Effects of Large-Scale Transient Loading and Waste Heat Rejection on a Three Stream Variable Cycle Engine

Michael William Corbett

Wright State University

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EFFECTS OF LARGE-SCALE TRANSIENT LOADING AND WASTE HEAT REJECTION
ON A THREE STREAM VARIABLE CYCLE ENGINE

A thesis submitted in partial fulfillment of the requirements for the degree of Master of Science in Engineering

By

MICHAEL WILLIAM CORBETT

B.S., Wright State University, 2006

2011
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ABSTRACT


The objective of the research presented in this document was to gain an understanding of the feasibility of extracting large amounts of transient shaft power from a variable cycle engine and to reject waste heat into the third stream bypass duct. This first required the development of a transient engine and controller simulation. After performing basic verification of the model, such as ensuring conservation of mass, tests were run for three missions using a low-efficiency periodic load which both required shaft power and created low quality waste heat. The waste heat generated by that load was lifted by a cooling system and rejected into the third stream duct. The impact of the external load and the mission conditions on cooled cooling air behavior and engine performance was assessed. Overall, the engine was capable of performing the missions, providing the required shaft power, and rejecting the waste heat to the third stream.
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## NOMENCLATURE

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<thead>
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<th>Description</th>
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<tr>
<td>$A$</td>
<td>area</td>
</tr>
<tr>
<td>$BPR$</td>
<td>bypass ratio</td>
</tr>
<tr>
<td>Btu</td>
<td>British thermal unit</td>
</tr>
<tr>
<td>$C$</td>
<td>heat capacity rate</td>
</tr>
<tr>
<td>$C_{&lt;&gt;}$</td>
<td>coefficient of...</td>
</tr>
<tr>
<td>CCA</td>
<td>cooled cooling air</td>
</tr>
<tr>
<td>$const$</td>
<td>generic constant value defined for use in a particular equation</td>
</tr>
<tr>
<td>$COP$</td>
<td>coefficient of performance</td>
</tr>
<tr>
<td>$c_p$</td>
<td>specific heat at constant pressure</td>
</tr>
<tr>
<td>$D$</td>
<td>drag</td>
</tr>
<tr>
<td>EU</td>
<td>European Union</td>
</tr>
<tr>
<td>$F_n$</td>
<td>uninstalled net thrust</td>
</tr>
<tr>
<td>$F_{n,inst}$</td>
<td>installed net thrust</td>
</tr>
<tr>
<td>$f(...)$</td>
<td>function of parameters enclosed in parentheses</td>
</tr>
<tr>
<td>$frac_{HX}$</td>
<td>fraction of duct air entering the heat exchanger (the balance enters the air gap)</td>
</tr>
<tr>
<td>ft</td>
<td>feet</td>
</tr>
<tr>
<td>GE</td>
<td>General Electric</td>
</tr>
<tr>
<td>$h$</td>
<td>specific enthalpy</td>
</tr>
<tr>
<td>HP</td>
<td>high-pressure</td>
</tr>
<tr>
<td>HPC</td>
<td>high-pressure compressor</td>
</tr>
<tr>
<td>HPT</td>
<td>high-pressure turbine</td>
</tr>
<tr>
<td>HX</td>
<td>heat exchanger</td>
</tr>
<tr>
<td>$i$</td>
<td>generic index</td>
</tr>
<tr>
<td>$J$</td>
<td>polar moment of inertia</td>
</tr>
<tr>
<td>lbf</td>
<td>pound force</td>
</tr>
<tr>
<td>lbm</td>
<td>pound mass</td>
</tr>
<tr>
<td>LHV</td>
<td>lower heating value of the fuel</td>
</tr>
<tr>
<td>LP</td>
<td>low-pressure</td>
</tr>
<tr>
<td>LPC</td>
<td>low-pressure compressor</td>
</tr>
<tr>
<td>LPT</td>
<td>low-pressure turbine</td>
</tr>
<tr>
<td>$MN$ or $M$</td>
<td>Mach number</td>
</tr>
<tr>
<td>$\dot{m}$</td>
<td>mass flow rate</td>
</tr>
<tr>
<td>$\dot{m}_c$</td>
<td>corrected mass flow rate</td>
</tr>
<tr>
<td>$\dot{m}_f$</td>
<td>fuel mass flow rate</td>
</tr>
<tr>
<td>MOBY</td>
<td>MOdulating BYpass engine</td>
</tr>
<tr>
<td>NASA</td>
<td>National Aeronautics and Space Administration</td>
</tr>
<tr>
<td>NPSS</td>
<td>Numerical Propulsion System Simulation</td>
</tr>
<tr>
<td>$P$</td>
<td>pressure</td>
</tr>
<tr>
<td>$P_s$</td>
<td>static pressure</td>
</tr>
<tr>
<td>$P_t$</td>
<td>total pressure</td>
</tr>
<tr>
<td>PR</td>
<td>pressure ratio</td>
</tr>
<tr>
<td>PID</td>
<td>Proportional-Integral-Derivative (controller)</td>
</tr>
<tr>
<td>POA</td>
<td>Power Optimized Aircraft</td>
</tr>
<tr>
<td>$\dot{Q}$</td>
<td>heat transfer rate</td>
</tr>
<tr>
<td>$q$</td>
<td>dynamic pressure</td>
</tr>
<tr>
<td>R</td>
<td>Rankine</td>
</tr>
<tr>
<td>$R$</td>
<td>gas constant</td>
</tr>
<tr>
<td>$s$</td>
<td>seconds</td>
</tr>
<tr>
<td>SAE</td>
<td>Society of Automotive Engineers</td>
</tr>
<tr>
<td>SLS</td>
<td>sea-level static conditions</td>
</tr>
<tr>
<td>$t$</td>
<td>time</td>
</tr>
<tr>
<td>$T$</td>
<td>temperature</td>
</tr>
<tr>
<td>$T_s$</td>
<td>static temperature</td>
</tr>
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</table>
### Symbol Description

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>$T_t$</td>
<td>total temperature</td>
</tr>
<tr>
<td>$T_q$</td>
<td>torque</td>
</tr>
<tr>
<td>TMS</td>
<td>Thermal Management System</td>
</tr>
<tr>
<td>TSFC</td>
<td>Thrust Specific Fuel Consumption</td>
</tr>
<tr>
<td>$V$</td>
<td>velocity</td>
</tr>
<tr>
<td>$V_e$</td>
<td>nozzle exit velocity</td>
</tr>
<tr>
<td>VABI</td>
<td>Variable Area Bypass Injector</td>
</tr>
<tr>
<td>VAPCOM</td>
<td>Variable Pumping Compressor</td>
</tr>
</tbody>
</table>

### Symbol Description

<table>
<thead>
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<th>Symbol</th>
<th>Description</th>
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<tbody>
<tr>
<td>VCE</td>
<td>variable cycle engine</td>
</tr>
<tr>
<td>$W$</td>
<td>power (work rate)</td>
</tr>
<tr>
<td>$\alpha$</td>
<td>vane angle</td>
</tr>
<tr>
<td>$\gamma$</td>
<td>ratio of specific heats</td>
</tr>
<tr>
<td>$\Delta$</td>
<td>change in...</td>
</tr>
<tr>
<td>$\varepsilon$</td>
<td>effectiveness</td>
</tr>
<tr>
<td>$\pi$</td>
<td>pressure ratio</td>
</tr>
<tr>
<td>$\eta$</td>
<td>efficiency</td>
</tr>
<tr>
<td>$\rho$</td>
<td>density</td>
</tr>
</tbody>
</table>

Additional numeric subscripts correspond to stations numbers given in Table 1 and other subscripts and variations of the above symbols are defined as used.
I. INTRODUCTION

The majority of modern military and commercial fixed-wing aircraft is powered by turbofan engines. Though there are alternatives such as turboshaft, turbojet, ramjet, and more exotic approaches like pulse detonation engines, turbofans generally outshine other options for aircraft traveling at high subsonic and low supersonic speeds. Among turbofans, there is a wide span from low-bypass ratio (BPR) turbofans such as the F100-PW-229 that powers some F-16s (BPR = 0.36) (Pratt & Whitney 2011) to high BPR turbofans such as the GEnx-1B70 which powers some Boeing 787s (BPR = 9.6) (GE Aviation 2009). The BPR determines aspects of the engine such as noise, fuel efficiency, maximum speed, and outer diameter necessary to achieve a desired thrust.

Variable cycle engines (VCE) strive to achieve the most desirable aspects of low and high BPR engines through the use of variable geometry within the engine. The VCE considered in this research makes use of a second bypass stream in addition to the standard core bypass of a conventional turbofan. This outer bypass duct is referred to as the third stream. The addition of the third stream has the benefit of enabling thrust to vary for a fixed inlet mass flow simply by modulating the airflow through different engine flow paths (J. Kurzke 2010).

As aircraft move toward more electric architectures, the amount of shaft power takeoff required from the engine increases. For many commercial aircraft, this tends to be paired with a decrease in bleed air requirements, as more aspects of the cooling system are driven electrically rather than pneumatically (Slingerland and Zandstra 2007). However,
for engines that push peak operating temperatures higher and higher, significant compressor bleed flow is required for cooling within the engine even though other aircraft needs are being met electrically. Before looking at the impact of power extraction (bleed or shaft) on a VCE, it is prudent to survey the underlying concepts individually.

**Overview of Variable Cycle Turbine Engines**

Turbojet engines are known for high specific thrust, which makes them well suited for high speed supersonic flight, short distance takeoff, and highly responsive thrust control (e.g. for intercept or combat maneuvering). These benefits come at the expense of significant noise at takeoff, high exhaust gas temperatures, and poor fuel economy during subsonic cruise (using thrust specific fuel consumption, TSFC, as the metric). Turbfans, specifically high-bypass turbfans, achieve better fuel economy than turbojets at subsonic cruise speeds by accelerating a much larger mass of air to a comparatively lower velocity to produce the same thrust. Turbfans have additional benefits of lower exhaust gas temperatures and reduced noise, but come with the penalty of significantly larger inlet areas (and maximum diameter). The root causes for these apparent tradeoffs are evident from examining the contributing factors in overall engine cycle efficiency.

Overall engine cycle efficiency is often defined as the product of the propulsive efficiency (the ability to produce thrust power from kinetic energy on a rate basis) and the thermal efficiency (the ability to produce kinetic energy on a rate basis and/or mechanical shaft power from thermal energy input on a rate basis) (Mattingly, Heiser and Pratt 2002).

\[
\eta_o = \eta_p \eta_{th} \tag{1}
\]

\[
\eta_p \equiv \frac{\text{Thrust Power}}{\text{Kinetic Energy Generation Rate}} \tag{2}
\]
where $\eta_o$ is the overall cycle efficiency, $\eta_p$ is the propulsive efficiency, $\eta_{th}$ is the thermal efficiency, $F_n$ is the engine net thrust (parallel to the direction of flight), $V_0$ is the aircraft velocity (equivalent to the air velocity coming to the engine from an engine-based coordinate system), $\dot{m}_{010}$ is the mass flow rate of air entering the engine, $V_e$ is the nozzle exhaust flow velocity, $\dot{m}_f$ is the mass flow rate of the fuel being consumed by the engine (primary burner plus afterburner), and $LHV$ is the lower heating value of the fuel.

Note the following simplifying assumptions were used in the above derivation for clarity in explanation: 1) moving from Eq. 3 to Eq. 4 assumes that the additional mass from the fuel is negligible and that the flow exiting a single engine exhaust nozzle is perfectly expanded, thus eliminating additional terms in the uninstalled net thrust; 2) in establishing Eq. 7 from the definition in Eq. 6, simplifying assumptions are made to neglect shaft power extraction terms (i.e. mechanical energy generation rate can be neglected) and that the combustion process can be reasonably represented by a constant pressure process with no external heat or work interactions; 3) the entire derivation assumes that customer bleed is
negligible, so all flow entering the engine is ejected through the single nozzle; and 4) installation effects on net thrust are not considered (Mattingly, Heiser and Pratt 2002).

By examining Eq. 8 in isolation and assuming that the aircraft velocity and fuel lower heating value are known inputs, it would appear that increasing \( \dot{m}_{010} \), decreasing \( \dot{m}_f \), and increasing \( V_e \) are each beneficial to the overall cycle efficiency. However, these terms are not independent of one another. Taking a look at the propulsive and thermal efficiencies separately provides further insight.

From Eq. 5, it is seen that minimizing the nozzle exhaust flow velocity will maximize the propulsive efficiency, i.e.:

$$\lim_{V_e \to V_0} \eta_p = \lim_{V_e \to V_0} \frac{2V_0}{V_e + V_0} = \frac{2V_0}{V_0 + V_0} = 1 \quad \text{Eq. 9}$$

This is the approach taken by a high-bypass turbofan—accelerate a large amount of air to a relatively low nozzle exit velocity. On the other hand, Eq. 7 suggests that minimizing the nozzle exhaust flow velocity actually drives down the thermal efficiency:

$$\lim_{V_e \to V_0} \eta_{th} = \lim_{V_e \to V_0} \frac{\frac{1}{2} \dot{m}_{010} \cdot (V^2_e - V_0^2)}{\dot{m}_f \cdot LHV} = \frac{\frac{1}{2} \dot{m}_{010} \cdot (V^2_e - V_0^2)}{\dot{m}_f \cdot LHV} = 0 \quad \text{Eq. 10}$$

This apparent conflict between thermal efficiency and propulsive efficiency suggests that there is no good solution. There are observations that can still lead to an efficient design. The first is that a turbofan engine can have more than one nozzle (i.e. one nozzle with a high nozzle exhaust flow velocity and one with a low one) and the second is that variable geometry within a variable cycle turbofan can enable it to behave either more like a low-bypass turbofan (high thermal efficiency) or more like a high-bypass turbofan (high propulsive efficiency), depending on the mission segment.
By taking advantage of both of these points, dual nozzles and internal variable geometry, a VCE can achieve the additional benefit of flow holding to reduce spillage drag. Spillage drag occurs when the engine’s operating point requires that it swallow less air than the inlet is capable of capturing. This extra air is spilled around the engine, creating drag. This concept is illustrated in Fig. 1.

![Diagram of engine inlet and spillage drag](image)

Fig. 1 Spillage drag due to airflow mismatch (Simmons 2009)

In the upper view (Fig. 1a), the engine is at its maximum power design point. In this case, the engine ingests all air captured by the inlet, creating very little spillage drag (note that there are, of course, other drag terms which remain). When a conventional turbofan is throttled back to a reduced power setting, the fan speed is normally reduced and therefore the engine requires less air. The air that is decelerated by the inlet but not consumed by the engine must go around the engine, creating spillage drag (Fig. 1b). The magnitude of the spillage drag is governed by the mismatch between the inlet capture airflow and the airflow ingested by the engine, though the worst spillage drag may occur at intermediate
supersonic Mach numbers ($MN$) where the product of the spillage drag coefficient and dynamic pressure is the largest (Simmons 2009).

The VCE enables flow holding to reduce the spillage drag. Flow holding is an approach whereby internal engine geometry is adjusted such that the engine’s total airflow demand remains at the maximum that the inlet can provide despite a need for reduced thrust. To accomplish this, the corrected fan speed is maintained at 100% of the design value and the airflow is distributed between the engine’s various flow paths to achieve the desired thrust. There is, however, a point where flow holding is no longer beneficial. When the turbomachinery variable geometry settings are pushed too far from their ideal positions in order to accomplish the flow modulation, the benefits of reduced spillage drag are outweighed by the reduced efficiencies of compressors and turbines (Simmons 2009).

High speed flight, short takeoff distance, and similar aggressive mission requirements cause an inlet design point with a large capture area. When lengthy portions of the mission dictate a lower engine air flow in a conventional turbofan, the result would be high spillage drag at cruise, resulting in higher overall mission fuel consumption. Since in both military and civil aviation there has historically been a desire to have a single aircraft that can perform multiple missions (or drastically different segments within a single mission), much research into the viability of VCEs has been conducted since the 1950s.

For example, several supersonic transport programs for commercial aircraft have required an engine capable of efficient cruise at both subsonic and supersonic speeds, while meeting noise and emission generation limits (National Research Council Committee on High Speed Research 1997). Similarly, the U.S. Air Force investigated VCEs for military applications. For example, in the 1980s, General Electric (GE) proposed their YF120 VCE for use in the Advanced Tactical Fighter, though it was not selected due to the perceived risk
associated with the variable cycle architecture (Aronstein, Hirschberg and Piccirillo 1998). Many of the technologies proposed for that engine were later proposed for use in the Joint Strike Fighter Alternate Engine Program (GE Aviation 1996).

NASA was heavily involved in supersonic civil transport research and evaluated many candidate architectures including mixed flow turbofans, VCEs, Fan-on-Blade (Flade) engines, and inverting flow valve engines, among others. This research included involvement from major engine and aircraft companies such as GE, Pratt & Whitney, and Boeing (Berton, et al. 2005). Significant research into using VCEs for supersonic transport has also been conducted in academia, with some of the most pronounced findings coming from Cranfield University and Chalmers University of Technology.

Many of the most critical specific technology developments for VCEs happened in the 1960s and 1970s. Some developments by GE documented in the open literature are presented here. The VCE arguably began in the 1960s with the Variable Pumping Compressor (VAPCOM) concept. This was a conventional turbofan that could drive its BPR nearly to zero by using stator vanes in the fan, compressor, and turbines. In the 1970s, a three-spool Modulating Bypass Ratio (MOBY) engine was developed to specifically address spillage drag and aft body drag (i.e. putting the focus on installed thrust rather than uninstalled thrust). This engine had three compressors, three turbines, three spools, and two bypass ducts (one of which had a duct burner). Though this cycle was very versatile, its complexity made it heavy, costly, and risky. Many of MOBY’s concepts were later transitioned to a two-spool, three stream cycle by using a split fan approach in an attempt to reduce complexity. Another useful feature introduced in the 1970s was the Variable Area Bypass Injector (VABI). Rather than using open/closed type diverter values at the entrance
to a duct, the VABI allowed changeable areas for the two incoming flows at the mixing plane (Johnson, Variable Cycle Engine Concepts 1996).

Largely based on published work by industry, academia has continued research into VCEs, including selective bleed engines. This engine is a two-spool afterburning turbofan with two low-spool driven compressors and one high-spool driven compressor. It is a three stream architecture, having two bypasses around the core. The outermost bypass stream discharges through a separate nozzle, while the inner bypass mixes with the core flow before being ejected through the primary convergent-divergent nozzle (Ulizar and Pilidis 1997). Later dubbed a double bypass engine, the selective bleed architecture is very similar to the one used in this research. The example double bypass engine shown in Fig. 2 was designed for two operating modes—a subsonic mode and a supersonic mode.

![Double bypass engine](image)

**Fig. 2** Double bypass engine (Aleid and Pilidis 1998)

This engine used variable geometry to effectively close off flow to either the inner bypass (subsonic mode) or to the outer bypass (supersonic mode) (Aleid and Pilidis 1998). The cycle used in the research presented in the later chapters strives to have a more continuous
operating mode, as it has the additional requirement that both the second stream (inner 
bypass) and third stream (outer bypass) maintain a minimum flow level.

**Overview of Engine Power Extraction**

The vast majority of the energy available on an aircraft resides in the fuel. The main 
engines are the primary consumers of the fuel, and are therefore responsible for supplying 
nearly all of the aircraft’s power needs. For turbofans, a substantial portion of total engine 
power goes toward producing thrust (thrust power). The two other ways of producing 
power from the engine are through bleed air extraction and shaft power extraction. The 
latter tends to be subdivided into hydraulic power (from shaft or gearbox driven pumps), 
mechanical power (directly shaft or gearbox driven devices such as fuel pumps) and 
electrical power (from shaft or gearbox driven generators).

While not intending to endorse or condemn a particular power extraction approach, 
the following discussion describes the uses for various types of power extraction as well as 
the trend toward more electric architectures. The European Union’s (EU) Power Optimized 
Aircraft (POA) program conducted an assessment of the value of electrical systems versus 
conventional systems on board an aircraft. Their conclusion was that, based on current 
technology, electrical components tended to be heavier than their conventional 
counterparts to perform each individual function (Falerio 2005). Furthermore, interactions 
between components on an electrical bus could be more challenging to adequately address. 
By tying the electrical system to so many aspects of the aircraft, fault tolerance, redundancy, 
and certification requirements became important considerations (AbdElhafiez and Forsyth 
2009). However, the electrical systems tended to be more energy efficient. This was due to: 
1) the inherently higher efficiencies of many electrical devices; 2) lower losses in electrical 
cabling than in hydraulic or pneumatic lines; 3) the on-demand nature of electrically driven
devices compared with always-on conventional devices that only need to operate for a small portion of the flight (e.g. the landing gear); 4) electrically driven devices can be more precisely designed for a function—they do not need to include pressure or flow reducing components, for instance; 5) maintaining fewer types of physical systems could reduce the maintenance logistics/spare parts trail; and 6) electrical devices tend to be more reliable (though a counterargument is that their failures tend to be more catastrophic) (Falerio 2005). Overall, a partial adoption of a more electric architecture was likely to be required as a first risk reduction step but was unlikely to show significant advantages since it would still require the weight and volume of traditional pneumatic and hydraulic systems to also be included on the aircraft; a full adoption of a more electric architecture that would completely eliminate the conventional approach (pneumatic and hydraulic systems) would show significant weight and efficiency benefits.

