviously would not be a desirable way to apply an advanced CDT capability.

A more plausible scenario is given in Figure 7 which shows results for an engine with a moderate (+70 degrees) increase in CDT capability accompanied by an aggressive increase (+450 degrees) in TIT capability on a 0.99 engine scale factor. This engine cycle has a slightly higher gas horsepower input for roughly the same power output, giving a "gas horsepower efficiency" of 76.2%. The gas horsepower losses for this scenario are similar to the baseline case, with the exception of the LP turbine. This difference is partially due to an increase in LP turbine efficiency because the corrected speed of the LP spool is higher at the cruise flight condition for this engine than the baseline. Clearly, this is a more attractive technology scenario than the previous case.

The final case of interest for this paper is the noise-suppressing nozzle scenario where the nozzle geometry required to meet noise constraints imposes a cruise thrust coefficient penalty and an additional tailpipe pressure loss. The tailpipe pressure loss due to the ejector chutes is set at 1% and the nozzle thrust coefficient is penalized by 0.5 point relative to the baseline to reflect what the nozzle losses will likely be. The losses for this scenario are shown in Table 2, which also summarizes the results for all other scenarios examined in this paper.

Note that the losses at all flow stations for the noise-suppressing nozzle scenario are increased marginally due to the increase in engine size required to compensate for the lost thrust at top-of-climb. The sensitivity of engine losses to nozzle performance is not surprising, especially given the importance of the nozzle to the production of supersonic thrust. Note also that the nozzle loss is quite sensitive to changes in thrust coefficient.

Exergy Analysis of Baseline Engine

The last item of interest is an analysis of exergy usage for the baseline engine. The objective of this analysis is to provide a point for comparison against the gas horsepower results. The results given herein account only for exergy losses due to pressure losses, component efficiencies, and mixing. It does not account for exergy losses due to non-equilibrium combustion, incomplete combustion, differences in species partial pressures, etc.