High pressure bleed air has traditionally been an important aspect of aircraft engine design. Because any flow not available to produce thrust is a debit to TSFC, there is a desire to minimize bleed. However, using the compressors in the main aircraft engine generally is more economical than including separate hardware to create high pressure pneumatic power. The bleed air has traditionally been used for cabin pressurization, the environmental control system (air conditioning), ice protection, onboard inert gas/oxygen generation, and cooling of some avionics (Wheeler 2009).

More recent aircraft programs have sought to reduce or eliminate the need for main engine bleed air. However, this cannot be done in isolation from the rest of the aircraft. There has been a push to assess the entire aircraft as a system during the design process. The driving force behind these efforts is that the federated subsystem approach, which optimizes components in isolation and then integrates them only to form a final product,
results in a suboptimal aircraft system. Furthermore, integration problems creep up due to poor interface definition or as a result of emergent behavior in complex systems (Ericsen 2008). The approach to addressing these problems is to design the aircraft as a system from the beginning, taking subsystem interactions into account early, and looking to the system-level for optimization metrics (such as minimal energy consumption to complete a given mission). A key aspect of this approach is that the transient performance of the integrated system, not just the steady-state operating points, can be the driving force behind dynamic interactions which introduce new failure modes and can dictate peak demands.

Using the POA program as an example, Fig. 3 suggests how electrification of more functions on board the aircraft can result in system-level benefits.

![Fig. 3 Conventional versus a potential power optimized architecture (Falerio 2005)](image)

Specific to bleed air reduction, the approach is to use an electrically driven vapor cycle system for environmental control and electrically driven wing anti-ice systems (Falerio 2005). Boeing, in developing the 787 Dreamliner, has pushed this a step further by going to a no-bleed system. In the no-bleed system, engine bleed air is only used for hydraulic
reservoir pressurization (non-consuming) and engine cowl ice protection as needed (wing anti-ice is handled by on-demand heating blankets bonded to the wing skin). Ram air is used to refresh cabin air and electrically driven compressors provide the cabin pressurization. This approach uses on-demand, variable speed compressors to provide exactly the required capacity instead of pulling a constant amount of bleed air and dumping the excess overboard as waste (Sinnett 2007). Though it is difficult to quantify all benefits of reducing engine bleed in lieu of increased shaft power extraction (or make broad sweeping generalizations that would hold for all aircraft types and missions), some studies indicate that a 2% reduction in TSFC could be achieved by using a more electric approach instead of a bleed air approach to drive the environmental control system (Slingerland and Zandstra 2007).

Even though the trend is to reduce bleed air extraction to improve TSFC, significant bleed air is still required for cooling within the engine. While this air is not completely lost outside of the engine control volume, by bleeding it and reinjecting it in a different location, it does not produce its full work potential (and may in fact require additional compression to be reinjected). Typical uses for this bleed air are for inlet flow control, burner liner cooling, turbine stator cooling, turbine rotor cooling, and nozzle cooling (Simmons 2009). In the case of turbine cooling, some of the cooling air is available to do useful work in the turbine, but not all. There are several ways to account for this. The method used in this research is by bookkeeping non-chargeable and chargeable turbine cooling air. This approach is described in further detail in the Cooled Cooling Air (CCA) section in the next chapter. The concept of CCA itself refers to cooling the compressor bleed air before using it as cooling flow elsewhere in the engine when the high-pressure compressor (HPC) exit bleed flow temperature is too high to be used for cooling directly (Jones and Boyle 2009).
As described above, shaft power extraction is a category that encompasses three large subcategories: hydraulic, mechanical, and electrical power. The first of these, hydraulic power, is undergoing a major industry-wide change. In conventional aircraft, shaft driven pumps provide continuous hydraulic power. They have a centralized hydraulic accumulator, and hydraulic distribution lines run to devices such as various flight control surface actuators, landing gear actuators, door actuators, and brakes (Wheeler 2009). This approach suffers from problems such as high line losses, more power than needed for some hydraulically driven devices, and short maintenance intervals. Modern aircraft are moving away from central hydraulic systems toward electric actuation. One major step in this direction was made for the Airbus A380 and the Lockheed Martin F-35 Joint Strike Fighter in the move to electrohydrostatic actuators for primary flight control (Falerio 2005). These actuators use small, localized hydraulic systems located at each actuator. Rather than having hydraulic lines running to each of these actuators, only electrical power lines are required. This was not a complete move away from the traditional approach, however, as both of these aircraft still maintain a hydraulic system for other devices such as the landing gear. A potential architecture that uses localized hydraulic systems for all actuation is shown in the potential optimized architecture of Fig. 3.

Development and technology maturation efforts look to take a step further by developing high-performance electromechanical actuators that do not use hydraulic fluid at all. While eliminating the need for hydraulic fluid is an obvious advantage, it does come at a cost. Electromechanical actuators can regenerate power as well as draw power, requiring careful design of the electrical system. Also of note, but not shown in Fig. 3, is that engine actuation has traditionally been accomplished using fueldraulics (a hydraulic system that uses the fuel as the working fluid). Engine manufacturers have been more resistant to
changing their actuation approach to a more electric approach, largely due to the harsh operating environment around the engine.

The second subcategory of shaft driven power extraction is mechanical power extraction. This generally refers to devices driven by the gearbox that is connected to the engine through the tower shaft. These devices, simply by being physically coupled to the engine, run continuously but vary in speed proportionally to the engine’s core speed. Oil and fuel pumps are prime examples of such devices (Wheeler 2009). A move toward electric variable displacement fuel pumps would again provide the benefit of an on-demand system better fitted to the capacity required. In addition to simply reducing the energy consumption, on-demand pumps would reduce the heat load that needs to be dissipated.

The final subcategory of shaft power extraction is electrical power generation. In a move toward a more electric aircraft, this load would increase in exchange for reductions in hydraulic, mechanical, and pneumatic loads. In conventional aircraft, electrical power is required for cabin lighting, avionics, fans, etc. It has traditionally been on the order of a few hundred kilowatts, but in a move toward a more electric aircraft architecture, it could easily exceed a megawatt (Wheeler 2009). High-pressure (HP) spool generators are traditionally attached to the gearbox, though in a more electric architecture, there is a push toward embedded starter/generators as indicated in Fig. 3. These starter/generators are contained within the engine itself and can be used both to generate electric power from the engine and to enable main engine start/in-flight restart by acting as a motor that spins up the HP spool (powered by the auxiliary power unit or by batteries) until light-off can occur. Furthermore, embedded starter/generators could lead to the elimination of the gearbox and ground support equipment required for main engine start (AbdElhafez and Forsyth 2009). Low-pressure (LP) spool generators can be driven through a gearbox or contained
in a nose-cone or tail-cone at either end of the engine. Shaft driven generators (directly mounted or connected through a gearbox) spin at a speed proportional to either the engine's core speed or fan speed. For this reason, the output of the generators typically requires power conditioning electronics (Wheeler 2009).

Some of the impact of bleed air power and shaft power extraction from an engine can be seen in Fig. 4.

![Figure 4: Effects of bleed and shaft power extraction on the HPC (J. Kurzke 1992)](image)

The graphic shows the effect that extracting HPC bleed or HP spool shaft power has on HPC operation at a low altitude and a high altitude. Bleed air extraction has a favorable impact on HPC stability as it drives down the pressure ratio at nearly constant corrected speed, moving the operating point away from the surge line. This is also the reason that bleed air extraction can be useful during engine start. However, it should certainly be reiterated that bleed air extraction has a detrimental impact on overall engine performance (e.g. increasing the TSFC) as it makes less air available to produce thrust. On the other hand, shaft power
extraction moves the operating point toward the surge line by dragging down the spool speed. As seen in Fig. 4, the trend of bleed or shaft power extraction is the same at sea-level static (SLS) and at high altitude operations, though the magnitudes differ significantly for the same power extraction values. For the example engine, the same shaft power offtake at SLS corresponded to only 0.5% of total HP spool power but is 4% of total HP spool power at high altitude cruise conditions. Similarly, the same HPC exit bleed flow rate represented 1% of HPC flow at SLS, but 6.5% of HPC flow at the high-altitude cruise conditions (J. Kurzke 1992).

While the example illustrated in Fig. 4 is representative, it does not present the entire picture. In the study engine, there are three compressors and two shafts capable of power extraction. The customer bleed air requirements are minimal (but non-zero) in these studies. A base level of power extraction is used from the HP spool, but transient shaft power extraction is from the LP spool. LP shaft power extraction has been shown to have more available power, operability benefits, and a less detrimental impact on TSFC at some operating points—especially when combined with HP power extraction (Zähringer, Stastny and Arday 2009). In contrast to the behavior shown in Fig. 4, LP power extraction can provide an operating point with increased HPC surge margin. However, the operation to extract the additional power from the LP shaft moves the operating point transiently toward the surge line on the HPC map (Corbett, et al. 2007). The reason for differing behaviors in the HPC between HP and LP spool power extraction lies with the coupling. The two shafts are not physically coupled through a gearbox; they are tied together instead by an aerothermodynamic coupling since the core flow passes consecutively through the high-pressure turbine (HPT) and low-pressure turbine (LPT) (Norman, et al. 2008).
II. MODEL DEVELOPMENT

The following sections outline the development of the model used in the studies presented in Chapter III. The engine modeling framework is first described, followed by a description of the engine cycle itself. Then an overview of the physics captured in the model is presented and a discussion is offered about the physics or empirical effects not included in this model. This chapter concludes with a description of the engine controller that was developed for this effort.

Engine Simulation Computational Framework

The Numerical Propulsion System Simulation (NPSS) is a computational framework/simulation environment developed by NASA under a cooperative effort with other government agencies, academia, and the propulsion industry. While it is a flexible framework that can be used to model many types of systems, the built-in thermodynamic tools make NPSS particularly well-suited for developing engine simulations. It is an object-oriented package, which fits well with the modular nature of engines (Lytle 2000). The tool is inherently a zero-dimensional aerothermodynamic cycle analysis tool, but it is also capable of interfacing with higher fidelity tools (such as computational fluid dynamics packages) using an approach referred to as zooming (Sampath, et al. 2004).

Using the modeling tool itself consists of: 1) selecting a thermodynamic package; 2) defining and instantiating components; 3) linking components together; 4) configuring the solver; and 5) executing a simulation. The thermodynamic package defines fluid properties from thermodynamic state conditions. Defining components involves establishing the
equations that describe the behavior of an element. This is followed by creating specific instances of these defined elements such as compressors, combustors, nozzles, and other engine components. Linking components defines the physical connections (e.g. shaft connections between compressors and turbines) and flow path connections (e.g. flow moves from the inlet to the compressor to the combustor, etc.). Configuring the solver refers to setting up the balance equations (e.g. conservation of mass, zero net torque on a shaft in steady-state, static pressure balance at a mixing plane, etc.) and adding pairs of dependent and independent variables to achieve target performance. For example, fuel flow might be added as an independent and target thrust added as a dependent. Constraints can also be added to solver dependents. Using the target thrust example, the dependent might be constrained such that the desire to achieve the commanded thrust is trumped by a maximum temperature constraint. Additional items, such as data viewers/loggers or high-level performance calculations, can also be added to the model. Finally, the model is executed to perform calculations and converge on a solution (S. M. Jones 2007).

The typical usage case for NPSS is to run a design point followed by a series of off-design points. In design mode, the user provides performance type numbers and the result of execution is sizing type numbers. After running that initial design point, the sizing type numbers from the design point are preserved and used for off-design analysis where the performance type numbers are the outputs. In addition to running single point off-design calculations, NPSS is capable of running transiently. In this case, balance equations are replaced with integrated variables in the solver simply by setting the solutionMode option variable to TRANSIENT (assuming that the required data has also been provided) (S. M. Jones 2007).
During the course of this work, it was determined that NPSS was not particularly well suited for control of the engine plant. For this reason, the thermodynamic cycle model was developed in NPSS, but position commands for variable geometry, fuel flows, etc. were passed into the model from an external source. As used in this work, the NPSS cycle model is controlled externally by Simulink. An additional factor in this decision is that future work will likely include integrating the study engine with additional Simulink models. The communication between NPSS and Simulink is facilitated by incorporating the NPSS model as an S-function within the Simulink model. A configuration file defines the NPSS parameters that are set by Simulink and the variables that are passed from NPSS back to Simulink. Though not the only viable approach, all data logging was done in Simulink (including NPSS cycle performance variables, which were passed out of the S-function). A more detailed discussion of the control logic and approach is included in the Controls section.

**Study Engine Model**

Based on the discussion in the previous section, an engine cycle model was developed in NPSS and its control characteristics were developed in Simulink. The overall engine cycle schematic is given in Fig. 5. This figure includes numbered flow stations that strived to be consistent with the Society of Automotive Engineers (SAE) Aerospace Standard AS755D’s Alternate Numbering System for Reduced Ambiguity (SAE Aerospace 2004). Descriptions of the station numbers used for this engine is given in Table 1.

Similarly, component instance names were chosen to be as consistent with the SAE Aerospace Recommended Practice ARP5571A as possible (SAE Aerospace 2008). In Fig. 5, the various flow paths through the engine are apparent. All airflow enters the inlet and is compressed by the fan.
Fig. 5  Study engine cycle schematic
<table>
<thead>
<tr>
<th>Station #</th>
<th>Location Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>Free stream (not shown in Fig. 5)</td>
</tr>
<tr>
<td>010</td>
<td>Inlet entrance</td>
</tr>
<tr>
<td>011</td>
<td>Inlet element exit&lt;br&gt;Inlet flow control bleed element entrance</td>
</tr>
<tr>
<td>015</td>
<td>Inlet flow control bleed element exit&lt;br&gt;Front frame (duct element) entrance</td>
</tr>
<tr>
<td>020</td>
<td>Front frame (duct element) exit&lt;br&gt;Fan entrance</td>
</tr>
<tr>
<td>021</td>
<td>Fan exit&lt;br&gt;First splitter element entrance</td>
</tr>
<tr>
<td></td>
<td><em>(looking at fan exit flow bypassed to the 3rd stream)</em></td>
</tr>
<tr>
<td>230</td>
<td>First splitter element exit into the 3rd stream duct&lt;br&gt;Duct D230 element entrance</td>
</tr>
<tr>
<td>231</td>
<td>Duct D230 element exit&lt;br&gt;Bleed B231 element entrance</td>
</tr>
<tr>
<td>240</td>
<td>Bleed B231 element exit&lt;br&gt;3rd stream heat exchanger (to cool external loads) entrance</td>
</tr>
<tr>
<td>245</td>
<td>3rd stream heat exchanger (to cool external loads) exit&lt;br&gt;Cooled cooling air heat exchanger entrance</td>
</tr>
<tr>
<td>250</td>
<td>Cooled cooling air heat exchanger exit&lt;br&gt;Duct D250 element entrance</td>
</tr>
<tr>
<td>251</td>
<td>Duct D250 element exit&lt;br&gt;Bleed B251 element entrance</td>
</tr>
<tr>
<td>270</td>
<td>Bleed B251 element exit&lt;br&gt;3rd stream nozzle entrance</td>
</tr>
<tr>
<td>280</td>
<td>3rd stream nozzle throat (not shown in Fig. 5)</td>
</tr>
<tr>
<td>290</td>
<td>3rd stream nozzle exit</td>
</tr>
<tr>
<td></td>
<td><em>(looking at fan exit flow continuing toward core)</em></td>
</tr>
<tr>
<td>022</td>
<td>First splitter element exit continuing toward core&lt;br&gt;Duct D022 element entrance</td>
</tr>
<tr>
<td>023</td>
<td>Duct D022 element exit&lt;br&gt;Low-pressure compressor entrance</td>
</tr>
<tr>
<td>024</td>
<td>Low-pressure compressor exit&lt;br&gt;Second splitter element entrance</td>
</tr>
<tr>
<td></td>
<td><em>(looking at low pressure compressor exit flow bypassed to the 2nd stream)</em></td>
</tr>
<tr>
<td>130</td>
<td>Second splitter element exit into 2nd stream duct&lt;br&gt;Duct D130 element entrance</td>
</tr>
<tr>
<td>131</td>
<td>Duct D130 element exit&lt;br&gt;Bleed B_HPCleak element entrance</td>
</tr>
<tr>
<td>Station #</td>
<td>Location Description</td>
</tr>
<tr>
<td>----------</td>
<td>---------------------</td>
</tr>
<tr>
<td>140</td>
<td>Bleed B_HPCleak element exit 2nd stream heat exchanger (to cool external loads) entrance</td>
</tr>
<tr>
<td>150</td>
<td>2nd stream heat exchanger (to cool external loads) exit Duct D150 element entrance</td>
</tr>
<tr>
<td>151</td>
<td>Duct D150 element exit Bleed B151 element entrance</td>
</tr>
<tr>
<td>160</td>
<td>Bleed B151 element exit Mixer bypass flow entrance  <em>(looking at low pressure compressor exit flow continuing toward core)</em></td>
</tr>
<tr>
<td>025</td>
<td>Second splitter element exit continuing toward core Duct D025 element entrance</td>
</tr>
<tr>
<td>026</td>
<td>Duct D025 element exit High-pressure compressor entrance</td>
</tr>
<tr>
<td>030</td>
<td>High-pressure compressor exit Primary combustor entrance</td>
</tr>
<tr>
<td>040</td>
<td>Primary combustor exit Bleed B041 element (for HPT non-chargeable cooling flow) entrance</td>
</tr>
<tr>
<td>041</td>
<td>Bleed B041 element (for HPT non-chargeable cooling flow) exit High-pressure turbine (rotor) entrance</td>
</tr>
<tr>
<td>042</td>
<td>High-pressure turbine (rotor) exit Bleed B042 element (for HPT chargeable cooling flow) entrance</td>
</tr>
<tr>
<td>044</td>
<td>Bleed B042 element (for HPT chargeable cooling flow) exit Bleed B045 element (for LPT non-chargeable cooling flow) entrance</td>
</tr>
<tr>
<td>045</td>
<td>Bleed B045 element (for LPT non-chargeable cooling flow) exit Low-pressure turbine (rotor) entrance</td>
</tr>
<tr>
<td>049</td>
<td>Low-pressure turbine (rotor) exit Bleed B049 element (for LPT chargeable cooling flow) entrance</td>
</tr>
<tr>
<td>050</td>
<td>Bleed B049 element (for LPT chargeable cooling flow) exit Duct D050 element entrance</td>
</tr>
<tr>
<td>060</td>
<td>Duct D050 element exit Mixer core flow entrance  <em>(looking at mixed flow)</em></td>
</tr>
<tr>
<td>061</td>
<td>Mixer exit Afterburner entrance</td>
</tr>
<tr>
<td>065</td>
<td>Afterburner exit Duct D065 entrance</td>
</tr>
<tr>
<td>068</td>
<td>Duct D065 exit Primary (mixed flow) nozzle cooling bleed flow element entrance</td>
</tr>
<tr>
<td>070</td>
<td>Primary (mixed flow) nozzle cooling bleed flow element exit Primary (mixed flow) nozzle entrance</td>
</tr>
<tr>
<td>080</td>
<td>Primary (mixed flow) nozzle throat (not shown in Fig. 5)</td>
</tr>
<tr>
<td>090</td>
<td>Primary (mixed flow) nozzle exit</td>
</tr>
</tbody>
</table>
At the exit of the fan, flow is split between that which enters the outer bypass duct (3rd stream) and that flow which continues to the low-pressure compressor (LPC). The 3rd stream flow passes through heat exchangers (HX) in the duct and is ejected through its own nozzle. The flow that enters the LPC splits again at the exit of the LPC, with some flow bypassing the core and the remainder entering the HPC. After exiting the HPC, the core flow goes through the combustor, HPT, and LPT. Then, this core flow is mixed with the inner bypass (2nd stream) flow at the variable area mixer. The mixed flow passes through the afterburner and is ejected through the primary nozzle. Also seen in Fig. 5 are the available customer bleed flows from the fan, LPC, and HPC, the available customer shaft power extractions from both the LP and HP shafts, and engine internal bleed flows. LPC exit bleed is injected into the inlet for flow control. HPC flow is bled from a middle stage for LPT and nozzle cooling since the pressure required to reinject the flow is not as high. Some HPC exit bleed is used directly for HPT cooling and some is directed to a HX in the 3rd stream before being used for HPT cooling. Finally, the components with variable geometry features are highlighted in Fig. 5: the fan, LPC, HPC, HPT, LPT, Mixer, and both Nozzles.

The top-level view of the study model is given in Fig. 6. In this figure, the NPSS cycle model is contained as an S-function in the subsystem labeled NPSS_3_Stream_Engine. The other main blocks at the top level of the study model are the AircraftDragPolar, LargePeriodicLoad, EngineController, and BusToFile. The BusToFile block is simply a means of logging model data. Detailed discussion of the LargePeriodicLoad and EngineController blocks are left to the Customer Power Extraction and Controls sections below. The AircraftDragPolar model is not the focus of this study and is therefore not discussed in great detail. Briefly, the AircraftDragPolar subsystem provides the aircraft thrust requirement for every point in the mission.
Fig. 6 Simulink model top level block diagram
This thrust requirement is a function of the operating conditions (altitude, Mach number, and day type) and the current aircraft mass. The integrated engine fuel burn rate is used to update the aircraft mass continuously throughout the mission. The block does not close the loop on thrust control; it simply requests thrust to maintain flight regardless of whether the engine can provide it.

**Physics Modeled**

In developing the NPSS engine model, many of the standard (supplied) elements were utilized. The underlying equations in those elements have been fully vetted and can be used with confidence. Elements that were defined specifically for use in this modeling effort are presented in more detail in the sections that follow. A particular focus is placed on those phenomena which may not be captured in all simulations.

**Ambient and Inlet Start**

Provided with NPSS is an Ambient element which calculates flight condition properties. Used together with the ambient element is the Inlet Start element. Together, these elements create a flow and define its properties (total and static temperature, total and static pressure, density, Mach number, dynamic pressure, etc.) from a set of inputs. Typical inputs (though there are other options) to set these properties are altitude and Mach number. A switch is available to select between day types (standard day, polar day, 10% hot day, etc.) defined using MIL-HDBK-310. Additionally, airflow (or corrected airflow) is specified at the inlet start element, though this value is frequently controlled by the solver (Numerical Propulsion System Simulation Consortium 2010, Reference Sheets).

**Inlet**

The inlet decelerates the incoming flow (which may include shock waves) to a reasonable subsonic Mach number for efficient engine operation. In this process, the
dynamic pressure of the incoming flow is transformed into a static pressure increase in the flow. At the same time, the total pressure decreases. The ability of the inlet to minimize the total pressure losses is referred to as inlet ram recovery (North Atlantic Treaty Organisation, Research and Technology Organisation 2007). The inlet element contains a placeholder for a ram pressure recovery calculation to be specified by the user. One approach to the ram recovery is to specify a constant value, usually in the range of 0.95 to 1.0 for all flight conditions (Simmons 2009). Another approach is to use the military specification MIL-E-5007D which defines ram pressure recovery of 1.0 for subsonic speeds and curves for supersonic and hypersonic speeds (Department of Defense 1973). Because both of these are merely approximations, the approach used for this research was to combine the two:

\[ \pi_{rec} = \pi_{rec,max} \cdot \pi_{rec.spec} \]  \hspace{1cm} \text{Eq. 11}

\[ \pi_{rec} = \begin{cases} 
0.97, & MN \leq 1.0 \\
0.97 - 0.075 \cdot (MN - 1.0)^{1.35}, & 1.0 < MN \leq 5.0 
\end{cases} \]  \hspace{1cm} \text{Eq. 12}

where \( \pi_{rec} \) is the inlet ram total pressure recovery factor (total pressure ratio), \( \pi_{rec,max} \) is a constant maximum achievable pressure recovery ratio (set at 0.97 for the study model), \( \pi_{rec.spec} \) is the pressure recovery from the military specification, and \( MN \) is the aircraft Mach number (or equivalently the incoming air flow Mach number from an engine frame of reference).

**Closed Loop Bridge**

The Closed Loop Bridge is simply an element that facilitates moving flow upstream in the engine. NPSS solves components one at a time, using the flow properties at each input port of a component to calculate the component performance and output port values. However, there are instances when some of the flow introduced to a particular element has not yet been calculated. An example of this behavior is the inlet flow control bleed. In this
case, a bleed element adds LPC bleed flow to the airflow exiting the inlet element. Since the LPC is downstream of the point where this bleed flow is inserted, the properties of this bleed air have not yet been calculated for the current operating point (steady-state or transient operating point). In fact, since the flow entering the LPC is a function of the flow that leaves the inlet flow control element, the LPC bleed flow is dependent on itself. This is referred to as an algebraic loop (Martin 2009). To handle this, the closed loop bridge element provides initial guesses at the point of insertion for the flow that is calculated further downstream. The properties guessed using the closed loop bridge are added to the solver, requiring it to iterate until the upstream guesses for the bleed flow match the computed bleed flow properties.

**Bleed**

A Bleed element facilitates the extraction or introduction of bleed flows. Additionally, it can be used to add or remove heat from the flow. Any number of inflow or outflow ports can be created to add flow to or remove from the element’s main flow. Incoming flows are mixed such that mass and energy are preserved, but no momentum calculations are done (Numerical Propulsion System Simulation Consortium 2010, Reference Sheets). In the study model, heat is not added or removed using bleed elements.

**Duct**

A Duct element models flow in a duct. There are two aspects to the model: pressure loss and heat addition. The pressure loss is modeled as an adiabatic process (constant enthalpy). The heat addition term simply increases (or decreases) the flow enthalpy. Either of these effects can be turned off or set to zero (Numerical Propulsion System Simulation Consortium 2010, Reference Sheets). In the study model, some duct elements use a constant normalized pressure drop, $\frac{dp}{p}$, and some use the following relationship:
\[
\frac{dP}{p} = \text{const} \cdot (MN)^2
\]
Eq. 13

where \(\frac{dP}{p} = \frac{P_{t,\text{out}} - P_{t,\text{in}}}{P_{t,\text{in}}}\) is the normalized total pressure drop, \(P_{t,\text{in}}\) is the total pressure entering the duct, \(P_{t,\text{out}}\) is the total pressure exiting the duct, \(MN\) is the Mach number of the flow entering the duct, and \(\text{const}\) is a constant that is computed at the design point such that Eq. 13 produces the desired pressure drop at the design point. The value of \(\text{const}\) is then held for use in off-design calculations (Numerical Propulsion System Simulation Consortium 2010, Reference Sheets).

**Fan/Compressor**

The basic element for fans and compressors is identical as they are both mechanical compression devices with the same basic operating principles. The function of the element is to calculate compressor performance (and thus the compressor outputs) from the incoming flow conditions and map definitions that are specific to each fan or compressor. There is more than one possible form for the maps, but the form used in the study model is that both pressure ratio, \(PR\), and corrected mass flow rate, \(\dot{m}_c\), are given in tables as a function of corrected speed, \(N_c\), and R-line as illustrated generically in Fig. 7. The R-line is an arbitrary second index of lines drawn approximately parallel to the surge line on a compressor map. These arbitrarily drawn R-lines are used because: 1) some compressor maps have hooked speed lines such that there is more than one corrected mass flow rate for a given pressure ratio (i.e. not always possible to uniquely find \(\dot{m}_c = f(N_c, PR)\)); and 2) some compressor maps have straight (vertical) portions of the speed lines such that there is more than one pressure ratio for a given corrected mass flow rate (i.e. not always possible to uniquely find \(PR = f(N_c, \dot{m}_c)\)). The compressor adiabatic efficiency is also mapped as a function of \(N_c\) and R-line (S. M. Jones 2007).
The compressor element also has the capability to scale the map values to meet desired performance at the design point. The approach used in scaling maps is discussed in Appendix C of NASA/TM—2007-214690 (S. M. Jones 2007). Caution is urged when allowing the design point to compute scale factors significantly different than unity, as it can lead to unrealistic off-design results (Simmons 2009). An additional dimension is also available for compressor maps to account for variable geometry. In this case, maps are said to be stacked or layered such that there is a collection of maps corresponding to the various vane angles. The map for each vane angle has pressure ratio, corrected mass flow, and efficiency as a function of R-line and corrected speed. In this case, the performance parameters are each cast as: $f(N_c, R\text{-line}, \alpha)$ where $\alpha$ is the vane angle.
Since the compressor/fan is treated as a lumped element rather than calculated stage-by-stage, bleed air can be extracted at any location in the compressor. The bleed location is specified as a fraction of the pressure rise and enthalpy rise across the compressor element (the fractions can be the same or different for pressure and enthalpy). The flow rate for bleed air extraction can either be specified as a fraction of the flow entering the compressor or as an absolute flow rate.

The compressor has both a mass flow rate entering from the upstream component and a mass flow rate computed from the compressor map lookup table (and scale factors). From the conservation of mass, the main flow rate exiting the compressor must be equal to the flow rate entering it minus any bleed flow extraction. Therefore, a solver balance pair is added for each compressor or fan to ensure continuity of flow (Numerical Propulsion System Simulation Consortium 2010, Reference Sheets). The independent is the R-line parameter and the dependent condition is that the entering mass flow rate be equal to the mass flow rate computed from the map (bleeds are extracted after the lookup tables determine the exit conditions).

The compressor element consumes torque and requires speed as an input, so it is connected to a Shaft element. Additionally, the user can specify the inertia for the compressor for use in transient operation. This is discussed in further detail in the Shaft element section. The compressor also can include the Heat Soak effect, which is described in more detail in the appropriate section below.

**Splitter**

The Splitter element allows the flow to be split into two streams. While the Bleed element is also capable of performing this function, the splitter is capable of applying
separate pressure loss terms (in $\frac{dP}{p}$ form) for each of the output streams. Additionally, the splitter calculates the BPR:

$$BPR = \frac{\dot{m}_{\text{bypass}}}{\dot{m}_{\text{core}}}$$

Eq. 14

where $\dot{m}_{\text{bypass}}$ is the flow rate exiting the splitter that enters the bypass duct and $\dot{m}_{\text{core}}$ is the mass flow rate that continues to the core (Numerical Propulsion System Simulation Consortium 2010, Reference Sheets). The value of the BPR is normally controlled by the solver as an independent variable. The corresponding dependent is often a static pressure balance at a mixing plane. This is discussed further in the Mixer section.

**Duct with Air Gap and Heat Exchanger (Specified $\dot{Q}$)**

A custom NPSS Element was developed to model one side of a HX in a fan duct. Specifically, this refers to an air-to-air HX located in the 3rd stream duct that has a heat transfer rate into the duct flow, $\dot{Q}$, specified externally in lieu of requiring details about the hot side of the HX. This element assumes that there are $n$ identical HXs distributed around the duct annulus and that there are then also $n$ identical air gaps between the HXs. It is further assumed that the total duct flow, $\dot{m}_{\text{duct}}$, is evenly split among the $n$ gap/HX pairs. This simplifies the modeling such that pressure loss maps (referred to standard conditions) are then only required for one such air gap and one such HX. The split of the duct flow between the air gap and HX is solved iteratively until the total pressure drop of the air squeezing through the gap is the same as the total pressure drop for the air passing through the HX. This balance is contained within the element, not in the global NPSS solver.

The specified $\dot{Q}$ is transferred into the portion of the flow passing through the HX (cold side) and no heating is assumed to take place in the air gap (not even frictional heating). Again this heat transfer rate is divided evenly among the $n$ identical HXs. To solve
the element, a guess is made for the fraction of total duct flow that passes through the HX ($frac_{HX}$). For calculations after the first converged point, the initial guess for $frac_{HX}$ is simply its previous converged value (steady-state or transient). The guessed mass flow rates through the HX and air gap are then:

$$m_{HX} = \frac{\dot{m}_{duct} \cdot frac_{HX}}{n}$$  \hspace{1cm} \text{Eq. 15}$$

$$m_{GAP} = \frac{\dot{m}_{duct} \cdot (1 - frac_{HX})}{n}$$  \hspace{1cm} \text{Eq. 16}$$

where $\dot{m}_{HX}$ is the mass flow rate through one of the $n$ identical HXs and $\dot{m}_{GAP}$ is the mass flow rate through one of the $n$ identical air gaps. Within one HX, the heat transfer rate is then $\dot{Q}_{HX} = \dot{Q}/n$. The output total temperature of the cold (duct) side of the HX, $T_{out,HX\,cold}$, is then:

$$T_{out,HX\,cold} = T_{in,cold} + \frac{\dot{Q}_{HX}}{\dot{m}_{HX} \cdot c_{p,cold}}$$  \hspace{1cm} \text{Eq. 17}$$

where $T_{in,cold}$ is the total temperature of the cold duct flow entering the HX and $c_{p,cold}$ is the specific heat (at constant pressure) of the incoming duct flow.

The drop in total pressure through the HX, $\Delta P_{HX,cold}$, is found from table data, the average total temperature, the average total pressure in the HX, and constants:

$$\Delta P_{HX,cold} = \Delta P_{corr,HX,cold} \cdot \frac{P_{STD}}{P_{avg,HX,cold}}$$  \hspace{1cm} \text{Eq. 18}$$

where $\Delta P_{corr,HX,cold} = f(\dot{m}_{HX})$ is the referred total pressure loss from a lookup table (as a function of HX cold flow rate), $P_{STD}/P_{avg,HX,cold}$ is the ratio of standard pressure to standard temperature (at sea level), $T_{avg,HX,cold} = T_{out,HX\,cold} + T_{in,cold}/2$ is the average HX total pressure.
temperature, and the average total pressure in the HX is
\[ P_{\text{avg,HX cold}} = \frac{P_{\text{out,HX cold}} + P_{\text{in,cold}}}{2} \]. The incoming flow's total pressure, \( P_{\text{in,cold}} \), is known, but \( P_{\text{out,HX cold}} \) is determined by the pressure drop itself. This requires a few internal iterations for \( \Delta P_{\text{HX cold}} \) to settle.

The total temperature in the air gap is assumed to be constant (equal to \( T_{\text{in,cold}} \)), and when the air gap and HX are balanced the exiting total pressures are equal (and thus the average pressures are as well). Therefore, the air gap total pressure drop, \( \Delta P_{\text{GAP,cold}} \), is:

\[ \Delta P_{\text{GAP,cold}} = \Delta P_{\text{corr,GAP,cold}} \cdot \frac{P_{\text{STD}}}{T_{\text{STD}}} \cdot \frac{T_{\text{in,cold}}}{P_{\text{avg,HX cold}}} \] \hspace{1cm} \text{Eq. 19}

where \( \Delta P_{\text{corr,GAP,cold}} = f(\dot{m}_{\text{GAP}}) \) is the referred pressure loss in the air gap from a table (as a function of gap mass flow rate). Updates are then made to the value guessed for \( \dot{m}_{\text{HX}} \) (and Eq. 15-Eq. 19 are solved again) until \( \Delta P_{\text{GAP,cold}} \) and \( \Delta P_{\text{HX,cold}} \) are equal within tolerance.

At this point, the total pressure is known at the element exit. The mass flow rate through all HXs collectively is \( n \cdot \dot{m}_{\text{HX}} \). Heat is added directly to the enthalpy of the flow passing through the HX rather than using the calculated \( T_{\text{out,HX,cold}} \) to avoid the introduction of small errors created by assumptions such as using the \( c_p \) value for the incoming flow as a constant in Eq. 17. This did not introduce significant error to the pressure drop calculation because the pressure drop is only a weak function of the HX exit temperature calculated using a constant value of \( c_p \) (which, in turn, is only a weak function of temperature). However, with the true \( \dot{Q} \) available, the HX exit specific enthalpy, \( h_{\text{out,HX,cold}} \), is then:
where \( h_{in,cold} \) is the specific enthalpy of the incoming cold duct flow. At this point, the flow exiting the \( n \) HXs is energetically mixed with the flow exiting the \( n \) air gaps. The flow exiting the element has decreased in total pressure and increased in enthalpy by the specified \( \dot{Q} \).

**Fan Duct Heat Exchanger with Air Gap (Calculated \( \dot{Q} \))**

The 3\textsuperscript{rd} stream duct has two HXs. The first facilitates the addition of heat from an external source as described in the Duct with Air Gap and Heat Exchanger (Specified \( \dot{Q} \)) section. For the second HX (the CCA HX), since the conditions are known on both the hot side and the cold side, a more traditional approach to the heat transfer calculations could be adopted. This custom NPSS element uses a hot side effectiveness map that is a function of the mass flow rates on the hot and cold sides of the HX, \( \dot{m}_{HXhot} \) and \( \dot{m}_{HXcold} \), respectively. The map is specified as hot side effectiveness because in this application, the hot side will always have the minimum heat capacity rate, \( C = \min (\dot{m} \cdot c_p) \) (between the hot and cold sides), where \( c_p \) is a relatively weak function of temperature and \( \dot{m}_{HXcold} \gg \dot{m}_{HXhot} \).

As in the Duct with Air Gap and Heat Exchanger (Specified \( \dot{Q} \)) section, \( n \) identical HXs and \( n \) identical cold side air gaps are assumed. All hot flow is assumed to pass through the HX. Again, a guess is made for the fraction of the total cold duct flow that passes through the HX (\( frac_{HX} \)). The flow rates are then:

\[
\dot{m}_{HXcold} = \frac{\dot{m}_{duct} \cdot frac_{HX}}{n} \quad \text{Eq. 21}
\]

\[
\dot{m}_{GAP} = \frac{\dot{m}_{duct} \cdot (1 - frac_{HX})}{n} \quad \text{Eq. 22}
\]

\[
\dot{m}_{HXhot} = \frac{\dot{m}_{HPCh\text{bleedCCA}}}{n} \quad \text{Eq. 23}
\]
where $\dot{m}_{HX\text{cold}}$ is the mass flow rate through the cold side of each identical HX, $\dot{m}_{\text{duct}}$ is the total duct mass flow rate entering the element, $\dot{m}_{GAP}$ is the duct flow rate through each identical air gap, $\dot{m}_{HX\text{hot}}$ is the mass flow rate through the hot side of each identical HX, and $\dot{m}_{HPC\text{bleedCCA}}$ is the total HPC exit bleed flow rate to be cooled in the CCA HX element.

From the definition of heat transfer effectiveness in a HX, the hot side exit total temperature, $T_{out,HX\text{hot}}$, is:

$$T_{out,HX\text{hot}} = T_{in,HX\text{hot}} - \varepsilon \cdot (T_{in,HX\text{hot}} - T_{in,HX\text{cold}})$$  \hspace{1cm} \text{Eq. 24}

where $T_{in,HX\text{hot}}$ is the total temperature of the HPC bleed flow entering the HX hot side, $\varepsilon = f(\dot{m}_{HX\text{cold}}, \dot{m}_{HX\text{hot}})$ is the HX effectiveness (which is mapped as a function of the hot and cold side mass flow rates), and $T_{in,HX\text{cold}}$ is the total temperature of the cold duct flow entering the HX. Making the assumption that $c_p$ is approximately constant on the HX hot side (and using the $c_p$ value based on $T_{in,HX\text{hot}}$), the heat transfer rate in each HX, $\dot{Q}_{HX}$, is:

$$\dot{Q}_{HX} = \dot{m}_{HX\text{hot}} \cdot c_{p,\text{hot}} \cdot (T_{in,HX\text{hot}} - T_{out,HX\text{hot}})$$  \hspace{1cm} \text{Eq. 25}

where positive values of $\dot{Q}_{HX}$ are in the direction of hot side to cold side. The heat transfer rate can then be used to calculate the HX cold side exit total temperature, $T_{out,HX\text{cold}}$:

$$T_{out,HX\text{cold}} = T_{in,HX\text{cold}} + \frac{\dot{Q}_{HX}}{\dot{m}_{HX\text{cold}} \cdot c_{p,\text{cold}}}$$  \hspace{1cm} \text{Eq. 26}

where the similar assumption is made that using a constant $c_p$ value based on $T_{in,HX\text{cold}}$ is reasonable. The remaining calculations for obtaining total pressure losses and balancing the total pressure drop between the HX and air gap on the cold side by iterating on $\frac{\Delta P}{\dot{m}_{HX\text{hot}}}$ are the same as in the Duct with Air Gap and Heat Exchanger (Specified $Q$) section. There is an additional calculation for the pressure drop on the hot side, $\Delta P_{HX\text{hot}}$, that is handled in
the same manner as $\Delta P_{HX\text{cold}}$—requiring a few internal iterations since $\Delta P_{HX\text{hot}}$ itself is a function of the hot side exit pressure (as a part of the hot side average total pressure).