The results of the exergy analysis are given in Table

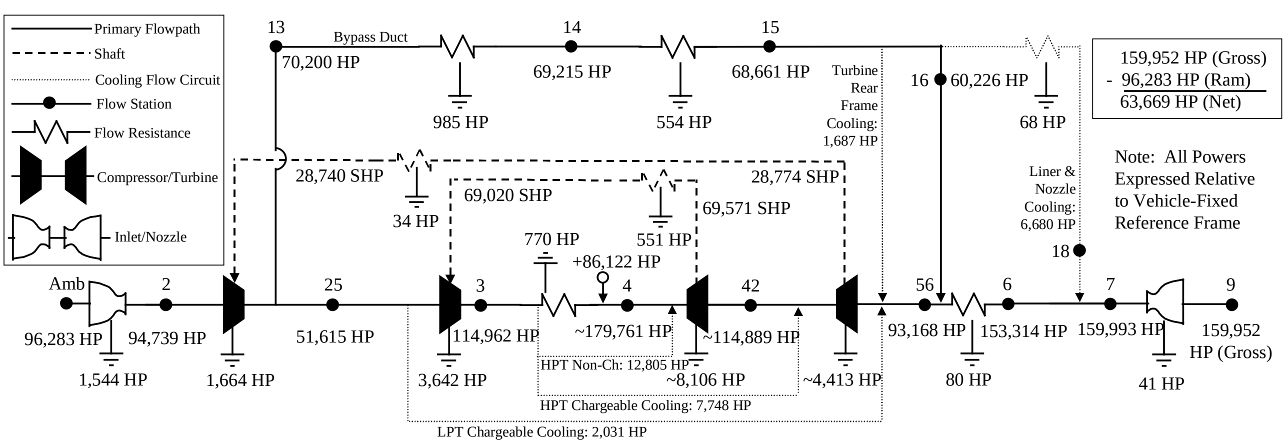

Figure 6: Advanced CDT Engine Performance Schematic

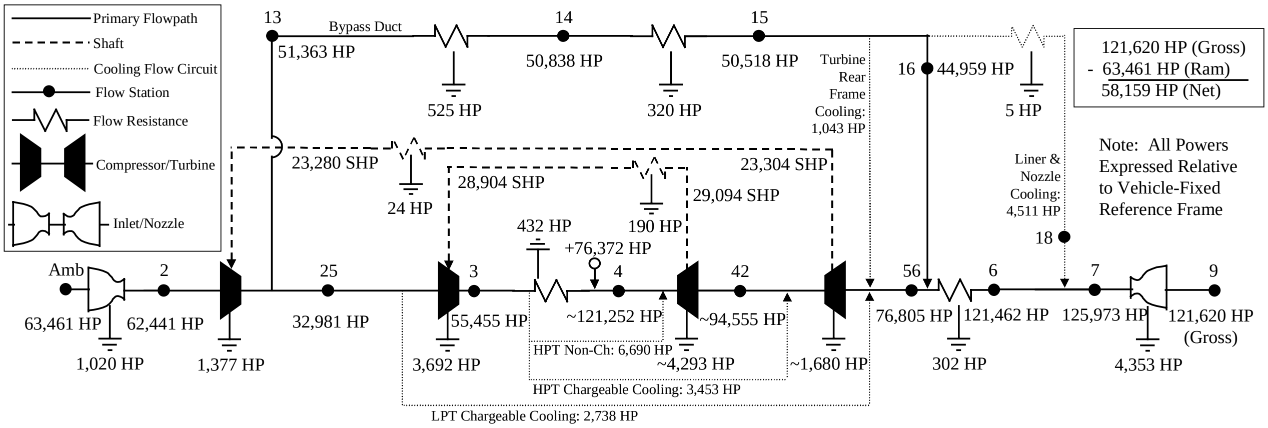

Figure 7: Advanced CDT and TIT Engine Performance Schematic

2 and show that the exergy losses inside the engine are very similar to the losses calculated by the gas horsepower method. This is because the low pressure and temperature at HSCT cruise altitudes is very favorable from an energy recovery point of view. At lower altitudes and higher ambient temperatures, it becomes increasingly difficult to recover latent exhaust heat, and as a result, the exergy method should show much larger losses at these flight conditions.

In general, the exergy loss due to a particular component is less than the gas horsepower loss because the lost work appears as heat in the flow stream, a portion of which can later be used in the nozzle to do useful work. The exergy losses due to mixing have been implicitly lumped into the turbine exergy and tailpipe exergy loss terms in Table 2, and as a result, the apparent exergy loss for these components is larger than the corresponding gas horsepower loss. Furthermore, the exergy loss due to heat in the exhaust stream is lumped with nozzle exergy loss shown in Table 2. Note that from an exergy point of view, the efficiency of the baseline engine is lower that that calculated using a gas horsepower approach. This is due to exergy lost in exhaust waste heat and mixing losses. Finally, note that both the gas horsepower and exergy are additive, as required by equations 5 and 6.

Conclusions

This paper has outlined the fundamental principles and differences between the exergy method of thermal systems analysis and the gas horsepower method often used to measure core engine power output. In this comparison, the exergy method was identified as being a Carnot reference figure of merit, while gas horsepower is a Brayton cycle figure of merit. In effect, the gas horsepower method gives a means to compare losses on a "apples to apples" basis with the Brayton cycle as a reference, while the exergy method gives a means of comparing losses relative to the best theoretical (Carnot) cycle.

The relative merits of the exergy and gas horsepower figures of merit were discussed, and it was pointed out that the gas horsepower FoM treats exhaust heat loss differently than the exergy method. A method for application of these concepts to engine internal loss analysis was presented wherein only the state and

Table 2: Magnitudes of Loss Sources for Various Technology Scenarios (Constant Top-of-Climb Thrust)

Loss Source (All Losses in Horsepower)	Baseline Engine | Loss/GHPin	CDT Tech.	CDT + TIT Tech.	Nozzle Tech.	Base (Exergy)
Inlet Recovery + Internal Drag	1,040 / 1.6%	1,544	1,020	1,048	868
Fan Loss	1,258 / 2.0%	1,664	1,377	1,266	1,214
Fan Mid-frame Pressure Loss	578 / 1.2%	985	525	580	502
Bypass Duct Pressure Loss	293 / 0.6%	554	320	296	294
Compressor Loss	1,981 / 5.4%	3,642	3,692	4,384	1,869
Combustor Pressure Drop	523 / 1.0%	770	432	525	510
HP Turbine Loss	4,935 / 3.6%	8,106	4,293	4,966	5,338
LP Turbine Loss	304 / 2.9%	4,113	1,680	3,026	3,484
Tailpipe Pressure Loss	302 / 0.2%	80	302	304	2,345
Liner/Nozzle Cooling Pressure Loss	232 / 4.8%	68	5	234	177
HP Shaft Mechanical Loss	198 / 0.6%	551	190	200	198
LP Shaft Mechanical Loss	24 / 0.1%	34	24	23	23
Nozzle Aerodynamic Loss	3,065 / 2.4%	41	4353	5,881	14,547
Total Loss	17,433	22,152	18,213	22,733	31,369
Gross Power Output	122,905	159,952	121,620	120,089	122,905
Ram Power Required	64,857	96,283	63,461	65,289	64,857
Net Power Output	58,048	63,669	58,159	54,800	58,048
Net Power/Combustor Power Input	76.9%	73.9%	76.2%	72.1%	64.9%

composition of the gas entering and leaving each component is required to estimate component losses.