The final calculations done in this element are to set the exit conditions on both sides. The calculated $T_{out,HX\text{cold}}$ and $P_{out,HX\text{cold}}$ are used to set the thermodynamic state at the exit of the HX cold side. For the air gap, the thermodynamic state is set using $T_{in,HX\text{cold}}$ and $P_{out,HX\text{cold}}$. The flow from the cold side exit of the $n$ HXs is mixed energetically with the flow from the $n$ cold side air gaps, resulting in the final thermodynamic state and mass flow rate at the duct side exit of the element. Thus, the total heat transfer rate for the entire element is:

$$\dot{Q} = \dot{m}_{\text{duct}} \cdot (h_{out,\text{duct}} - h_{in,\text{duct}}) \quad \text{Eq. 27}$$

where $h_{out,\text{duct}}$ and $h_{in,\text{duct}}$ are the specific enthalpies at the element's duct side exit and entrance. The thermodynamic state of the hot side exit flow is set using the calculated $P_{out,HX\text{hot}}$ based on the hot side pressure drop and the hot side exit enthalpy, $h_{out,HX\text{hot}}$ which is calculated as:

$$h_{out,HX\text{hot}} = h_{in,\text{HPCbleedCCA}} - \frac{\dot{Q}}{\dot{m}_{\text{HPCbleedCCA}}} \quad \text{Eq. 28}$$

where $h_{in,\text{HPCbleedCCA}}$ is the specific enthalpy of the HPC bleed flow entering the element. The resulting CCA is delivered to the HPT at a flow rate of $\dot{m}_{\text{HPCbleedCCA}}$ (split between non-chargeable and chargeable bleed) at a specific enthalpy of $h_{out,HX\text{hot}}$ and total pressure of $P_{out,HX\text{hot}}$.

**Burner**

The provided Burner element performs calculations to mix the incoming air and fuel and perform burn calculations to increase the mixed flow's enthalpy as a quasi-isobaric (constant pressure) process. Additionally, the burner can account for pressure losses.
(which are enforced before the burn calculation), efficiencies (which are part of the burn calculation), and heat transfer with the casing (which is accounted for after the burn calculation) (Numerical Propulsion System Simulation Consortium 2010, Reference Sheets). The heat transfer between the products of combustion and the casing uses the same approach described in the Heat Soak section. As used in the study engine, fuel flow rate is specified directly (from Simulink) rather than added to the NPSS solver. In the study model, a constant burner efficiency is used and the total pressure drop is given as:

$$\frac{dP}{P} = \left(\frac{dP}{P}\right)_{des} \cdot \frac{\left(1 - \frac{P_{s,in}}{P_{t,in}}\right)}{\left(1 - \frac{P_{s,in}}{P_{t,in}}\right)_{des}}$$

Eq. 29

where \(\left(\frac{dP}{P}\right)_{des}\) is a user-defined normalized pressure loss at the design point, \(P_{s,in}\) is the incoming flow static pressure, \(P_{t,in}\) is the incoming flow total pressure, and the quantity \(\left(1 - \frac{P_{s,in}}{P_{t,in}}\right)_{des}\) is the value of the quantity saved from the design point. Alternate combustor pressure loss approaches are given in references: RTO Technical Report TR-AVT-036 (North Atlantic Treaty Organisation, Research and Technology Organisation 2007) and the "Development of Methods for Analysis and Optimization of Complex Jet Engine Systems" Ph.D. thesis (T. Grönstedt 2000).

**Turbine**

Like a compressor element, a turbine element makes use of maps, interacts with a shaft element, and can also include Heat Soak effects. In contrast to a compressor, a turbine extracts energy from the flow rather than adding energy to it. In removing energy from the flow, the turbine reduces the total temperature and total pressure of the flow as well as generates shaft power (torque) using the energy extracted from the flow.
Turbine maps, like compressor maps, can be layered and require two indices on each layer as illustrated in Fig. 8. The first index is corrected speed, \( N_c \). Because the speed curves tend to have flat sections where the flow is choked, each corrected mass flow value may not correspond to a unique pressure ratio, \( PR \). However, each \( PR \) on a speed curve does correspond to only one corrected flow. Therefore, corrected flow, \( \dot{m}_c \), can be found on a turbine map as follows: 
\[
\dot{m}_c = f(N_c, PR, \alpha)
\]
where \( \alpha \) is the selected vane angle. The adiabatic efficiency can also be mapped as a function of the same three variables (S. M. Jones 2007). As with the compressor, the turbine has both a mass flow rate that is specified from the upstream component and a mass flow rate that is determined from the lookup tables. This requires that the turbine pressure ratio be added to the solver as an independent and that a solver dependent condition be added to match those two mass flow rates at a converged solution (to maintain continuity of flow).

Because turbines operate in an extremely hot environment, thermal barrier coatings and cooling air are both critically important to their safe operation (Jones and Boyle 2009). The cooling air is often HPC exit air that is bled off and plumbed to both the turbine stators and rotor blades as shown in Fig. 9. The cooling air is typically pushed through the hollow rotor blades through small holes on all sides of the airfoils (pressure side, suction, side, leading edge, trailing edge, and blade tip). Additionally, cooling air comes out of holes in the stators, the shroud, and even squeezes through the gaps in the seal between stator and rotor rows. Because cooling flow reenters through both stators and rotors, not all of the cooling air flow is available to do useful work in the turbine. There is more than one way to account for this effect, but a common modeling approach for turbines that are treated as lumped components (or are indeed single stage turbines) is to split the cooling bleed into non-chargeable and chargeable cooling air (North Atlantic Treaty Organisation, Research and Technology Organisation 2007).
Fig. 8 Generic turbine map (S. M. Jones 2007)

Fig. 9 Turbine cooling flow illustration (North Atlantic Treaty Organisation, Research and Technology Organisation 2007)
The non-chargeable cooling air is reintroduced into the flow upstream of the turbine rotor (before the work calculations). This cooling air is mixed energetically at the assumed constant total pressure of the main core flow (i.e. $P_{t041} = P_{t040}$), so:

$$h_{t041} = \frac{\dot{m}_{040} \cdot h_{t040} + \dot{m}_{HPTnonChrg} \cdot h_{HPTnonChrg}}{\dot{m}_{040} + \dot{m}_{HPTnonChrg}}$$  \hspace{1cm} \text{Eq. 30}

where $h_{t041}$ is the specific enthalpy of the fluid at station 041 (between the first HPT stator and rotor as indicated by the “41” in Fig. 9), $\dot{m}_{040}$ is the core mass flow rate coming from the combustor to the first HPT stator at station 040 (combustor exit), $h_{t040}$ is the specific enthalpy of the fluid at station 040, $\dot{m}_{HPTnonChrg}$ is the non-chargeable bleed air flow rate entering through the HPT stator (indicated by the “A” in Fig. 9), and $h_{HPTnonChrg}$ is the specific enthalpy of the non-chargeable cooling flow.

The remaining air, chargeable air (shown as “D” in Fig. 9), is inserted downstream of the turbine (after the turbine work calculations) using an energetic mixing equation similar to that of Eq. 30. The liner cooling air, platform cooling air, and disk rim sealing air (labeled “a”, “c”, and “d”, respectively in Fig. 9) cannot do useful work in the rotor and are lumped together with the chargeable cooling air (“D”). The selection of the fractional split between non-chargeable and chargeable air when modeling the turbine can have a noticeable impact on its performance and should be carefully considered (North Atlantic Treaty Organisation, Research and Technology Organisation 2007).

The LPT is given a similar treatment to the HPT in that it can have both non-chargeable cooling flow added upstream of the rotor and chargeable cooling air added downstream of the rotor. In the NPSS engine model, this is modeled by using bleed elements located upstream and downstream of both turbines as shown in Fig. 5. The
appropriate bleed flows are inserted at those elements to ensure the proper calculation of work from the bleed flow in the turbines themselves.

In the study engine, some HPC exit flow is bled off to be used directly for cooling the turbines. However, some flow is bled from the HPC exit but is first passed through a HX in the third stream before being injected at the HPT (in a combination of non-chargeable and chargeable flows). This air, the CCA, is able to reject heat into the outer bypass duct first, thus enhancing its cooling ability in the HPT. Both the CCA and the conventional cooling air are handled in the same manner.

**Mixer**

The Mixer element mixes two flow streams while conserving energy, momentum, and mass. Mass is conserved simply by ensuring that the flow rate exiting the mixer is equal to the summation of the flow rates entering. Energy and momentum are conserved by iterating on impulse and momentum balance equations until the exit enthalpy and exit total pressure satisfy both of these equations. A mixing efficiency term is used in the mixer model to account for the fact that in a finite length duct mixing is not ideal (complete) (North Atlantic Treaty Organisation, Research and Technology Organisation 2007). Also, the mixer model used in this model accounts for VABI technology by having variable areas for the two incoming flow streams. The total area of the mixer is constant, but the fraction of the inlet area for the core versus bypass is adjustable.

**Afterburner**

The Afterburner is similar to the burner in that its purpose is to add energy to the flow at an ideally constant pressure through the combustion of fuel. As with the primary burner, there are losses that both reduce the total pressure and the completeness of
combustion. The primary difference between the burner and afterburner elements is the additional steady-state control modes offered for the afterburner operation.

**Nozzle**

The Nozzle element is capable of modeling a convergent or a convergent-divergent nozzle. For a supersonic capable aircraft, the convergent-divergent nozzle is required since the convergent only nozzle cannot accelerate the flow to supersonic speeds. The nozzle exit area is calculated such that the flow is fully expanded to ambient pressure. As modeled, a normalized pressure loss term can be added from the nozzle entrance to the nozzle throat. A solver dependent condition is added which seeks to balance the throat area required to pass the flow entering the nozzle with the actual nozzle throat area (Numerical Propulsion System Simulation Consortium 2010, Reference Sheets).

**Shaft**

The Shaft element facilitates the transmission of mechanical power between components. In steady-state operation, the sum of the torque consumed by each compressor and external load (often termed power offtake) on the shaft must be equal to the sum of the torque produced by each turbine on the same shaft (Numerical Propulsion System Simulation Consortium 2010, Reference Sheets). The same relationship is also sometimes equivalently modeled using a balance of power on the shaft instead of torque (SAE Aerospace S-15 Committee 2001). During transient operation, changes in fuel flow, airflow through the turbomachinery, or external shaft loads create a torque (or power) imbalance that results in acceleration or deceleration. This effect is given by the rotational form of Newton's Second Law of Motion (conservation of angular momentum) (North Atlantic Treaty Organisation, Research and Technology Organisation 2007):
\[ \frac{dN}{dt} = \frac{\sum T_{q_i}}{\sum J_i} \]  

Eq. 31

where \( \frac{dN}{dt} \) is the rate of change of shaft speed (acceleration), \( N \) is the shaft speed, \( t \) is time, \( T_{q_i} \) is the torque produced by the \( i \)th component on the shaft (\( \sum T_{q_i} \) being the summation of all torques on the shaft), \( J_i \) is the polar moment of inertia of the \( i \)th component on the shaft (\( \sum J_i \) being the summation of all inertias on the shaft). Also note that there could be components on the shaft which do produce torques but do not have inertial terms (e.g. friction) or components which contribute to the total inertia but do not introduce torque (e.g. the shaft itself).

In the NPSS model, each shaft element introduces both a solver independent and a dependent. The shaft mechanical speed is the independent. Rather than a normal solver dependent, the shaft creates a solver integrator, which is a special type of dependent. During steady-state simulation, the integrator behaves just as a normal dependent which satisfies the condition that the net torque on the shaft is zero (within tolerance) at a converged solution. However, when operating transiently, the state variable \( N \) must equal the integrated value of the state derivative \( \frac{dN}{dt} \), using the chosen numerical integration routine. The state derivative equation (Eq. 31) is integrated as follows:

\[ \int_{t_1}^{t_2} dN = \int_{t_1}^{t_2} \frac{\sum T_{q_i}}{\sum J_i} dt \]

\[ N_{t_2} = N_{t_1} + \int_{t_1}^{t_2} \frac{\sum T_{q_i}}{\sum J_i} dt \]  

Eq. 32
where \( t_2 \) and \( t_1 \) are consecutive points in simulated time. Note that the inertial summation is typically constant with time and could be pulled outside the integral; the torque, however, is a function of time.

The selection of the numerical integration routine can affect both the speed and accuracy of the simulation. Explicit integration routines such as the Euler method only use the value of the state derivative at the previous time step. This approach eliminates the need for iteration since the new value of the state variable is not dependent on the state derivative at the current time step (though the solver may need to iterate for other reasons). Implicit integration routines can provide more accurate solutions at each point in time, but they require iteration on the value of the state derivative at each time step. NPSS supports one explicit integration routine (Euler) and three implicit integration routines (Trapezoidal, 1\(^{\text{st}}\) Order Gear, and 2\(^{\text{nd}}\) Order Gear). The difference between the three implicit integration methods lies in the weighting factors for the current time state derivative versus the state derivative at the previous time step (Numerical Propulsion System Simulation Consortium 2010, User Guide). The numerical integration of Eq. 32 using each of NPSS’s supported integration methods is as follows:

\[
\text{Euler:} \quad N_{t_2} = N_{t_1} + \left( \frac{\sum T_{q_i}}{\Sigma f_i} \right)_{t_1} \cdot (t_2 - t_1) \quad \text{Eq. 33}
\]

\[
\text{Trapezoidal:} \quad N_{t_2} = N_{t_1} + \frac{1}{2} \cdot \left( \frac{\sum T_{q_i}}{\Sigma f_i} \right)_{t_1} + \left( \frac{\sum T_{q_i}}{\Sigma f_i} \right)_{t_2} \cdot (t_2 - t_1) \quad \text{Eq. 34}
\]

\[
\text{1\(^{\text{st}}\) Order Gear:} \quad N_{t_2} = N_{t_1} + \left( \frac{1}{3} \cdot \frac{\sum T_{q_i}}{\Sigma f_i} \right)_{t_1} + \left( \frac{2}{3} \cdot \frac{\sum T_{q_i}}{\Sigma f_i} \right)_{t_2} \cdot (t_2 - t_1) \quad \text{Eq. 35}
\]

\[
\text{2\(^{\text{nd}}\) Order Gear:} \quad N_{t_2} = N_{t_1} + \left( \frac{1}{3} \cdot \frac{\sum T_{q_i}}{\Sigma f_i} \right)_{t_1} + \left( \frac{2}{3} \cdot \frac{\sum T_{q_i}}{\Sigma f_i} \right)_{t_2} \cdot (t_2 - t_1) \quad \text{Eq. 36}
\]

where \( t_2 - t_1 \) is the simulation’s time step, \( \Delta t \).
**Heat Soak**

Heat Soak is not an Element in NPSS, but is instead a Subelement that provides additional calculations for use within an element. Heat soak refers to the phenomenon that a mismatch in temperature between the gas temperature and materials it comes in contact with creates a heat flux. In steady-state, the material and gas temperatures are the same, but transiently one may heat the other. In turbomachinery, for example, the stator, the rotor, and the casing can all transfer heat to or from the gas. A simplifying assumption is made that all of this material can be treated as one lumped mass of consistent material properties. This heat transfer rate is a function of the material properties (thermal conductivity, specific heat, etc.) and the interface area (surface area where the gas and material are in contact). Additionally, the mass flow rate of the gas and the mass of the material determine the rate at which these two temperatures balance out. An example of the heat soak effect is seen in a cold throttle slam in which a fully soaked engine at idle is suddenly accelerated to max power. This can result in a more sluggish transient performance as a significant portion of the energy added to the flow is lost into the engine material rather than producing useful work (Walsh and Fletcher 2004).

The heat transfer rate from the flowing gas (which can be pure air or the mixed products of combustion) into the material is given by:

\[ \dot{Q} = C_{hx} \cdot A_{hx} \cdot (T_{t,gas} - T_{mat}) \]  

Eq. 37

where \( \dot{Q} \) is the heat transfer rate (positive heat transfer assumes that the gas is hotter than the material), \( C_{hx} \) is the heat transfer coefficient, \( A_{hx} \) is the interface area for the heat transfer, \( T_{t,gas} \) is the total temperature of the flowing gas at a location across the element defined by a user-specified weighting factor, and \( T_{mat} \) is the bulk material temperature. The
coefficient $C_{hx}$ is often cast as a function of Reynolds number and Prandtl number or is determined experimentally. The material temperature is then governed by:

$$\frac{dT_{mat}}{dt} = \frac{\dot{Q}}{m_{mat} \cdot c_{p,mat}}$$  \text{Eq. 38}$$

where $\frac{dT_{mat}}{dt}$ is the derivative of the state variable $T_{mat}$, $m_{mat}$ is the mass of the material, $c_{p,mat}$ is the specific heat of the material, and $t$ is time. The gas temperature is also adjusted by removing $\dot{Q}$ from the flow in the element that contains the heat soak subelement (Numerical Propulsion System Simulation Consortium 2010, Reference Sheets).

**Spillage Drag**

Spillage drag, $D_{spill}$, as mentioned in the Overview of Variable Cycle Turbine Engines section, is an additional debit to engine performance. The spillage drag is given by the following equation (Oates 1978):

$$D_{spill} = q_0 \cdot C_{D,spill} \cdot A_{capture}$$  \text{Eq. 39}$$

where the quantity $q_0 = \left( \frac{1}{2} \cdot \rho_0 \cdot V_0^2 \right)$ is the dynamic pressure of the free stream flow, $C_{D,spill}$ is the spillage drag coefficient, $\rho_0$ is the air density in the freestream, $V_0$ is the aircraft velocity (equivalent to the air velocity coming to the engine from an engine-based frame of reference), and $A_{capture}$ is the inlet’s physical capture area. The spillage drag coefficient is, in turn, a strong function of the engine mass flow rate and flight Mach number. For a supersonic inlet, $C_{D,spill}$ is composed of two terms, $C_{D,spill,ref}$ and $C_{D,spill,area}$. $C_{D,spill,ref}$ is a parabolic curve that is a function of Mach number and $C_{D,spill,area}$ is a function of Mach number and the ratio of inlet stream tube area to inlet capture area, $A_0/A_{capture}$ (Simmons 2009). The following equation can be used to calculate the stream tube area (Oates 1978):
where $A_0$ is the inlet stream tube area, $\dot{m}_{c,010}$ is the corrected flow being consumed by the engine, $M_0$ is the aircraft Mach number (or equivalently the incoming flow Mach number from an engine frame of reference), $P_{STD}$ is the standard day pressure at sea level, $R_{\text{air}}$ is the gas constant for air, $T_{STD}$ is the standard temperature at sea level, and $\gamma$ is the ratio of specific heats for air (a weak function of temperature). Note that $A_0$ goes to infinity at $M_0 = 0$. This is not concerning though, as $C_{D,\text{spill\,ref}}$ is zero at $M_0 = 0$ and $C_{D,\text{spill\,area}}$ is zero when $A_0/A_{\text{capture}} = \infty$, creating no spillage drag. In the NPSS model, the spillage drag is calculated at the end of all other calculations and is simply a debit to the installed thrust produced by the inlet. The installed thrust is discussed further in the Controls section.

**Aft Body Drag**

The aft body (exhaust nozzle) drag, $D_{\text{aft\,body}}$, can be treated similarly to spillage drag. It is largely a function of the nozzle’s exit area. The aft body drag is given by the following equation (Oates 1978):

$$D_{\text{aft\,body}} = q_0 \cdot C_{D,\text{aft\,body}} \cdot A_{\text{aft\,body}}$$

Eq. 41

where the quantity $q_0 = \left(\frac{1}{2} \cdot \rho_0 \cdot V_0^2\right)$ is the dynamic pressure of the freestream flow, $C_{D,\text{aft\,body}}$ is the aft body drag coefficient, $\rho_0$ is the air density in the freestream, $V_0$ is the aircraft velocity (equivalent to the air velocity coming to the engine from an engine-based frame of reference), and $A_{\text{aft\,body}}$ is the aircraft aft body area. $C_{D,\text{aft\,body}}$ can be given as a function of Mach number and the ratio of aft body area to nozzle exit area, $A_{\text{aft\,body}}/A_e$ (Simmons 2009). Note that for the study engine, the nozzle exit area, $A_e$ is the sum of the primary (mixed flow) nozzle area, $A_{090}$ and the 3rd stream nozzle area, $A_{290}$: $A_e = A_{090} + A_{290}$.
Flow Holding

As discussed previously, flow holding is the approach of keeping the total engine airflow constant when operating at a reduced thrust point in an attempt to reduce installation drag. This is accomplished by changing variable geometry within the engine to force air into the bypass ducts or to allow it to enter the core. The first key to enabling the engine to swallow the maximum amount of air is to maintain the fan at 100% corrected speed. It was determined by Simmons that core flow can first be reduced by using LPT and HPT variable geometry to choke the flow in the turbines. The flow not accepted by the core must then go to the bypasses. To further reduce core and inner bypass flow, the variable geometry in the LPC can be adjusted; the excess flow must then go into the outer bypass (3rd stream). To enable a more efficient mixing process, the VABI is adjusted to achieve the desired total pressure ratio at the mixing plane. The fan and LPC surge margins are kept within reasonable ranges by adjusting the 3rd stream nozzle and primary nozzle throat areas, respectively (by setting the back pressure on those compressors) (Simmons 2009).