The results for the gas horsepower analysis of the HSCT propulsion system show that hot section components are generally responsible for the bulk of the loss in the engine from a gas work potential point of view. The inlet was found to account for a surprisingly small amount of lost work even in spite of the relatively low recovery at this flight condition while the nozzle was found to have high losses and high sensitivity of losses to changes in thrust coefficient. Moreover, the simple technology scenarios investigated for the HSCT propulsion system show only small changes in "horsepower cycle efficiency" due to improvements in fundamental cycle limitations (these being compressor discharge and turbine inlet temperature capability). However, the current analysis did not include the impact of installation effects, and the calculation method used for gas thermodynamic property evaluation of vitiated products of combustion is only an approximate. Finally, it is important to bear in mind that the size impact of various technology scenarios is very significant.

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Appendix A: Comparison of Exergy and Gas Horsepower

As a simple example of the typical procedure used to calculate gas horsepower and exergy, consider a perfect mixer with two incoming air streams and one exit stream as shown in Figure 8. The power available in the inlet and exit streams of this component can be analyzed using either the gas horsepower or exergy approaches. Each will give slightly different results due to the way they treat latent exhaust heat. This section will give examples of the calculation procedures used for each method and explain the differences in the results of the two approaches.

Gas Horsepower Analysis:

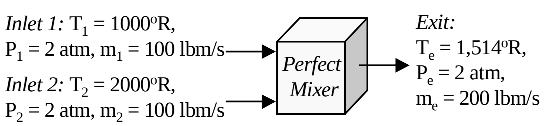
First, consider the flow in stream 1. At 1000^{o}R, the relative pressure of air is 12.339, and the enthalpy is 111.1 BUT/lbm. If this stream were isentropically expanded to 1 atmosphere, the relative pressure would be 6.17, with an exit temperature of 825^{o}R, and exit enthalpy of 67.9 BTU/lbm. The change in enthalpy is equal to the work output, and is 87.4 BTU/lbm. At a flow rate of 100 lbm/s, this translates to 4,310 BTU/s or 6,117 HP.

Likewise for the second stream, the initial temperature of 2000^{o}R gives an enthalpy of 375.1 BTU/lbm, and a relative pressure of 87.25. After imaginary isentropic expansion, the enthalpy is 287.7 at a temperature of 1681^{o}R. The stream gas horsepower is therefore 12,367 HP. The sum of the gas horsepower going into the mixer is 12,367 + 6,117 or 18,484 HP.

For the exit stream, the enthalpy will be the mass average of the enthalpies of the inlet streams, or 243.1 BTU/lbm. This corresponds to a temperature of 1,514^{o}R and a relative pressure of 58.06. After expansion, the temperature is 1,262^{o}R and the enthalpy is 173.3 BTU/lbm. This corresponds to a work output of 65.8 BTU/lbm or 18,619 HP. Thus, the delta between the entrance and exit states is +138 HP or an increase of 0.7%. This delta is insignificant inasmuch as the accuracy bounds of the gas table and the interpolation procedure are not sufficient to discriminate such a small change in work potential. Therefore, for all practical purposes, the gas horsepower of the exit stream is the sum of the gas horsepower of the inlet streams.

Exergy Analysis:

To calculate the exergy of the gas entering and leaving the mixer, one must define not only the ambient pressure, but also the ambient temperature. In this case, the ambient temperature is taken as 520°R. Therefore, the exergy available in stream 1 can be calculated as shown in Figure 8 and is 58.9 BTU/lbm, or 8,339 HP. Likewise for stream 2, the exergy is 32,344 HP. The exergy for the mixed stream is calculated to be 38,431 HP, yielding a loss of 2,316 HP or 5.9% between inlet and exit. This loss is not due to mixer inefficiencies, as it was assumed that the mixer is perfect. Rather, this difference represents the lost work that could have been done by a heat engine operating with the high temperature stream as a heat source and the low temperature stream as a heat sink. This is one of the fundamental results of exergy analysis: mixing of two dissimilar streams always results in a loss of work potential, a result that *is not* reflected in the gas horsepower calculations. Note also the exergy calculations reveal that there is much more work potential available (nearly double) if the latent heat of the exhaust can be harnessed to do work. This should give one an idea of the relative magnitude of the difference between Carnot power output and Brayton power output achievable using these same two streams.