It should be reiterated that there is a point when flow holding no longer becomes a more efficient approach overall as the desired thrust is cut back. This happens when the turbomachinery components are driven too far from an efficient operating point by using the variable geometry. Developing a control algorithm to make this determination is difficult even in steady-state, but is extremely challenging during transient operation. The research by Simmons identified the most important variable geometry features to control for double bypass engines destined for certain mission types by using a genetic algorithm optimization and a complex set of penalty functions on simulation results that produced undesirable performance or violated the laws of physics (Simmons 2009). For normal operation of the engine, this approach is less feasible, so a simpler method was adopted that did not require the external optimizer.
This simpler method involved setting several variable geometry features at fixed values and constraining the others. Specifically, the fan, HPT, and LPT were set to fairly open vane settings and left fixed at those values (the HPC did not have a vane dimension to its map). The LPC vane angle was used as the critical parameter since it governs the amount of flow that is allowed to enter the core and inner bypass versus the third stream. As mentioned, the key to flow holding is maintaining 100% corrected fan speed. Therefore an NPSS solver dependent was added to keep the corrected fan speed at 100%. The corresponding solver independent added was the airflow entering the engine. The corrected fan speed dependent was constrained, however. These constraints allow flow holding to turn off when it is no longer appropriate. Examples of constraints that would trump the desire to maintain 100% corrected fan speed include maximum or minimum LPC vane angles, maximum or minimum flow rates in the 3rd stream, and maximum Mach number in the 3rd stream duct. Note that this approach still does not guarantee the optimal selection of the other variable geometry parameters. By running a matrix of tests with this steady-state solver approach, a flight envelope was developed for a wide range of altitudes, Mach numbers, and throttle settings. The engine’s thrust output, the various variable geometry settings, and the fuel flow rates were saved into tables. These tables are then used for transient runs that use Simulink as the controller for the engine. The full description of this approach is presented in the Controls section.

**Cooled Cooling Air**

At some engine operating points, especially at high speed and altitude flight, the engine is constrained by its maximum temperature. This limit is often enforced as a maximum $T_{r0+1}$ limit (at the entrance to the first HPT rotor) even though the highest temperature is actually at the combustor exit (discounting a possibly higher temperature downstream during augmented operation) (Boyer and Meador 1976). For this reason, the
cooling of the HPT is critical to maximizing overall engine performance. The traditional approach to cooling the HPT is to push HPC discharge air through the HPT stators, rotors, and shroud. This was included in the model, but this cooling flow was also supplemented by CCA to assist at peak operating temperatures.

The CCA is also HPC discharge bleed, but it is first passed through a HX in the 3rd stream. The 3rd stream heat sink temperature is a strong function of flight speed, but is certainly cooler than the HPC discharge bleed at all operating conditions. Because all flow bled from the HPC is a debit to the overall cycle efficiency (even when reintroduced back into the main flow downstream), the CCA is only used when $T_{1041}$ is near or at its maximum. The details of passing the HPC discharge bleed through the 3rd stream HX are discussed in the Fan Duct Heat Exchanger with Air Gap (Calculated $\dot{Q}$) section, but a key point is emphasized here: the heat exchanger's physical design and the mass flow rate of the bleed flow through that HX contribute significantly to the pressure drop seen by that flow. If the pressure drop is too high, then the CCA may need to be compressed slightly before it is at a high enough pressure to be injected at the HPT. This would require an additional compression device operating in a hot environment. The final determination of the need for this increase in CCA pressure is outside the scope of this study since the localized phenomenon associated with boundary layers on rotating blades can dictate a level of complexity in the pressure distribution well beyond what is available from the lumped turbine maps used in this work. Also, it may have been more efficient to have the CCA providing continuous cooling and to use additional HPC discharge bleed as necessary, but making that determination was outside the scope of this work since it would have required extensive HX design analysis.
**Customer Power Extraction**

While a detailed design of the entire aircraft with all its subsystems is a huge task, some integrated studies can be performed by making some simplifications. For the studies presented in this research, a simplifying assumption was made that the details of the electrical system could be neglected since its dynamics tend to happen at much higher frequencies than those of the engine. A second simplifying assumption was that a detailed six-degree-of-freedom model of the aircraft was not necessary. Since the missions used in these studies are arbitrary and were created only to exercise/stress certain aspects of engine operation, a simple drag polar model was sufficient to demand a thrust from the engine as a function of mission conditions. The observation that thermal management system (TMS) design tends to be specific to a particular aircraft and mission led to a third simplifying assumption that TMS trends coupled with engine-relevant transient phenomenon would be sufficient. Again because the overall model is only generic and does not represent any fielded or planned design, using a simplified model rather than a detailed model still enables drawing useful conclusions.

These assumptions led to the development of a generic system of loads on the engine system. The overall architecture is assumed to be more electric, increasing the electric power demands and reducing the bleed demands to meet aircraft needs. Because the aircraft is designed for two engines, the needs of the aircraft are split between the two. While a true no bleed aircraft certainly may be possible, relatively small low pressure and high pressure bleeds for aircraft use were allowed in this design. The low pressure pneumatic power provided from the fan exit does vary in pressure throughout the mission, but the flow rate was set at a constant 1.0 lbm/s (0.5 lbm/s from each engine). Additionally, higher pressure pneumatic needs were met using HPC exit flow set at a constant 1.0 lbm/s (again the absolute pressure varies throughout mission but the flow rate required from
each engine is 0.5 lbm/s. No LPC bleed is extracted beyond that needed for inlet flow control. Though the element is in place to facilitate its use, the HX in the inner bypass duct is effectively eliminated by adding/removing no heat and having no pressure losses. Instead, all heat rejection is to the 3rd stream (or to the fuel). The role of fuel as a heat sink is not modeled, though it could be considered in detail in future studies. While some aircraft- or engine-generated heat may be rejected to the fuel, this is not explicitly modeled and the assumption is that 200 Btu/s (100 Btu/s per engine) is the base heat load rejected to the 3rd stream at all times. Similarly, there is an assumption of a total aircraft base shaft power load of 2,000 hp (approximately 1.5 MW). This load is split evenly between the HP and LP spools of the two engines such that each engine has a base load of 500 hp from the HP spool and 500 hp from the LP spool.

The key element in the studies presented is the loading above and beyond these base values. No pneumatic power is needed for this additional load. It is assumed to be an electrical load that is powered by shaft driven generators. The electrical load itself is assumed to be a low efficiency periodic load which creates a large amount of waste heat that must be rejected to the 3rd stream. Rather than develop a detailed model of the electrical load and the thermal management system that cools such a load (which could contain sensitive information), a generic correlation is used for the relationship between the 3rd stream sink temperature and the coefficient of performance (COP) for an air refrigeration cycle to lift the low quality waste heat to a sufficiently high temperature for rejection into the 3rd stream.

The large periodic load is defined by: 1) the amplitude of the electrical power required; 2) the electrical efficiency of the load (the inefficiency is assumed to be waste heat that must be rejected to the 3rd stream); 3) the efficiency of the generator that converts
shaft mechanical power into the required electrical power; 4) the period of the large load; 5) the duty cycle of the large load (portion of the period which the load is applied); 6) a maximum rise or fall rate for the load-on and load-off transients (large instantaneous shaft load changes are both numerically problematic and unrealistic); and 7) the mission times or mission segments during which the periodic load is in use (only the base loading applies when the periodic load is not in use). The diagram in Fig. 10 illustrates the relationship between the generic load itself, the 3rd stream heat sink, and the engine shaft. The electrical power needed by the load, $\dot{W}_{\text{load,elec}}$, is pulled from an LP spool shaft mounted generator with assumed constant efficiency, $\eta_{\text{gen}}$. The electrical efficiency of the periodic load, $\eta_{\text{load}}$, results in both the desired output and in waste heat generation:

$$\text{Desired output} = \eta_{\text{load}} \cdot \dot{W}_{\text{load,elec}} \quad \text{Eq. 42}$$

$$\text{Waste heat} = \dot{Q}_{\text{load}} = (1 - \eta_{\text{load}}) \cdot \dot{W}_{\text{load,elec}} \quad \text{Eq. 43}$$

where $\dot{Q}_{\text{load}}$ is the waste heat from operation of the periodic load which needs to be lifted by the cooling system and then rejected into the engine's 3rd stream HX.

Looking at the energy balance in the generically represented cooling system as a control volume (Fig. 10) yields:

$$\dot{Q}_{\text{load}} + \dot{W}_{\text{driving,elec}} = \dot{Q}_{\text{reject}} + \dot{W}_{\text{loss}} \quad \text{Eq. 44}$$

where $\dot{W}_{\text{driving,elec}}$ is the electrical power required to run the cooling system, $\dot{Q}_{\text{reject}}$ is the heat load to be rejected into the 3rd stream, and $\dot{W}_{\text{loss}}$ is a generic term to include losses in the cooling system that consume electrical power but are not sunk to the 3rd stream. Examples of contributors to the lumped $\dot{W}_{\text{loss}}$ term include losses in the cooling system's compressor and bearing losses.
Assuming that these losses can be neglected (and thus all the electrical power put into the cooling system is producing lift on the waste heat), Eq. 44 simplifies to the following:

\[ Q_{\text{reject}} = Q_{\text{load}} + W_{\text{driving,elec}} \quad \text{Eq. 45} \]

The coefficient of performance (COP) is (Çengel and Boles 2002):

\[ COP \equiv \frac{\text{Required Input}}{\text{Desired Output}} = \frac{\text{Heat Extraction Rate from Load}}{\text{Power Input to Cooling System}} \quad \text{Eq. 46} \]

\[ COP = \frac{Q_{\text{load}}}{W_{\text{driving,elec}}} \quad \text{Eq. 47} \]

\[ W_{\text{driving,elec}} = \frac{Q_{\text{load}}}{COP} \quad \text{Eq. 48} \]

Substituting Eq. 43 into Eq. 48 gives an expression for the cooling system input power as a function of the periodic load:
The engine shaft power required by the generator to electrically power the periodic load and the cooling system is then:

$$ W_{gen, mech} = \frac{\dot{W}_{load, elec} + W_{driving, elec}}{\eta_{gen}} \quad \text{Eq. 50} $$

$$ W_{gen, mech} = \frac{\dot{W}_{load, elec} + (1 - \eta_{load}) \cdot \dot{W}_{load, elec}}{\eta_{gen}} \quad \text{Eq. 51} $$

$$ \dot{W}_{gen, mech} = \left( \frac{1}{\eta_{gen}} \right) \cdot \left( 1 + \frac{1}{\text{COP}} \right) \cdot \dot{W}_{load, elec} \quad \text{Eq. 52} $$

where $\dot{W}_{gen, mech}$ is the engine shaft power required by the generator.

On the output side of the cooling system, the heat rejection rate into the engine's 3\textsuperscript{rd} stream HX is found by substituting Eq. 43 and Eq. 49 into Eq. 45:

$$ \dot{Q}_{reject} = (1 - \eta_{load}) \cdot \dot{W}_{load, elec} + \frac{(1 - \eta_{load}) \cdot \dot{W}_{load, elec}}{\text{COP}} \quad \text{Eq. 53} $$

$$ \dot{Q}_{reject} = (1 - \eta_{load}) \cdot \left( 1 + \frac{1}{\text{COP}} \right) \cdot \dot{W}_{load, elec} \quad \text{Eq. 54} $$

In summary, a large-amplitude, periodic load that is turned on for certain mission legs is defined by the load electrical amplitude, electrical efficiency, period, duty cycle, and rise/fall rate. By additionally defining the LP generator efficiency and a curve fit estimate for COP as a function of total temperature in the 3\textsuperscript{rd} stream duct, the total shaft power required from the engine's LP spool is given by Eq. 52 and the total heat rejection rate into the 3\textsuperscript{rd} stream is given by Eq. 54. The Simulink implementation of these two equations and the subsystem mask to provide the required input parameters is in Fig. 11.
Physics Not Modeled

Many simplifying assumptions were used in developing the model described above. This section briefly describes some of the effects that were not modeled or were modeled in a simplified form. To increase a model’s detail, these additional effects can be added. There is a tradeoff, however, between the increase in accuracy afforded by the additional detail and the corresponding increase in computational time. Another consideration is simply how important an effect is compared to the uncertainty involved in implementing it. Obtaining the additional data required can be difficult for some effects.
Transient tip clearance within turbomachinery is an effect related to heat soak discussed above in that it has to do with heat transfer within the material. It specifically refers to the thermal expansion of the rotor blades versus the casing around the blades. That gap changes as the engine switches its operating point transiently because the thermal time constants for the blades and casing are different. Tip clearances can also be influenced by rotational forces. For most lumped element (0-D) models, this effect is modeled empirically as modifiers to the turbomachinery maps; for stage-by-stage models, it can be modeled in more of a physics-based manner (North Atlantic Treaty Organisation, Research and Technology Organisation 2002).

There is a level of modeling detail for the combustor beyond adding a transient response to the fuel delivery and assuming a perfect, instant, and complete combustion subject to efficiency and pressure loss terms. Improving the model to reflect the real combustion process could include adding the following detail: acoustic effects, hot spots and regions of more- and less-complete combustion, primary and secondary zones which are in different thermodynamic states, uneven pressure distribution, chemical kinetics, combustion stability/flameholding concerns, backmixing and recirculation, liner cooling, and non-ideal fuel chemistry (Mattingly, Heiser and Pratt 2002). Many of these additional effects are only appropriately modeled using higher dimensional fidelity.

There is sometimes a need to model shaft dynamics beyond the rate of change in speed. An example of this is for shaft break analysis where the transient torques at various locations on the shaft contribute to a shaft wind-up or twisting effect. Modeling shaft windup would require shaft material properties, geometrical information (at a minimum) to even represent the shaft as a simple torsional spring and damper system. This effect would not likely be important for aircraft-level performance characteristics, but would come into
play when looking at the interaction between the engine and the power generation system (e.g. a large transient torque imposed on the shaft by the generator will certainly twist the shaft and create angular jerk). This is often more of a failure mode consideration than a mission performance consideration.

Volume dynamics (also termed volume packing or mass storage) refers to the time propagation of the flow through the engine. Without including volume dynamics, the assumption is effectively that all effects in the engine happen at the same time and that the mass ingested by the engine is immediately ejected. Modeling the volume dynamics can be done in more than one way, each with pros and cons. The simpler approach is to have an occasional control volume (plenum) element after a physically large engine component. This volume makes corrections to the flow characteristics (often pressure). The problem with this approach is that is not a true implementation of flow continuity through the engine. However, it can avoid a requirement for tiny simulation time steps because it is only capturing the larger time constant (slower) volume dynamics (Martin 2009). The other approach is to model connected control volumes from end to end of the engine. This benefits from being a more accurate representation of the physical system, but can come at a high computational cost. Both approaches require a considerable amount of engine geometry data. For a three stream engine where modulating air flow into different paths is a part of engine operation, this effect might be even more important than it would be in a simpler turbofan. It has not yet been implemented for the study engine because the model merely generically represents a cycle concept rather than a particular design.

There are many engine elements which are often given a more detailed treatment, whether within a complete engine performance model or only as an isolated component for design. The compressors and turbines are prime examples. These elements are often
modeled as lumped, zero-dimensional elements using maps for performance models. However, the true detailed design obviously involves an incredible level of detail (number of stages, number of blades, airfoil shapes, blade twist, etc.). If the required data is available, the compressors and turbines can be given a one-dimensional effect by performing stage-by-stage calculations with separate heat transfer to/from the stators, rotors, and casing. The inlet and nozzle are also elements that are often given a more thorough treatment for design. This is due to their tight integration with the airframe and the importance of capturing detailed flow effects such as shock waves (internal or external). Some inlets have serpentine ducts which can suffer from significant flow separation. Additionally, the flight trajectory itself and the engine/airframe interface can produce a highly non-uniform flow that can lead to distortion at the fan face. These non-ideal effects in the inlet can have significant impact on engine operation (Longley and Greitzer 1992).

There are also other effects that generally play a more minor role. Some of these are transient-specific and some are just secondary effects that affect both steady-state and transient operation. An incomplete list of effects not mentioned previously that could be important to capture in a model (depending on its purpose) includes: humidity (water-to-air ratio) effects; Reynolds number effects; engine deterioration (e.g. surface roughness changes due to sand ingestion or hours of use); a statistical distribution of gas properties within each component; imperfect mixing (especially important during afterburner operation where there will be different combustion regions); flow radial non-uniformity; blade untwist due to rotational forces in high-speed turbomachinery; localized stall of airfoils; blade flutter; interactions between flow turbulence and chemistry in burners; flame blow-out and relight; sand/rain/hail ingestion effects on combustion; engine start, in-flight restart, and sub-idle operation; thrust vectoring backpressure effects; engine actuation effects; sensor performance; and flow leakage effects. Each of these effects presents a
deviation from the assumptions used to develop a simple model. The importance of each of these effects and the ease with which they can be modeled varies greatly from one engine to another.

**Controls**

The overarching goal of the engine controller is to set engine parameters appropriately to achieve a desired performance. Typically this desired performance is to match the thrust required by the aircraft. In the case of the study engine, the primary combustor fuel flow rate and the afterburner fuel flow rate are the main tuning parameters to achieve the required thrust. The variable geometry within the engine also has an impact on the engine’s ability to achieve the requested thrust, though its central role is achieving that thrust in a more efficient manner (e.g. by flow holding when appropriate) while keeping appropriate safety/stability margins (e.g. surge margins).

The control approach developed for this initial work was simple, but the intention was to develop it in such a way that a more elaborate controller could be used in the future. The variable geometry is scheduled in an open loop manner based on steady-state values (discussed further in the Variable Geometry Control section) and the fuel flow for the primary combustor and for the afterburner is controlled with proportional-integral-derivative (PID) controllers that include feed-forward terms (discussed further in the Fuel Control section). The engine strives to meet the installed thrust requested by the engine. For this reason, having a consistent definition of the installed thrust between the engine and aircraft is important.

From the engine’s perspective, thrust is produced at nozzles and drag is created at inlets. The difference between these two terms is the uninstalled net thrust, $F_{net}$
(Numerical Propulsion System Simulation Consortium 2010, Reference Sheets). For the study engine which has two nozzles:

\[ F_{net} = \sum F_{gi} - F_{ram} \quad \text{Eq. 55} \]

\[ F_{net} = (\dot{m}_{090} \cdot V_{090} + \dot{m}_{290} \cdot V_{290} - \dot{m}_{010} \cdot V_0) \]

\[ + \left( P_{s,090} - P_{s,0} \right) \cdot A_{090} + \left( P_{s,290} - P_{s,0} \right) \cdot A_{290} \quad \text{Eq. 56} \]

where \( F_{gi} \) is the gross thrust produced by the \( i^{th} \) nozzle (and \( \sum F_{gi} \) is the summation of nozzle gross thrusts), \( F_{ram} \) is the ram drag cause by the inlet, \( \dot{m}_{090} \) and \( \dot{m}_{290} \) are the mass flow rates exiting the primary and 3\(^{rd}\) stream nozzles, \( V_{090} \) and \( V_{290} \) are the primary and 3\(^{rd}\) stream nozzle exit velocities, \( \dot{m}_{010} \) is the mass flow rate entering the engine, \( V_0 \) is the flight speed (air velocity entering the engine), \( P_{s,090} \) and \( P_{s,290} \) are the static pressures at the primary and 3\(^{rd}\) stream nozzle exits, \( A_{090} \) and \( A_{290} \) are the exit areas of the primary and 3\(^{rd}\) stream nozzles, and \( P_{s,0} \) is the free stream static pressure. For the study engine which assumes that the nozzles fully expand to atmospheric pressure, Eq. 56 simplifies to:

\[ F_{net} = (\dot{m}_{090} \cdot V_{090} + \dot{m}_{290} \cdot V_{290} - \dot{m}_{010} \cdot V_0) \quad \text{Eq. 57} \]

Additional installation drags that are normally tabulated with the engine are the spillage drag and aft body drag since both of these drags are functions of engine mass flow rates and engine areas that may change (though also a function of flight Mach number). While there are certainly other drags associated with the aircraft system, those drags are not usually as sensitive to engine operation; they are more a function of the airframe geometry, flight control surface position, external stores, and flight trajectory. Those drag terms are generally accounted for at the aircraft-level such that it requests of the engine enough thrust to overcome all of those drags. The spillage drag and aft body drag contribute to engine installation losses. Those terms create the difference between installed and uninstall net thrust:
\[ F_{net, inst} = F_{net} - D_{spill} - D_{aftBody} \]  

where \( F_{net, inst} \) is the installed net thrust of the engine, \( D_{spill} \) is the spillage drag (discussed in the Spillage Drag section), and \( D_{aftBody} \) is the aft body drag (discussed in the Aft Body Drag section). It is \( F_{net, inst} \) which the engine controller seeks to provide.

**Overall Controller Structure**

The main structure for the engine’s controller is given in Fig. 12. The controller is broken down into six major portions:

1. Pre-lookup current altitude and Mach number. The engine controller makes heavy use of lookup tables that have altitude and Mach number as indices for two of their dimensions. It is more computationally efficient to compute the floor index integer and fraction for both the altitude and Mach number only once and pass those indices directly to the tables that require them.

2. Determine whether the afterburner should be used. There are three modes for the afterburner: ON, OFF, and AUTO. As suggested by the names, ON forces the afterburner to be in operation regardless of whether it is actually required, OFF completely disables the afterburner even if the engine cannot generate the demanded thrust without it, and AUTO attempts to use the afterburner only when dry operation cannot produce sufficient thrust. All of the missions studied used the AUTO mode of operation. The logic used to govern the AUTO mode is simple, but imperfect. In AUTO mode, the thrust demanded by the aircraft is compared with a table lookup of the estimated maximum dry thrust (mil power) that can be produced by the engine in steady-state at the current altitude and Mach number (index integers and fractions for tables are provided by step 1). This approach requires that an accurate table be generated that has mil power thrust accurately captured for the full flight envelope. If the table is highly inaccurate or
if the engine’s true operating condition is far from the conditions used to generate the
table (e.g. if there is a severe mismatch between shaft or bleed power extraction at
current conditions versus conditions during table generation), then it may not trigger
the usage of the afterburner even though the required thrust cannot be achieved
without it.

3. Determine required fraction of available mil or max thrust. Based on the current
altitude and Mach number, the controller needs to know approximately what portion of
engine power is required. This is effectively determining the engine throttle stick
setting. For the current altitude and Mach number (indices and fractions provided by
step 1 above), a pair of table lookups at partial engine power (in 5% increments from 5
to 100%) and partial augmented operation (0, 50, and 100% afterburner operation)
results in a curve of estimated thrust capability as a function of throttle setting. An
inverse lookup is performed to find the throttle setting floor index integer and fraction
that defines where the demanded thrust falls on this estimated thrust curve. This
throttle index is then used in step 4 below. Also in this portion of the controller, the
estimates for mil power thrust available, max power thrust available, afterburner usage
flag (determined in step 2 above), and thrust demanded by the aircraft are grouped
together for data logging.

4. Determine fuel flows and variable geometry settings. This section of the controller uses
the altitude and Mach number indices and fractions from step 1 and the throttle setting
index and fraction from step 3 in determining the fuel flow and variable geometry
settings to command to the engine. The details of this portion of the controller are given
in the Fuel Control and Variable Geometry Control sections below. Control over bleed
flow rates is also handled in this portion of the Simulink engine controller.
Fig. 12: Simulink controller block diagram for NPSS engine.
The customer bleeds from the fan, LPC, and HPC are all merely checked against minimum bleed flow rates and maximum fractional flow rates before being passed through. The CCA bleed is slightly more complex. As discussed in the Cooled Cooling Air section, the CCA is not needed when $T_{t,041}$ is below a user specified threshold. Below that temperature, the CCA flow rate is set to its minimum. Similarly, there is a $T_{t,041}$ temperature above which the maximum CCA flow rate is used. Between those two temperatures, the amount of CCA flow demanded is linear.

5. Transport delays and actuation dynamics. Though step 4 of the controller computes fuel flow and variable geometry commands for the engine, those commanded changes are not realized instantaneously inside the engine. There are both transport delays and actuation dynamics associated with each of these controlled variables (Jaw and Mattingly 2009). A first order approximation is made for these effects.

6. Check other limits. This portion of the controller merely monitors other engine variables of interest that are not directly part of the control logic. Many of the items monitored are related to verifying that the model is running in a physically viable manner (e.g. negative absolute temperatures are a clear violation of physics and would flag a problem).

**Fuel Control**

The primary combustor fuel flow rate is governed by a PID controller with a feed-forward term, subject to other limits as illustrated in Fig. 13. The feed-forward term is simply an approximation of the fuel flow required. This approximation is from a lookup table that is a function of altitude, Mach number, and the throttle setting (indices and fraction terms from steps 1 and 3 in the Overall Controller Structure section). The PID controller compensates for the feed-forward term being an inexact value to produce the demanded thrust. A normalized thrust error term is calculated:
\[ F_{\text{error}} = \frac{F_{DMD} - F_{\text{net,inst}}}{F_{\text{net,inst}}} \]  

where \( F_{\text{error}} \) is the normalized thrust error term, \( F_{DMD} \) is the thrust demanded by the aircraft, and \( F_{\text{net,inst}} \) is the installed net thrust produced by the engine at the previous time step.

This error term is modified by correction factors. These correction factors can offset or even overpower the thrust deficiency when other engine limits are encountered. Currently, the only correction factor included is a maximum \( T_{t,041} \) limit. When \( T_{t,041} \) is detected to be over its limit, an error term starts to grow (the amount above \( T_{t,041,max} \) and time above \( T_{t,041,max} \) contribute to the size of this error). The \( T_{t,041} \) limit correction factor is only enforced in one direction—the \( T_{t,041} \) limit does not try to push the engine toward \( T_{t,041,max} \), only away from it. After accounting for limits-based correction factors, the thrust error is passed to a PID controller.

This is a standard PID control approach with anti-windup to avoid the problem of the error term continuing to grow when the engine is against a limit such as a minimum fuel flow. The importance of the anti-windup logic is that it allows the controller to behave normally once the engine comes back from the limit; without this logic, the integral term could take a long time to reduce its contribution to the error term (Åström and Murray 2008). The PID portion of the primary combustor fuel control is shown in Fig. 14. The PID control contribution and feed-forward contribution are summed and result in a new fuel flow command. This command is checked against maximum and minimum fuel flow rates before being passed to the actuation dynamics portion of the controller. The afterburner fuel flow control is set up similarly. Because it is downstream of the turbine, it is not constrained by a \( T_{t,041} \) limit. It could encounter a maximum nozzle temperature that would be handled in the same way. However, this is not currently implemented.
Fig. 13 Primary combustor fuel control block diagram
The unique aspect of the afterburner fuel control is that its operation is subject to the value of the flag that determines its usage (determined in step 2 of the Overall Controller Structure section).

**Variable Geometry Control**

As discussed in the Overall Controller Structure section, the variable geometry is set by open-loop commands based on table data that is a function of altitude, Mach number, and throttle setting. The fan, HPT, and LPT have a variable geometry dimension in their maps, but these are held at constant values to simplify the control approach (the HPC map used does not have this additional dimension). The LPC vane angle, primary nozzle throat area, 3rd stream nozzle throat area, and mixer core flow entrance area are all commanded from table lookups (the mixer bypass flow entrance area is inherent since the total area is fixed and the core flow area is commanded). Rate limits, absolute minimum and maximum values, and actuation dynamics are all applied to these commands. For afterburning operation, all variable geometry settings except the primary nozzle throat area are held at

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**Fig. 14** PID control with anti-windup
their mil power values. The primary nozzle throat area command has two tables: one for
dry operation and one for augmented operation. The selection between the two is made by
the afterburner operation flag determined in step 2 of the Overall Controller Structure.
III. RESULTS

The results of the studies presented are organized into four sections. The first section attempts to assess the quality of the model through some basic verification steps. The remaining three sections address simulation results obtained from running the model over each of the three missions that were created to exercise or stress the engine.

Basic Model Verification

The first step in building confidence in the model was to ensure that the basic laws of physics were followed. Checking for the conservation of mass in the engine was the simplest such check. The mass balance for the engine is as follows:

\[ m_{\text{gained}} = \sum \dot{m}_{\text{in}} - \sum \dot{m}_{\text{out}} \]  
\[ m_{\text{gained}} = (\dot{m}_{010} + \dot{m}_{\text{fuel,pri}} + \dot{m}_{\text{fuel,aug}}) \]
\[ -(\dot{m}_{090} + \dot{m}_{290} + \dot{m}_{\text{bldFan}} + \dot{m}_{\text{bldLPC}} + \dot{m}_{\text{bldHPC}}) \]

where \( m_{\text{gained}} \) is the mass flow rate that is numerically gained (should be identically zero to conserve mass), the summation terms \( \sum \dot{m}_{\text{in}} \) and \( \sum \dot{m}_{\text{out}} \) refer to the total incoming flows and total outgoing flows, \( \dot{m}_{010} \) is the airflow entering the engine, \( \dot{m}_{\text{fuel,pri}} \) and \( \dot{m}_{\text{fuel,aug}} \) are the fuel flow rates in the primary combustor and afterburner, \( \dot{m}_{090} \) and \( \dot{m}_{290} \) are the flow rates exiting the primary and 3rd stream nozzles, and the customer bleed extraction from the fan, LPC, and HPC are \( \dot{m}_{\text{bldFan}} \), \( \dot{m}_{\text{bldLPC}} \), and \( \dot{m}_{\text{bldHPC}} \). Internal engine bleeds that reenter the main flow do not need to be accounted for since they do not leave the control volume of the engine. While the numerical precision of computers makes it very difficult to
have a perfect mass balance, a properly configured system and reasonable solver tolerances should yield a reasonably close value. The mass balance over one mission is given in Fig. 15. Note that in this figure, the maximum absolute value of the deviation from true mass conservation is the very small value of $3.4106 \times 10^{-13} \text{lbm/s}$. Even the integration of the absolute value of all mass flow lost or gained over the course of a mission is only on the order of $1 \times 10^{-10} \text{lbm/s}$. This provides confidence that the model’s solver tolerances are reasonable on mass conservation type balances.

The next check verified both physical laws and model reasonableness. The 3rd stream BPR (ratio of outer bypass flow to flow entering the LPC) and the 2nd stream BPR (ratio of inner bypass flow to flow entering the HPC) are shown in Fig. 16. A negative BPR would indicate negative (reversed) flow in the bypass ducts.

Fig. 15 Checking for conservation of mass
An overall BPR greater than 3 or 4 could indicate an engine design problem since this engine is unlikely to be a high BPR design. As seen in the figure, the BPR range for both engine split points falls in the range between 0.04 and 0.82, with the overall BPR not exceeding 1.55.

Similar to assessing the BPR is assessing the Mach number in the bypass ducts. There is a desire to keep all flows upstream of the nozzles subsonic. Therefore checking the bypass duct Mach numbers and mixer Mach numbers provides another measure of confidence in the reasonableness of the model design. These Mach numbers are shown in Fig. 17. Because the duct Mach numbers do not closely approach 1, the ducts may have been slightly oversized. However, this is preferred over undersized ducts.
Note also that negative Mach Numbers would again indicate flow moving in the opposite direction. Seeing only non-negative values is important.

To verify both a reasonable mission design (i.e. a mission that does not try to fly to envelope points that are unattainable) and a reasonable controller design (i.e. one that qualitatively does a reasonable job of tracking the demanded thrust), the thrust demanded by the aircraft and the installed thrust produced by the engine are plotted together over the mission. Such a plot for one mission is given in Fig. 18a. An unreasonable mission design would have a section where there was a sustained separation between the demanded thrust and the achieved thrust. Since the two lines are nearly indistinguishable, this suggests a reasonable controller design and a feasible mission.

Fig. 17 Checking bypass duct and mixer Mach numbers to verify subsonic flow
a) Demanded thrust versus actual installed thrust produced by the engine

b) The ratio of actual installed thrust to demanded thrust

Fig. 18 Checking the engine’s ability to track thrust demand
The same data can also be displayed differently as in Fig. 18b, which plots the ratio of actual produced engine thrust to demanded engine thrust. Sustained values significantly different than unity would indicate a problem. Though there are transient spikes (upward or downward), there are no large sustained mismatches that remain in steady-state. This figure also highlights the fact that while the engine (with its controller) cannot instantaneously produce the thrust demanded by the aircraft, it does quickly settle to the desired value. Also, it is seen that in steady flight such as cruise, the controller very accurately achieves the desired thrust.

Because air in the 3rd stream only experiences compression by the fan, the pressure entering that duct is not extremely high compared to ambient. With blockages such as the 3rd stream HX that cools the external heat load and the CCA HX that cools an internal load, the flow can experience a significant pressure drop. Even the unobstructed duct itself creates a small pressure drop. The sum of these pressure drops cannot exceed the pressure increase provided by the fan. Stated another way, the total pressure entering the nozzle must be greater than the ambient static pressure throughout the mission. Fig. 19 shows the ratio of the 3rd stream nozzle entrance total pressure to the ambient static pressure. It can be seen that this ratio is greater than 1 for the entire mission, as required. However, it can be observed that the margin is slim at ground idle conditions where the nozzle entrance total pressure is only about 3% higher than ambient. This is because the fan pressure ratio itself is less than 1.2 and the mass flow rate in the 3rd stream is relatively high at this low engine power condition.

Additional checks performed for each model run include ensuring that all compressors maintain a positive surge margin, ensuring that LP and HP shaft physical speeds do not exceed 110% of the design speed (a 10% overspeed is allowable, but above
that is a concern), and very basic items such as making sure that all absolute temperatures and pressures are positive.

Fig. 19 Checking for a positive pressure ratio in the 3rd stream nozzle

The last check done was an evaluation of the simulation time step. For model results to be believable, the time step must be small enough to capture the transient response accurately. However, shrinking the time step comes at a computational cost since it takes longer to run each simulation when the time step is smaller. Therefore, it is desirable to find a time step that is as large as possible while still producing accurate results. This determination is made by starting with a large time step and running the simulation repeatedly with smaller and smaller time steps until the results are identical within a reasonable tolerance. The challenge, however, is in determining the variables of importance for this comparison. Since many calculations in the NPSS model are the same
for steady-state and transient operation, these computations should be time-independent. Therefore, the focus is on those variables which are integrated, namely the material temperatures in heat soak components and the shaft speeds. A time step comparison is given in Fig. 20. Note that for both the full mission view (Fig. 20a) and the single pulse view (Fig. 20b), the difference in results for each time step size is not easily distinguished. Fig. 21 shows small-scale responses. The figure reveals differences in the response on smaller time scales. Fig. 21a highlights a relatively small, yet noticeable difference in peak LP spool speed during a transient and Fig. 21b shows that the HP spool is also affected by the choice of time step size (note that a different transient event is shown in this plot). Both portions of the figure suggest that the 10 ms time step is the clear outlier.

Two additional time steps were also attempted but did not produce a numerically stable simulation. Table 2 summarizes the time steps attempted.

<table>
<thead>
<tr>
<th>Fixed Time Step, [s]</th>
<th>Qualitative Model Behavior</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.050</td>
<td>Model failed to run</td>
</tr>
<tr>
<td>0.025</td>
<td>Model failed to run</td>
</tr>
<tr>
<td>0.010</td>
<td>Model ran successfully - very slight data inaccuracies</td>
</tr>
<tr>
<td>0.005</td>
<td>Model ran successfully - used as time step for missions</td>
</tr>
<tr>
<td>0.001</td>
<td>Model ran successfully - slow runtime and large data files</td>
</tr>
<tr>
<td>0.0005</td>
<td>Model ran successfully - very slow runtime and very large data files (~30 GB of data for a ~2 hour mission)</td>
</tr>
</tbody>
</table>

The model failed to run with the 50 ms or 25 ms time step. With the 10 ms time step, the model successfully ran to completion, but did not seem to capture all of the transient behavior as exemplified in Fig. 21b. Reducing the time step to 5 ms allowed the ripple to be captured reasonably well though still not in perfect agreement with the higher temporal resolution result.
Fig. 20 Checking the large-scale impact of the fixed time step size

a) LP spool speed over the entire mission

b) LP spool speed during a single pulse
a) LP spool speed over a 0.30 second period

b) HP spool speed over a 0.30 second period

Fig. 21 Checking the small-scale impact of the fixed time step size
The difference between the 1 ms time step and the 0.5 ms time step was minimal in the data accuracy, but the factor of 2 difference in time step resulted in a factor of 2 difference in the resulting data generated from the model. The difficulty in processing 30 GB data files alone eliminated that time step from serious consideration. Based on data accuracy, there was a preference toward the 1 ms time step, but the reduction in simulation time and data file size by approximately a factor of 5 favored the 5 ms time step enough that it was chosen for use in the various mission studies. It should be noted, however, that the data file size generated is proportional to the number of signals logged; reducing the number of logged signals while keeping the time step the same would also result in smaller data files.

**Ground Attack Mission**

The first mission used to stress the engine operation is a generic ground attack mission as given in Fig. 22. The basic legs of the mission, as indicated in the figure, are:

A. Ground idle  
B. Takeoff/climb  
C. Subsonic cruise  
D. Climb to loiter altitude  
E. Loiter*  
F. Dive to penetration conditions*  
G. Low altitude penetration/retreat*  
H. Climb*  
I. Subsonic cruise  
J. Descend/land  
K. Ground idle

* = periodic load is active

Because the studies did not consider fuel tank temperatures or other conditions which would have benefitted from simulating soak conditions at ground idle, those mission segments were shorter than they might be for a real mission. Similarly, the distance traveled or time elapsed for each mission leg is not necessarily representative of a real flight mission.
Furthermore, while fuel usage was tracked to update the aircraft mass, no attempt was made to ensure there was sufficient fuel to complete the mission, required reserve fuel, etc. The only check on reasonableness of the fuel mass usage was to note that the mission used significantly less fuel than the total aircraft gross takeoff mass.

**Overall Mission Performance**

For this mission, the periodic load is active during the Loiter*, Dive to penetration conditions*, Low altitude penetration/retreat*, and Climb* mission segments, resulting in the external periodic load's electrical pulse profile (per engine) shown in Fig. 23.

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Fig. 22  Ground attack mission Mach number and altitude profiles
The base heat and shaft power extraction loads apply throughout the mission, but for mission legs not marked with an *, only the base load values apply, as discussed in the Customer Power Extraction section. Looking at the whole mission, the flow distribution through the engine varies significantly. The portion of the total engine air that flows through the engine's core (at flow station 026), inner bypass (2\textsuperscript{nd} stream flow—station 130), and outer bypass (3\textsuperscript{rd} stream flow—station 230) is shown over the course of the Ground Attack mission as a stacked plot in Fig. 24a. The height of each colored section corresponds to the magnitude of its flow, with the three flows adding up to 100% at all times (overboard bleed flow is neglected). A minimum amount of flow is maintained in each flow path throughout the mission, but the amount of bypassed air is a greater portion at reduced power settings. The low altitude penetration leg, which requires significant thrust, sees a significant upswing in core flow.
a) Stacked flow distribution

b) Raw flow distribution

Fig. 24 Ground attack mission flow distribution
This behavior is further emphasized in Fig. 24b, which compares the raw flow rates at the same three engine locations throughout the mission (as percentages of the maximum overall flow seen during the mission) and in Fig. 25, which shows the overall BPR. The total airflow through the engine is at a maximum during the low altitude dash segment. During this segment, all streams see an increase in flow, though it is most pronounced in the core. The BPR is highest during the ground idle segments and during the descent (where minimal thrust is required), at a medium value during cruise, and at lower values during the high altitude loiter and the low altitude dash (where a greater fraction of the engine's total power is required to produce thrust). It is at these high thrust demand conditions that the VCE acts more like a turbojet or traditional low-bypass turbofan than a traditional high-bypass turbofan (though it never closes off the bypasses completely).

![Graph showing overall BPR over time](image-url)

**Fig. 25** Ground attack mission overall BPR
Fig. 26 presents the engine’s total power in four different forms. Fig. 26a shows the thermal energy generation rate (turning the fuel into thermal energy), the thrust power (thrust multiplied by aircraft velocity—pushing the aircraft through the sky), and the shaft power extracted for external loads. The separation between the curves is an indicator of the engine’s overall cycle efficiency. The stacked values of thrust power and shaft power are given in Fig. 26b to illustrate their contributions to the total engine output power as well as the magnitude of that output power throughout the mission. The portions of the thermal energy generation rate that results in thrust or shaft power are shown in Fig. 26c. Again, these curves are indicative of, but not identical to, the overall cycle efficiency. The balance of total output power between thrust and extracted shaft power (and neglecting external bleed) is shown in the stacked plot of Fig. 26d.