Gas Horsepower Analysis

For Stream 1: T_1 = 1000°R, h_1 = 111.1 BTU/lbm, PR_1 = 12.339 ⋗ $Pr_{p=1atm}$ = 12.339/2 = 6.17 ⋗ $T_{p=1atm}$ = 825°R, $h_{p=1atm}$ = 67.9 BTU/lbm, Δh = 43.2 BTU/lbm ⋗ GHP = 43.2(778.17)/550*100 = **6,117 HP**

For Stream 2: T_2 = 2000°R, h_2 = 375.1 BTU/lbm, PR_2 = 174.5 ⋗ $Pr_{p=1atm}$ = 174.5/2 = 87.25 ⋗ $T_{p=1atm}$ = 1,681°R, $h_{p=1atm}$ = 287.7 BTU/lbm, Δh = 87.4 BTU/lbm ⋗ GHP = 87.4(778.17)/550*100 = **12,367 HP**

For Exit Stream: T_e = 1,514°R, h_e = 243.1 BTU/lbm, PR_e = 58.06 ⋗ $Pr_{p=1atm}$ = 58.06/2 = 29.03 ⋗ $T_{p=1atm}$ = 1,262°R, $h_{p=1atm}$ = 177.3 BTU/lbm, Δh = 65.8 BTU/lbm ⋗ GHP = 65.8(778.17)/550*200 = **18,619 HP**

Change in Gas Horsepower = +138 HP = 0.7% (insignificant)

Exergy Analysis

For Stream 1: $ex_1 = h_1 - h_{amb} - T_{amb}(S^o - S^o_{amb} - R\ln\{P_1/P_{amb}\})$ ⋗
ex_1 = 111.1 -(-5.8) - 520(1.72525-1.5935-0.069ln2) = 58.9 BTU/lbm
Ex1 = 58.9(778.17)/550*100 = **8,339 HP**

For Stream 2: $ex_2 = h_2 - h_{amb} - T_{amb}(S^o - S^o_{amb} - R\ln\{P_2/P_{amb}\})$ ⋗
ex_1 = 375.1 -(-5.8) - 520(1.9342-1.5935-0.069ln2) = 228.6 BTU/lbm
Ex1 = 228.6(778.17)/550*100 = **32,344 HP**

For Exit Stream: $ex_e = h_e - h_{amb} - T_{amb}(S^o - S^o_{amb} - R\ln\{P_e/P_{amb}\})$ ⋗
ex_e = 243.1 -(-5.8) - 520(1.8588-1.5935-0.069ln2) = 135.8 BTU/lbm
Ex1 = 135.8(778.17)/550*200 = **38,431 HP**

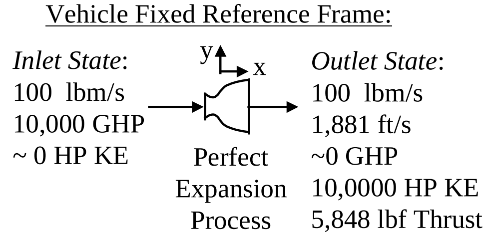
Change in Exergy = -2,316 HP = -5.9%

Figure 8: Comparison of Gas Horsepower and Exergy Results for a Perfect Mixer

Appendix B: Thrust Power Frames of Reference

Thrust power is usually expressed as thrust times flight velocity with respect to a stationary observer standing on the ground. When calculating thrust power in this reference frame, one must add the kinetic energy of the gas upstream of the nozzle relative to the ground to the overall gas power available in order to obtain a consistent measure of gas work potential available at the nozzle entrance. For this reason, the thrust power relative to the earth-fixed reference frame will not, in general, be equal to the gas horsepower available at the nozzle exit in the vehicle-fixed reference frame (except for the static thrust case).

The idea of consistent frames of reference for thrust power calculations is illustrated in Figure 9 for a simple nozzle in two frames of reference. First, in the vehicle-fixed reference frame, a flow of 100 lbm/s with a work potential of 10,000 HP and negligible kinetic energy can be expanded in a perfect nozzle to a velocity of 1,881 ft/s and a kinetic energy of 10,000 HP. In the earth-fixed reference frame, the relative velocity between the earth and the nozzle must be accounted for in the power calculations. Thus, the work potential of the gas entering the nozzle is 10,000 GHP plus 2,826 HP due to the kinetic energy of the propellant gas inside the nozzle reservoir relative to the ground. After expansion, the thrust work is 10,633 HP, and the residual (wasted) kinetic energy of the gas is 2,193 HP.



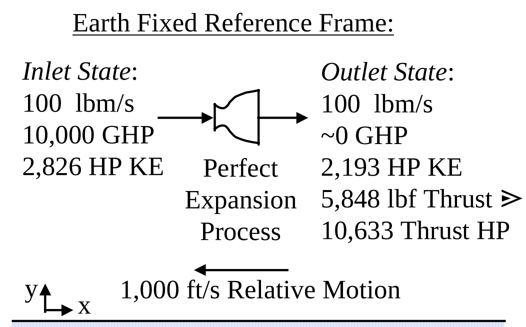


Figure 9: Thrust Power Calculations in Vehicle-Fixed and Earth-Fixed Reference Frames