![Graphs showing engine power outputs and energy rate input, stacked output powers, output powers as portions of input, and output power balance.](image-url)
Overall, Fig. 26 shows that the periodic load is a fairly sizeable portion of the total power during the loiter segment, but very little during the dash segment. It also shows that even the base load is non-negligible for many mission segments (e.g. subsonic cruises where it represents about 10-15% of the total output power). Note that thrust power is truly zero at ground idle since the aircraft velocity is zero (and therefore the shaft power should constitute 100% of the total output power at zero speed). However, for numerical stability reasons in the model, the Mach number was artificially limited to 0.2 which results in non-zero thrust power during those low speed segments (ground idle, takeoff, and landing). Therefore, approximately the first and last 5 minutes of the mission as shown in Fig. 26 are not accurate.

A major focus of this study is to assess the heat sink capability of the 3rd stream both in cooling external aircraft loads and for internal cooling of turbine cooling air. Therefore, performance in the 3rd stream is examined briefly over the whole mission and then in more detail for important mission segments. Based on the chosen engine arrangement, the 3rd stream air first passes through the HX which dissipates the aircraft loads (including the periodic load) since those can be expected to be at a lower temperature than the CCA which is the second HX in the 3rd stream. For the following discussion, the n identical HXs in an annular arrangement are treated as a single lumped unit for the purposes of presenting heat transfer rates and mass flow rates. (refer to the Duct with Air Gap and Heat Exchanger (Specified $\dot{Q}$) and Fan Duct Heat Exchanger with Air Gap (Calculated $\dot{Q}$) sections for an explanation of this approach).

The heat being sunk to the 3rd stream to cool the external periodic load (and the base aircraft heat load) is shown in Fig. 27. For the mission conditions where the periodic load is active and the aircraft is flying low and fast (500 ft altitude, Mach 1.5), the air in the
$3^{rd}$ stream duct is hot, requiring the waste heat to be lifted significantly to be rejected. This results in both huge shaft power demands to run the cooling system (as shown in Fig. 28) and in huge heat transfer rate peaks shown in Fig. 27. The peak shaft power required of the engine by the base load plus the periodic load (when active) and the cooling system (and accounting for losses due to generator efficiency) is nearly 5000 hp (about 3.7 MW) as seen in Fig. 28. Similarly, the peak heat transfer rate is over 2600 Btu/s (around 2.8 MW) per engine as shown in Fig. 27. The corresponding normalized total pressure drop in the duct (and air gap) for the flow passing through that $3^{rd}$ stream HX is shown in Fig. 29. The drop in total pressure caused by the HX that rejects the external heat load ranges from almost negligible to just over 6% of the total pressure entering the HX. Not surprisingly, there is a rough correlation between the corrected mass flow rate through the HX and the pressure drop that results, as shown in Fig. 30.

*Fig. 27* Ground attack mission $3^{rd}$ stream HX heat transfer rate
Fig. 28  Ground attack mission total LP Shaft power extraction

Fig. 29  Ground attack mission normalized pressure drop through 3rd stream HX
The CCA also passes through a HX in the 3rd stream duct. This HX is downstream of the one that cools the external load and therefore has incoming cold side flow with a lower total pressure but a higher total temperature. The CCA HX hot side (HPC exit bleed) mass flow rate, which is used as CCA for the HPT (Fig. 31), is limited to a maximum value for two reasons. The first reason is that the pressure loss on the hot side of the HX is a strong function of the corrected mass flow rate (similar to what was shown in Fig. 30). To minimize or avoid the need for pressure boosting devices before the CCA is injected at the HPT, pressure losses are minimized by capping the mass flow rate. The second reason for capping the flow rate is that the heat transfer in the CCA HX is subject to the law of diminishing returns. For a given cold side (duct) flow rate, the effectiveness of the HX goes down as the hot side mass flow rate increases. This does not mean that increasing the hot side mass flow rate will not reject more heat to the cold sink (it certainly will reject more);
however, it does mean that each additional pound mass per second of hot side flow rate is less effective at transferring heat than the previous one.

The heat sunk into the 3rd stream to cool the cooling air over the mission, along with the effectiveness in the CCA HX over the mission, is shown in Fig. 32. While the highest heat transfer rate does occur during the low altitude dash, it happens at an effectiveness of less than 0.20. This suggests that the HPC bleed used for CCA represents a significant penalty to the engine’s overall efficiency. However, if engine or aircraft performance at this condition is paramount, then it may be acceptable to sacrifice slightly on engine efficiency to achieve a higher $T_{1040}$ by using additional CCA at the HPT. On the duct side (cold side) of the CCA HX, the normalized pressure loss for the HX (and equivalently through the air gap) is shown in Fig. 33. This pressure loss is significantly higher than that seen for the 3rd stream HX which cools the external load (Fig. 29).
Fig. 32  Ground attack mission heat transfer rate and effectiveness in the CCA HX

Fig. 33  Ground attack mission cold side normalized pressure drop in the CCA HX
The reason for this is that the geometric dimensions and arrangement of CCA HXs is different than that for the HXs that reject the external heat loads. This is further illustrated in Fig. 34 which shows the fraction of the duct flow that enters each of the HXs in the 3rd stream duct. The flow which does not enter the HX passes through the air gaps between HX modules (the pressure drop is balanced between the flow through HX modules and the air gaps). The air gaps are much smaller between CCA HX modules, thus forcing more air through the HXs and incurring a greater total pressure drop. This study did not look to optimize aspects of the two HXs such as the geometry, fin density, and annular arrangement in the duct. However, performing such a study would provide an excellent opportunity to achieve performance benefits by selecting the appropriate arrangement to minimize pressure losses and maximize heat transfer.

Fig. 34  Ground attack mission percentage of mass flow rate entering HX
In looking at the magnitude of power required to drive the cooling system, one might look at ways to drastically reduce this quantity. Indeed, the coefficient of performance (COP) curve fit used for the representative closed-loop air cycle machine is not very high. One might desire to use a vapor cycle system which traditionally has a much higher COP. The problem, however, is that the 3rd stream duct temperature is quite high during certain mission segments as shown in Fig. 35, peaking near 875 °R. There are very few refrigerants that are well suited for rejecting to a sink at this temperature. Even thermal transport fluids such as poly-alpha-olefins (PAO) commonly used on aircraft have coking limits near these duct temperatures, making it difficult to use them for heat transport fluids to enhance heat transfer (Radco Industries, Inc. 2011). The most likely solution to improve the efficiency of the cooling system would be to use a cascaded cooling system whereby the waste heat is initially rejected to a high COP vapor refrigeration cycle.

Fig. 35  Ground attack mission 3rd stream total temperature (at station 240)
The final lift is then made from the vapor cycle to the ultimate heat sink of the 3\textsuperscript{rd} stream using an air cycle system. Another approach that might be considered is to include thermal energy storage devices that provide a local heat sink for peak heat from periodic loads but, in turn, reject only average heat to another sink.

**Mission Leg Performance When the Periodic Load Is Inactive**

A brief discussion of the mission segments with the periodic load inactive provides a point of reference for those segments which do have the load active. Looking first at the ground idle portions of the mission (A and K), the first observation is that the airflow (and corrected airflow) into the engine is far below the design point value. This indicates that flow holding has been terminated. This is not surprising since the engine is at such a low power setting (as low as it would stably operate). As the engine is in steady-state during these mission legs, there are no dynamics of interest to be discussed. During the subsonic cruise legs (C and I), the engine is not in steady-state, but its operation is not of great interest since there are no salient dynamics. Similarly, the climb segment from subsonic cruise conditions up to the loiter conditions (D) happens slowly enough that there are no dynamics of interest. Therefore, the only two segments which may be interesting (when the periodic load is inactive) are the Takeoff/climb (B) and Descend/land (J) segments.

Since more thrust is required for takeoff and climb than at ground idle, the general trend is that: corrected airflow increases; the overall pressure ratio increases; the BPR to the 2\textsuperscript{nd} stream decreases, causing the overall BPR to decrease (though the BPR to the 3\textsuperscript{rd} stream is approximately the same, as shown in Fig. 36); fuel flow increases (especially initially, but less dramatically as the aircraft increases altitude); and internal engine total temperatures and pressures generally increase (except for the 3\textsuperscript{rd} stream total pressure which decreases due to the drastically lower ambient pressure coming into the engine).
Note that TSFC decreases because even though the engine requires more thrust during this segment, it is able to do so more efficiently. The corrected fan speed moves from a very low value at ground idle to nearly the design point value, thus keeping spillage drag low.

A similar assessment can be done for the descent and landing portion of the mission. The aircraft requires very little thrust from the engine as it tries to both descend and decelerate. Therefore the trends seen are very much the opposite of those described for the takeoff and climb.

![Graph showing BPR during ground attack mission takeoff and climb](image)

**Fig. 36** BPR during ground attack mission takeoff and climb

**Mission Leg Performance When the Periodic Load Is Active**

The periodic load is turned on at the beginning of the high altitude Loiter* (E), remains active during the Dive to penetration conditions* (F) and Low altitude
penetration/retreat* (G), and turns off at the end of the Climb* (H). The total heat load and total LP shaft load while the periodic load is active (resulting from the base load plus the periodic load) are given in Fig. 37. During the loiter at a constant altitude, the thrust required to maintain flight at the specified conditions drops slightly as the aircraft mass reduces with fuel burn. Correspondingly, the overall BPR slowly climbs as the demanded thrust slowly goes down since the thrust can be met in an increasingly higher-bypass mode. Also, it should be noted that during the loiter segment, the engine holds very close to 100% corrected fan speed, thus resulting in very little installation drag as shown in Fig. 38. The aft body drag is almost non-existent and the spillage drag offsets less than 4% of the installed thrust produced by the engine.

Fig. 37  Active periodic load: heat transfer rate and LP shaft power extraction
Looking at a single pulse during the loiter segment, there are some variables which stabilize (e.g. LP spool speed) and some which do not (e.g. HPT heat soaked material temperature) during that transient. Fig. 39 shows the LP shaft power extracted in the top portion to provide a clear reference for the timing and magnitude of the pulse. In the middle portion of the figure, the thrust demanded by the aircraft and the installed thrust produced by the engine are plotted together to illustrate the transient effects of the change in shaft loading. As the load comes on, there is a quick spike in thrust followed by a droop before stabilizing back on the demanded value. The upward spike can be attributed to a brief bump in the 3rd stream nozzle exit velocity. This spike caused an error term in the fuel controller that suggested the engine was producing too much thrust.

Fig. 38  Loiter mission segment: corrected fan speed and installation drag
The corresponding small downward notch in the primary combustor fuel flow is in the bottom portion of Fig. 39. Because the thrust error quickly reverses, the fuel flow rate is then increased. However, the increase in fuel flow is not immediate and the thrust therefore has a short transient in its tracking (around 2 seconds in total for the load coming on). When the load comes back off, the process is reversed.

The engine power is shown for the single pulse during the loiter mission segment in Fig. 40. This figure shows that the thermal energy generation rate has approximately the same shape as the fuel flow in Fig. 39. The thrust power is approximately constant since the aircraft simply attempts to maintain steady flight; it is almost exclusively the shaft power output that changes its contribution during the pulse (corresponding to the increase in thermal energy generation rate).
As discussed in the Overview of Engine Power Extraction section, shaft power extraction can have a significant impact on the operating condition within compressors. The surge margin for each of the compressors is shown in Fig. 41. The effect on the fan of the large load being applied to the LP shaft is that the operating point initially moves sharply toward surge due to a sudden bump up in pressure ratio, but then settles on an only slightly lower steady-state surge margin with the load on than with the load off. The LPC, on the other hand, has a very slight reduction in surge margin (also due to an initial bump up in pressure ratio), but then settles on a higher surge margin with the load on because the drop in steady-state LPC pressure ratio more than makes up for the reduction in LPC corrected flow. The HPC, which is thermodynamically coupled but not physically coupled to the LP spool, sees a small initial upward spike in surge margin when the load comes on due to a brief dip in HPC pressure ratio. The HPC then sees a severe dip in surge margin as the
pressure ratio increases significantly. Then as the mass flow through the HPC catches up (it too had a slight initial dip, but its higher steady-state value settles more slowly than the pressure ratio), that surge margin is increased slightly from the minimum for the load-on steady-state point. The reverse effects happen in the compressors during the load removal.

![Graph showing surge margins](image)

Fig. 41 Single pulse during loiter mission segment: compressor surge margins

The effect of the periodic load turning on is a huge torque load on the LP shaft. This results in a slowdown of the shaft, though the increasing fuel flow does cause the LP speed to recover slightly from its minimum as shown in Fig. 42. The HP spool, however, is only thermodynamically coupled to the LP spool and results in an increase in HP spool speed after a very, very slight initial dip (not easily seen in Fig. 42). A final point of interest for a pulse during the loiter mission segment is that the CCA is barely in use during the loiter
mission segment, even during a load pulse, because the HPT inlet temperature is not too close to its limit.

For a pulse during the low altitude dash segment, most of the same behavior is seen as with the pulse during loiter. Fig. 43 is a figure similar to Fig. 39 that depicts the LP shaft power extraction for one pulse, the thrust (demanded and produced), and the fuel flow rates. However, the engine is running with much hotter airflow entering the inlet and the engine thus encounters a $T_{t041,\text{max}}$ limit with the pulse on as shown in Fig. 44. The controller reacts when $T_{t041}$ exceeds its limit and tries to bring the temperature back within range by reducing the primary combustor fuel flow (Fig. 43). It does not bring the temperature back under the limit before the load pulse turns back off.
Fig. 43  Single pulse during low altitude dash mission segment: thrust tracking and fuel flow rate

Fig. 44  Single pulse during low altitude dash mission segment: $T_{041}$
This could be addressed with more aggressive gains in the $T_{r041,max}$ correction to fuel flow (see the discussion in the Fuel Control section), but because the load is expected to be withdrawn quickly, this was not investigated. The decrease in primary combustor fuel flow rate due to the exceeded temperature limit is accompanied by an increase in the afterburner fuel flow rate as the controller attempts to achieve the demanded thrust. As seen by the mismatch between the demanded thrust and the installed thrust produced by the engine (Fig. 43), the controller gains were again not aggressive enough in increasing afterburner fuel flow to achieve the demanded thrust. They do, however, keep the small mismatch from growing dramatically while the load is on.

The surge margins in the compressors were analyzed over one load period during the low altitude dash mission segment and are given in Fig. 45.

![Surge Margin Graph](image)

Fig. 45  Single pulse during low altitude dash mission segment: surge margins
The initial transient behavior is the same as that seen in Fig. 41, but instead of settling on flat surge margins with the load on as seen for the pulse during the loiter, the continually changing fuel flow results in steady slopes to the surge margin curves (especially the fan and LPC) until the load comes back off. The LP and HP shaft speeds again exhibit behaviors opposite one another as seen in Fig. 46—the HP spool speed increases and the LP spool speed decreases with the load on. Note that the HP spool is in overspeed condition (>100% design speed) throughout the pulse. The general rule of thumb used is that turbomachinery and shaft mounted devices should be capable of operating nearly continuously at up to 110% of their design speeds. Based on that criterion, the overspeed condition is not concerning.

![Diagram showing LP and HP spool speeds](image)

**Fig. 46** Single pulse during low altitude dash mission segment: spool speeds
The two contributors to heat transfer into the 3rd stream are shown in Fig. 47. The heat transfer into the 3rd stream HX to cool the external load is simply a step function that mirrors the load itself but at a much higher magnitude (recalling that the approach taken in the “Duct with Air Gap and Heat Exchanger (Specified $\dot{Q}$)” HX was to specify the heat transfer rate into the engine’s 3rd stream directly as $\dot{Q}_{\text{reject}}$ in Fig. 10).

![Graph showing heat transfer rates over time](image)

**Fig. 47** Single pulse during low altitude dash mission segment: heat transfer rates

The CCA HX, on the other hand, has a much more realistic transient response. Notice that the CCA HX heat transfer rate actually decreases while the load is on even though the engine is exceeding the $T_{t041,max}$ limit as noted in Fig. 44. The reason for this is twofold: 1) the mass flow rate in the 3rd stream (HX cold side) is lower during the pulse, and 2) the sink temperature in the 3rd stream is much hotter as the flow enters the cold side.
of the CCA HX when the load is on. This is due to the heat transfer in the upstream HX which first cools the external load and thus heats the 3rd stream air.

While the accelerating dive from loiter conditions to the low altitude dash conditions is certainly a transient maneuver, there are no significantly different phenomenon present during a pulse during that transition. During the decelerating climb from the end of the low altitude dash segment to the start of the subsonic return cruise, five pulses occur. The effects of the pulses during this mission segment are shown in Fig. 48. The load on the LP shaft has a minor, though non-zero, impact on the mass flow rate through the 3rd stream duct. It has a major effect on the heat transfer rate in the HX that dissipates the external waste heat load into the 3rd stream.

Fig. 48 Ground attack mission deceleration/climb mission segment
The heating of the duct air by the external load, in turn, has an effect on the CCA heat transfer rate. Because the cold side flow entering the CCA HX is hotter when the periodic load is on, there is less cooling ability (smaller temperature difference) for the given CCA mass flow rate. The heat transfer rate in the CCA HX is also a strong function of the cold side flow rate (3rd stream duct mass flow rate). The combination of a slight reduction in cold side flow rate (3rd stream duct flow) and a higher $T_{m,cold}$ leads to a reduced heat transfer rate in the CCA HX with the load on.

**Periodic Load Parameter Variations**

Table 3 describes several of the key simulations that were run for the ground attack mission using various values for the periodic load parameters (shown in the Fig. 11 Simulink subsystem mask). Some of the tests produced differences in the results but no surprises. For example, improving the efficiency for the load (case J in Table 3) reduced both the heat rejected to the third stream and the amount of shaft power required.

A noteworthy observation is that the magnitude of shaft power required based on the load efficiency influenced the steady-state value of the surge margins with the load on. For the case with the improved load efficiency (J), the fan had a transient in the surge margin similar to that seen in Fig. 41 as the load pulse turned on (during the loiter), but its value after the initial transient settled out was almost exactly the same as without the load. Case K, which reduced the load efficiency produced the opposite effects from case J. The relationship between load efficiency, the peak LP shaft power extraction, and the peak heat transfer rate for cooling the external load in the 3rd stream is given in Fig. 49. In each case, the peak loads (shaft and heat) happened for a pulse during the low altitude dash segment.
Table 3 Parameter variations for ground attack mission

<table>
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<tr>
<th>Test Case</th>
<th>Successfully Ran to Completion</th>
<th>Load Period [s]</th>
<th>Load Duty Cycle</th>
<th>Load Rise/Fall Time [s]</th>
<th>Load Magnitude per Engine [kW]</th>
<th>Generator Efficiency</th>
<th>Load Efficiency</th>
<th>Notes</th>
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<tr>
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<td>0.85</td>
<td>0.25</td>
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<tr>
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<td>0.25</td>
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</tr>
<tr>
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<td>0.05</td>
<td>500</td>
<td>0.85</td>
<td>0.25</td>
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</tr>
<tr>
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<tr>
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<td>0.05</td>
<td>500</td>
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<tr>
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<tr>
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<td>500</td>
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<td>0.25</td>
<td>Periodic load active throughout entire mission</td>
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<td>500</td>
<td>0.75</td>
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</table>

Note: **Bold italic font** indicates a deviation from the baseline values of case A

Similarly, reducing the generator efficiency (case N) increased the amount of shaft power required to drive the system and increasing the generator efficiency (case M) decreased the required shaft power. There was no change in the heat transfer rate for cooling the load when adjusting this parameter because the cooling system is assumed to only handle losses from the periodic load and base load (dissipating waste heat from inefficiencies in the generator is not considered). Note that this study makes no general judgment on the reasonableness of being able to extract several megawatts of electrical power using a shaft mounted generator.
Whether generator technology to meet this requirement exists or not is outside the scope of this work; these studies simply assumed that the required electrical power could be provided and that the resulting shaft power calculated using a constant value for generator efficiency would be extracted from the engine’s LP spool.

Case L simply was an investigation into whether there would be any dynamics of interest during other mission segments. As expected, the pulses did not produce any unexpected behavior in those mission segments that did not normally have the pulse load active. Having the load active for the entire mission also did not change the behavior during the loiter, accel/dive, low altitude dash, and decel/climb segments.

Other than a corresponding increase or decrease in the magnitude of the shaft power extraction and heat transfer terms, the responses were very similar for the variables of interest using smaller and larger electric loads (cases B and C). The most noteworthy
item is that because in case B the external load HX is dumping significantly less heat into the 3rd stream during the low altitude dash segment, the CCA heat transfer rate is much less affected by the pulses. Conversely, for case C, the CCA heat transfer rate is more affected since the cooling flow has already absorbed significant heat before it enters the CCA HX. The second item of note is that even though a minimal emphasis was placed on the feasibility of shaft mounted generators extracting several megawatts of power, the sheer magnitude of peak shaft power extraction (over 7000 hp) in test case C is questionable.

For the 50% duty cycle and 10% duty cycle cases (D and E), there was little difference versus the baseline. For case D there was sufficient time for most variables to stabilize as they did in the baseline case and even for case E most variables were close to steady state values when the load was removed. The exception was that the heat soak term contributions differed slightly based on how long the load was on versus off. However, the contribution of heat soak terms to the overall performance is relatively small. Again for case F which lengthened the period of the pulse (both the on time and the off time) but kept the same 25% duty cycle, there was minimal difference; the engine variables were simply given more time to stabilize before the next load transient.

As shown in Fig. 50 through Fig. 53, case G does not have sufficient time to settle for each pulse. Rather than show a single pulse, these figures show several pulses over the same time period to simplify a comparison versus the baseline cases (Fig. 39, Fig. 41, Fig. 46, and Fig. 47). Fig. 50 shows the engine controller's attempt to maintain thrust during the large shaft power pulses. The compressor surge margins are shown in Fig. 51. The same transient behaviors are present; they just happen more frequently based on the shorter pulse period. Though the fan and HPC either reach or nearly reach the steady-state values with the load on, the LPC does not settle before the load is removed.
While there do not seem to be any problems due to the frequency at which the pulses occur, the potential does exist where the flow could start oscillating within the compressor, the magnitude of these oscillations could build, and severe damage could occur.

The time variation of the LP and HP spool speeds due to pulses during the low altitude dash is shown in Fig. 52. A minor note of interest is that both the LP and HP spool speeds are slightly higher overall for case G versus the baseline case (Fig. 46). The final figure to illustrate the impact of the shorter period for the external load is Fig. 53 which shows the impact of the pulses on heat transfer rates during the low altitude dash mission segment. As seen in the baseline case, the heat transfer rate in the CCA HX is lower while the pulses are on and the upstream HX rejects significant heat into the airstream.

Fig. 50  Case G (10 second load period) loiter pulses: thrust tracking and fuel flow rate
Additionally, the cold side air flow rate for the CCA HX (3rd stream duct flow) changes with the pulses and is lower when the load is on. This further contributes to fluctuations in the heat transfer rate in the CCA HX.

Cases H and I considered changes to the rise and fall rates of the load as it turned on and off. Case H reduced the rise/fall rate to 0.01 seconds from the baseline value of 0.05 seconds. This test did not complete successfully. The model failed to converge during the first pulse while diving/accelerating to the low altitude penetration conditions. Note that a rise/fall rate of 0.01 seconds means that the load on and load off transients are spread out over only 2 time steps (the model’s fixed time step is 0.005 seconds), whereas the baseline case spreads the loading transient over 10 time steps. This is a significant contributor to the convergence failure.

Fig. 51 Case G (10 second load period) loiter pulses: compressor surge margins
Fig. 52 Case G (10 second load period) dash pulses: spool speeds

Fig. 53 Case G (10 second load period) dash pulses: heat transfer rates
Before the model's failure to converge there were no significant differences in the response to pulses during the loiter segment. Case I, which increased the time to apply or remove the load to 0.1 seconds, ran successfully, but had results that were the same as the baseline case.

**Intercept Mission**

The second mission used to stress the engine operation is a generic intercept mission as given in Fig. 54. The basic legs of the mission, as indicated in the figure, are:

A. Ground idle
B. Takeoff/Climb
C. Accelerate at altitude
D. Supercruise*
E. Accel/Climb for intercept*
F. Supersonic dash*
G. Decel/Descend
H. Subsonic cruise
I. Descend/land
J. Ground idle

* = periodic load is active

The intercept mission replaces the first subsonic cruise mission leg of the ground attack mission with a supercruise leg, eliminates the loiter segment, and goes up instead of down in altitude for its main action segment. In the intercept mission, the aircraft climbs and accelerates to a maximum speed at a high altitude rather than performs a low altitude dash. After the high altitude dash, the aircraft decelerates and descends for a subsonic cruise and mission conclusion that is the same as in the ground attack mission. Though the intercept mission has a more aggressive takeoff, climb, and acceleration (compared to the ground attack mission), there is nothing of great interest during those segments not already addressed for the ground attack mission for the baseline case.
Most of the cases shown in Table 3 exhibit similar behavior for the intercept mission to that seen for the ground attack mission (addressed in the Periodic Load Parameter Variations section within the Ground Attack Mission section), but the exception is case C, which had a larger electrical load magnitude specified. In this case for the intercept mission, the model failed to converge during the acceleration segment that precedes the supersonic dash at high altitude. Whereas the key action segment of the ground attack mission resulted in the maximum overall airflow occurring during the low altitude dash (Fig. 24b), the intercept mission has its maximum flow during the initial takeoff climb. High altitude operation requires significantly less airflow, as shown in Fig. 55.
The thrust response for a pulse during supercruise (baseline case) is shown in Fig. 56. The response is very similar to that seen in the ground attack mission’s subsonic loiter (Fig. 39), though slightly more underdamped for the intercept mission. The intercept also sees similar shapes to the curves for the output power from the engine (like Fig. 40) except that the magnitudes of the thrust power output and thermal energy generation rate are not the same between missions. During the intercept mission's supercruise, the thrust power and thermal energy generation rate are about three times higher than for ground attack’s subsonic cruise due to the significantly higher speed. However, during the respective dash segments, the high altitude dash of the intercept mission requires only about half as much thrust power (and one-third the thermal energy generation rate) as the low altitude dash of the ground attack mission.

Fig. 55 Intercept mission raw flow distribution
The surge margin of the compressors for a pulse during supercruise is shown in Fig. 57. Again, the initial transients are very similar to that seen for pulses during subsonic loiter in the ground attack mission (Fig. 41), though more underdamped for the intercept mission. However, the fan surge margin settles on a slightly higher value with the load on for the intercept mission’s supercruise than for the ground attack mission’s subsonic cruise.

During the intercept mission’s high altitude supersonic dash, no pulses happen entirely within the segment. One pulse was in progress during the transition from the acceleration segment to the dash segment and the next pulse started during the high altitude, high speed dash but was cut short due to the end of the segment when the periodic load became inactive for the remainder of the mission. The two pulses are shown in Fig. 58 with the LP shaft power extraction, the thrust, and the fuel flow rates each plotted.
Fig. 57  Single pulse during intercept supercruise mission segment: surge margins

Fig. 58  Load pulses during intercept mission high altitude supersonic dash
As with the ground attack mission’s low altitude dash, the high altitude supersonic dash of the intercept mission encounters the \( T_{t041,max} \) limit when the pulses are on (Fig. 59). As was seen in the ground attack mission, the magnitude of the external heat load rejected into the 3\(^{rd}\) stream limits the amount of heat transfer that can occur for the CCA (Fig. 60). Also, the change in 3rd stream duct mass flow rate at the mission segment change causes a kink in the CCA heat transfer rate curve.

![Graph showing \( T_{t041} \) with load on during intercept mission high altitude dash](image)

Fig. 59 \( T_{t041} \) with load on during intercept mission high altitude dash
The last mission used to stress the engine operation is a generic dogfighting (air-to-air combat) mission as given in Fig. 61. Because the key portion of this mission takes place over a very short period of time, that portion of the mission is expanded in Fig. 62. The basic legs of the mission, as indicated in the figure, are:

A. Ground idle
B. Takeoff/climb
C. Accelerate at altitude
D. Supercruise*
E. Dive*
F. Combat maneuvers*
G. Decel/climb*
H. Subsonic cruise
I. Descend/land
J. Ground idle

* = periodic load is active

Fig. 60 Heat transfer rate during intercept mission high altitude dash

Dogfight Mission

The last mission used to stress the engine operation is a generic dogfighting (air-to-air combat) mission as given in Fig. 61. Because the key portion of this mission takes place over a very short period of time, that portion of the mission is expanded in Fig. 62. The basic legs of the mission, as indicated in the figure, are:

A. Ground idle
B. Takeoff/climb
C. Accelerate at altitude
D. Supercruise*
E. Dive*
F. Combat maneuvers*
G. Decel/climb*
H. Subsonic cruise
I. Descend/land
J. Ground idle

* = periodic load is active
The dogfight mission is similar to the intercept mission in that it involves a more aggressive takeoff/climb and supercruise. However, it is more like the ground attack mission in that its key action segment is at a low altitude. The dogfight mission is unique that its action segment is a series of altitude and Mach number changes. These are intended to very crudely model aggressive throttle transients to represent combat. Because only the Dive*, Combat maneuvers*, and Decel/climb* mission segments are unique to the dogfight mission, those are the only ones addressed in this section.

Though the dive segment starts from a higher speed in the dogfight mission than in the ground attack mission, there is nothing drastically different seen during that mission segment. Similarly, there is nothing of particular note during the decel/climb back up to the
subsonic cruise conditions. It is only the combat portion itself which is of interest. Because this portion is so short in duration (about 9 seconds), the exact placement of the pulse is influential in the response.

![Graph](image_url)

**Fig. 62** Dogfight mission profile with focus on air-to-air combat segments

For the first test, the timing was such that the pulse was on for the entire combat portion. Fig. 63 shows engine performance details during the combat maneuvers. The altitude and Mach number, which govern the throttle transients, are shown in the top portion. Next is the LP shaft power extraction, which both indicates when the pulse is on and shows how the magnitude of power required changes with the operating conditions. Third from the top in the figure is the thrust tracking. It is clear that the engine controller does not perfectly track the demanded thrust. It is also apparent that there are small
oscillations in the thrust when it is near its minimum value. The explanation for this is seen in the bottom portion of the figure. The fuel flow rate hits its minimum allowed by the controller. Though the controller has anti-windup to prevent integrator term problems when coming off a saturated minimum, the gains introduce oscillations when coming off that limit. Similar to discussion in earlier sections, the controller could be adjusted to prevent problems like this from occurring.

![Graphs showing various parameters over time](image)

**Fig. 63** Dogfight mission combat maneuvers with pulse on
The engine comes on and off of a maximum $T_{t041}$ limit during the mission segment as seen in Fig. 64, but the gains are not aggressive enough to bring the temperature back down before the next throttle transient occurs. The compressor surge margins are shown in Fig. 65. Though the surge margins bounce up and down with the throttle transient, the surge margin oscillations induced by the oscillating fuel flow do not present any unrecoverable instabilities in the compressors. The final consideration for the combat transients with the pulse on the whole time is shown in Fig. 66. This figure shows the heat transfer rate into the 3rd stream in both the external load HX and in the CCA HX. It can be seen that the CCA HX heat transfer rate is most strongly a function of the cold (duct) side flow rate, but it is also affected by the hot side flow rate which does dip down briefly and by the temperature in the duct as illustrated by the oscillations appearing in that heat transfer rate. Recall that the external load heat transfer rate is specified directly and is therefore not strongly affected by fast air temperature oscillations caused by fan PR fluctuations.

Fig. 64 Dogfight mission combat maneuvering: $T_{t041}$
Fig. 65 Dogfight mission combat maneuvering: compressor surge margins

Fig. 66 Dogfight mission combat maneuvering: heat transfer rates
A second test case was run that used a 10 second period for the pulse (case G in Table 3), which resulted in a short pulse being entirely contained within the combat maneuvering portion of the mission. There was actually very little difference between this case and the baseline dogfight mission (which had the pulse on for the entire combat maneuvering portion of the mission) for engine performance type variables. The heat transfer was the primary difference. Fig. 67 shows the heat transfer into the 3rd stream from both HXs for this alternative pulse profile. Though the mass flow rates still govern the heat transfer rate in the CCA HX, the impact of the pulse can again be seen as it lowers the heat transfer rate in the CCA HX.

Fig. 67  Combat maneuvering with short (10 second period) pulse
IV. CONCLUSIONS

A generic double bypass (three stream) turbofan VCE model was developed in NPSS. A simple controller was developed in Simulink to control the engine’s variable geometry and fuel flows. Simple verification steps were taken to build confidence in the model assembly. Examples of these steps included determining the appropriate simulation time step size, ensuring that absolute temperatures and pressures were non-negative, and ensuring the conservation of mass through the engine.

Three generic missions were created to exercise the engine model. Each of these missions was designed to stress certain aspects of the aircraft/engine. Additionally, the engine was subjected to periodic pulses from a low efficiency electric load on board the aircraft. The waste heat from this load was lifted with a cooling system and rejected into the 3rd stream. The power required to drive both the load itself and the cooling system was extracted from the engine shaft.

Particularly challenging mission segments included the ground attack mission’s low altitude dash (where the air is hotter and more dense, creating significant drag), the intercept mission’s high altitude maximum speed dash (where the aircraft’s high Mach number drives up the temperature in the 3rd stream duct), and the dogfight mission’s combat maneuvering (where the throttle transients make thrust tracking difficult). Table 4 provides a concise summary of the 3rd stream duct mass flow rate, the total temperatures entering each 3rd stream HX, and heat transfer rates in each HX. In this table it can be seen that the intercept mission produced the highest 3rd stream duct temperatures.
Correspondingly, the intercept mission also had the highest value of the heat transfer rate for the external load since it required significant lift by the cooling system in order to have a sufficient temperature difference to enable the waste heat to be rejected into the hot duct flow. The highest mass flow rate in the third stream duct was seen during both the intercept mission and the dogfight mission (during the aggressive takeoff/climb mission segment). All three missions had approximately the same heat transfer rate maximums into the 3rd stream for the cooling of turbine cooling air. The intercept mission had by far the highest peak shaft power requirement. Since the electrical load was the same in each case, this peak value was tied to the amount of power required by the cooling system. Therefore, it was not surprising that the mission with the highest peak heat transfer rate for cooling the external load would also have the highest peak shaft power load.

Table 4 Summary of 3rd stream HXs and LP shaft power extraction

<table>
<thead>
<tr>
<th></th>
<th>Ground Attack Mission</th>
<th>Intercept Mission</th>
<th>Dogfight Mission</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>min</td>
<td>mean</td>
<td>max</td>
</tr>
<tr>
<td>3rd stream mass flow rate, [lbm/s]</td>
<td>18</td>
<td>59</td>
<td>196</td>
</tr>
<tr>
<td>3rd stream total temperature entering external load HX, [°R]</td>
<td>545</td>
<td>595</td>
<td>873</td>
</tr>
<tr>
<td>3rd stream total temperature entering CCA HX, [°R]</td>
<td>553</td>
<td>614</td>
<td>1030</td>
</tr>
<tr>
<td>3rd stream external load heat transfer rate, [Btu/s]</td>
<td>100</td>
<td>197</td>
<td>2640</td>
</tr>
<tr>
<td>3rd stream CCA heat transfer rate, [Btu/s]</td>
<td>&lt; 1</td>
<td>639</td>
<td>2215</td>
</tr>
<tr>
<td>LP shaft power extraction, [hp]</td>
<td>500</td>
<td>680</td>
<td>4925</td>
</tr>
</tbody>
</table>

Note: **Bold italic font** indicates the mission(s) with the extremes
The transients associated with the periodic loading (pulses) and the flight condition changes dictated by the mission were the drivers for the studies presented. Overall, large amounts of shaft power could be extracted from the LP spool while maintaining engine stability. Though the engine control system developed was unsophisticated, it provided reasonable control over the engine. Certain cases were found where the gains used in the controller were not ideally suited. They either introduced small oscillations or allowed a small steady-state error to remain.

The thermodynamic relationship between the HX in the 3rd stream for cooling the external (periodic) load and the CCA HX was identified. Though the heat transfer from the HPC bleed to the 3rd stream air via the CCA HX was more strongly a function of the mass flow rate in the 3rd stream duct, the magnitude of the heat load being rejected in the upstream HX also had an influence. The split of the mass flow rate between HX blocks and the air gaps between them was described and the balanced pressure drop through the HX and air gap was explained. The temperature of the 3rd stream duct varied greatly throughout the mission. The highest temperature seen in the duct dictated that a vapor cycle cooling system using traditional refrigerants was infeasible and a more exotic approach would be required to achieve an efficiency (i.e. COP) benefit over the assumed air cycle baseline.

While no assessment was done concerning the feasibility of generator hardware being capable of efficiently and reliably extracting the required power, the studies presented indicated that very large amounts of power could be extracted from the engine while maintaining stability. Based on the generic low efficiency loads and simple cooling system representation used in these studies, the amount of power required by the cooling
system lift the waste heat to a reasonable temperature for rejection into the 3rd stream requires significantly more power than the electrical load itself.

The various parameters that defined the periodic load were adjusted to assess their impact on simulation results. The rise/fall time of the periodic load's on and off transients was found to be a fairly unimportant parameter with the caveat that there be a sufficient number of model time steps to discretize the slope to allow convergence at enough intermediate points between the full on and full off positions. The load duty cycle parameter did not impact the results significantly for most of its range. Only for extreme values close to zero or one did it impact results significantly. In those cases, the transients were not given sufficient time to stabilize. However, this did not produce system instabilities (e.g., unbounded growth in oscillations) for any of the test cases. Similarly, the load period parameter did not have a significant effect when set to larger values, but when set to sufficiently small values, the load transients again did not stabilize fully. As expected, the magnitude of the load itself, the efficiency of the load, and the efficiency of the generator all had significant impact on the amount of shaft power required from the engine. Additionally, the load efficiency and the magnitude of the load factored in to the magnitude of the heat load rejected into the 3rd stream.

**Future Work**

This work represented a significant step forward in modeling the transient effects of a relatively complex engine cycle. Of particular interest in this model was looking at heat rejection into the 3rd stream. To accomplish this, two HXs were included in the 3rd stream duct. The first one was a one sided HX whereby the heat transfer rate is specified and only the duct pressure drop is modeled. This is a significant simplifying assumption. However, to account for the behavior in the HX properly, a complete representation of the cooling
system is required. Future work in developing a cooling system model or integrating this engine model with an existing cooling system model would be valuable.

Adding transient effects to the heat transfer process within the HXs would appropriately increase the model detail. Currently, the HX model does not account for the thermal mass of the HX itself or take into account the time dimension of the heat flow from one side to the other. It simply uses a heat transfer rate to change the flow properties instantaneously on each side of the HX.

Another significant shortcoming of the current modeling approach is a lack of volume dynamics. As discussed in the Physics Not Modeled section, there is more than one way to approach modeling the volume dynamics. However, the current approach of ignoring them is not the most appropriate for an engine which both has large volumes and quickly modulates flow between those volumes (flow paths) to achieve better performance. Based on the time steps currently used for the model and the potential need to further reduce the time step anyway if realistic load rise/fall times should dictate it, the approach of modeling every volume in the engine may not impose a significant additional computational burden.

Future work will also likely include integrating the engine model with a higher fidelity aircraft model, aircraft thermal management system model, fuel thermal management model, and electrical system models to form an integrated tip-to-tail aircraft model. In such an integrated simulation, more realistic missions could be studied, the role of fuel in the thermal management system could be assessed, the drain sequence and bulk temperature fuel in tanks could be evaluated, and more realistic load usage profiles could be used.
Concerning the engine model itself, many of the elements should be modeled in more detail. Examples of components that deserve a more detailed treatment are the inlet, nozzle, and combustor, which operate as nearly ideal devices in the current model. Additionally, if sufficient data is available, it would be advantageous to model turbomachinery components in a stage-by-stage manner and more accurately account for cooling air injection in the turbines.

Future work should also include transitioning from the generic adaptive turbine engine presented in this research to an existing or developmental engine architecture for which some validation data is available. The preliminary steps taken in verifying the soundness of the modeling approach do build confidence, but validating a model against the performance of physical hardware is a clear next step.

While the controller developed for this research provided a stable engine, it is overly simple. Not only does it hold some variable geometry parameters as constants, those that the controller does adjust are simply controlled in an open loop fashion based on table data generated a priori from a matrix of steady state solutions. Furthermore, the logic used to determine the afterburner operation is imperfect and should be adjusted to provide a more robust system. Finally, the gains used for the fuel controller should be tweaked to both ensure stable control and optimize the thrust response characteristics of the engine.
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