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A Technique for Integrating Engine Cycle and Aircraft Configuration Optimization

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Summary

A method for conceptual aircraft design that incorporates the optimization of major engine design variables for a variety of cycle types has been developed. The method should improve the lengthy screening process currently involved in selecting an appropriate engine cycle for a given application or mission. The new capability will allow environmental concerns such as airport noise and emissions to be addressed early in the design process. The ability to rapidly perform optimization and parametric variations using both engine cycle and aircraft design variables, and to see the impact on the aircraft, should provide insight and guidance for more detailed studies.

This paper begins with a brief description of the aircraft performance and mission analysis program and the engine cycle analysis program that were used in this work. A new method of predicting propulsion system weight and dimensions using thermodynamic cycle data, preliminary design, and semi-empirical techniques is introduced. Propulsion system performance and weights data generated by the program are compared with industry data and data generated using well established codes. The ability of the optimization techniques to locate an optimum is demonstrated and some of the problems that had to be solved to accomplish this are illustrated. This paper concludes with results from the application of the program to the analysis of three supersonic transport concepts installed with mixed flow turbofans. The results from the application to a Mach 2.4, 5000 n.mi. transport suggest that the optimum bypass ratio is near 0.45 with less than 1% variation in minimum gross weight for bypass ratios ranging from 0.3 to 0.6. In the final application of the program, a low sonic boom fixed takeoff gross weight concept that would fly at Mach 2.0 overwater and at Mach 1.6 overland is compared with a baseline concept of the same takeoff gross weight that would fly Mach 2.4 overwater and subsonically overland. The results show that for the design mission, the low boom concept has a 5% total range penalty relative to the baseline. Additional cycles were optimized for various design overland distances and the effect of flying off-design overland distances is illustrated.

Introduction

In the early stages of aircraft conceptual design, a major stumbling blocks is often a lack of realistic propulsion system performance data. When accurate propulsion system performance data are available, or the capability to generate the data exists, the question arises: is the selected engine cycle right for the mission and baseline aircraft of interest? At this point the lengthy and often costly process of screening a variety of engine cycles, and design options for a given cycle, begins. The difficulty involved in matching an engine cycle and aircraft to a given mission typically increases as the Mach number increases or as the complexity of the mission increases. This difficulty is further complicated by environmental issues, such as nitrous oxides emissions and ozone depletion, airport and community noise, and sonic boom.

Engine cycle design has historically been driven by technology availability, cost, and mission requirements and, until recently, often has involved little interaction between the engine manufacturer and airframe manufacturer or customer (ref. 1). The available materials, engine life requirements, and high reliability typically set the temperature limits, that in turn set overall pressure ratio and turbine inlet temperature. The turbofan engines that powered the commercial transports in the early 1960's typically had bypass ratios near 2.0 and overall pressure ratios of 16 to 18 (ref. 2). As materials and turbine cooling technology advances allowed increased overall pressure ratios and turbine inlet temperatures, bypass ratios increased and specific fuel consumption decreased. By 1969, when the first Boeing 747 flew, Pratt & Whitney's JT9D series had a bypass ratio near 5.0 and an overall pressure ratio near 27. SNECMA and General Electric's CFM56 series, that began flying in the early 1980's, had bypass ratios up to 6.6 with overall pressure ratios up to 37.4. The GE90, scheduled for entry into service in 1995, will have a bypass ratio of 9 and an overall pressure ratio exceeding 40 (refs. 3 and 4). Additionally, the development of the GE90 represents a change in the philosophy in industry, from the outdated in-house design and sell, to the establishment of working groups that include "representatives from design, manufacturing, marketing, product support, sourcing, suppliers, and customers" (ref. 4). Though a heightened awareness of public concern over the environment had, and will continue to have an impact on the design of gas turbine engines for subsonic transports, the relative simplicity of the mission allows a less integrated approach to design than the rather integrated approach that the design of a multi-role fighter or supersonic transport might require.

A literature survey suggested that very little has been done on the subject of directly combining engine cycle analysis and optimization with aircraft preliminary conceptual design and optimization. In the cases where cycle optimization was performed, a pre-selected matrix of engine decks was used (ref. 5, and 6). In the method described in this paper the optimizer has direct control over engine cycle and aircraft design variables. In such cases, contour plots may be used to access the impact of design variables on the resulting design and constraints. In another case, equations, derived for the objective and constraints as functions of the design variables, are input as analytical functions to be optimized (ref. 7). Optimization of the entire propulsion system, accounting for performance, weight, life cycle costs, and installation effects, is documented in reference 8.

The purpose of the research described in this paper is to incorporate an engine cycle analysis capability into an aircraft conceptual design and optimization computer program and to include engine cycle design variables directly in the optimization process. The cycle analysis program selected for this purpose needed to have the capability to accurately, rapidly, reliably, and precisely simulate a variety of cycles. To maintain a high level of accuracy, it is necessary to have models that accurately simulate the off design characteristics of the individual components in the cycle. One such program is the Navy Engine Performance Computer Program (NEPCOMP, ref. 9). The Navy/NASA Engine Program (NEPP89, ref. 10) greatly expanded the capabilities of NEPCOMP to include the ability to simulate cycles with multiple modes of operation, to include the effects of chemical dissociation and a variety of fuel types, and it includes routines to allow plotting of compressor and turbine maps. While all these enhancements make the program much more versatile and user friendly, the added capabilities also greatly increase the size of the program and add complexities to the analyses, possibly leading to increased execution times and reduced reliability. Because reliability and execution time are of paramount importance, Quick NEP (QNEP, ref. 11), a modified version of NEPCOMP, was selected for this research. QNEP can use component maps to model off design operating characteristics of most components in the cycle and can accurately model the most commonly used cycles, including mixed and separate flow turbofans and turbojets.

This paper contains an overview of the analysis tools used for this research with emphasis on the cycle analysis capability. Results directed toward validating the new capability will be presented. An overview of the optimization capability in the program will be given along with a discussion of the problems initially encountered during optimization. The results of several optimization test cases will be presented along with an assessment of the validity of the resulting designs. This paper will conclude with results, illustrating the application of the program to a Mach 2.4 civil transport.

List of Major Symbols and Abbreviations

BPR	Bypass Ratio (bypass airflow / core airflow)
CDT	Compressor Discharge Temperature, °R
EIS	Engine In Service
FAR	Federal Aviation Regulation
FLOPS	FLight OPtimization System
FPR	Fan Pressure Ratio
MFTF	Mixed Flow Turbofan
NEPCOMP	Navy Engine Performance Computer Program
NNEP	Navy/NASA Engine Program
NOx	Nitrous Oxides
OPR	Overall Pressure Ratio
QNEP	Quick Navy Engine Program
SFC	Specific Fuel Consumption
SW	Wing Area, ft.^{2}
T4 or T_4	Burner outlet temperature, °R
T41 or T_{41}	Turbine inlet temperature, °R
TBE	Turbine Bypass Engine
TOFL	Takeoff Field Length, ft.
TOGW	Aircraft Takeoff Gross Weight, lb
Vapp	Approach speed, knots

I. Aircraft Conceptual Design System

A system which integrates engine cycle analysis into preliminary aircraft conceptual design has been developed. The primary focus of this work was the development and integration of a cycle analysis module into the FLight OPtimization System (FLOPS, ref. 12). FLOPS is a multidisciplinary system of computer programs for conceptual and preliminary design and evaluation of advanced aircraft concepts illustrated in figure 1.1. FLOPS can be used to perform a design point calculation, parametrically vary one or more design variables, perform an optimization, or parametrically vary any two design variables and prepare data for contour plotting. In addition to performing a complete

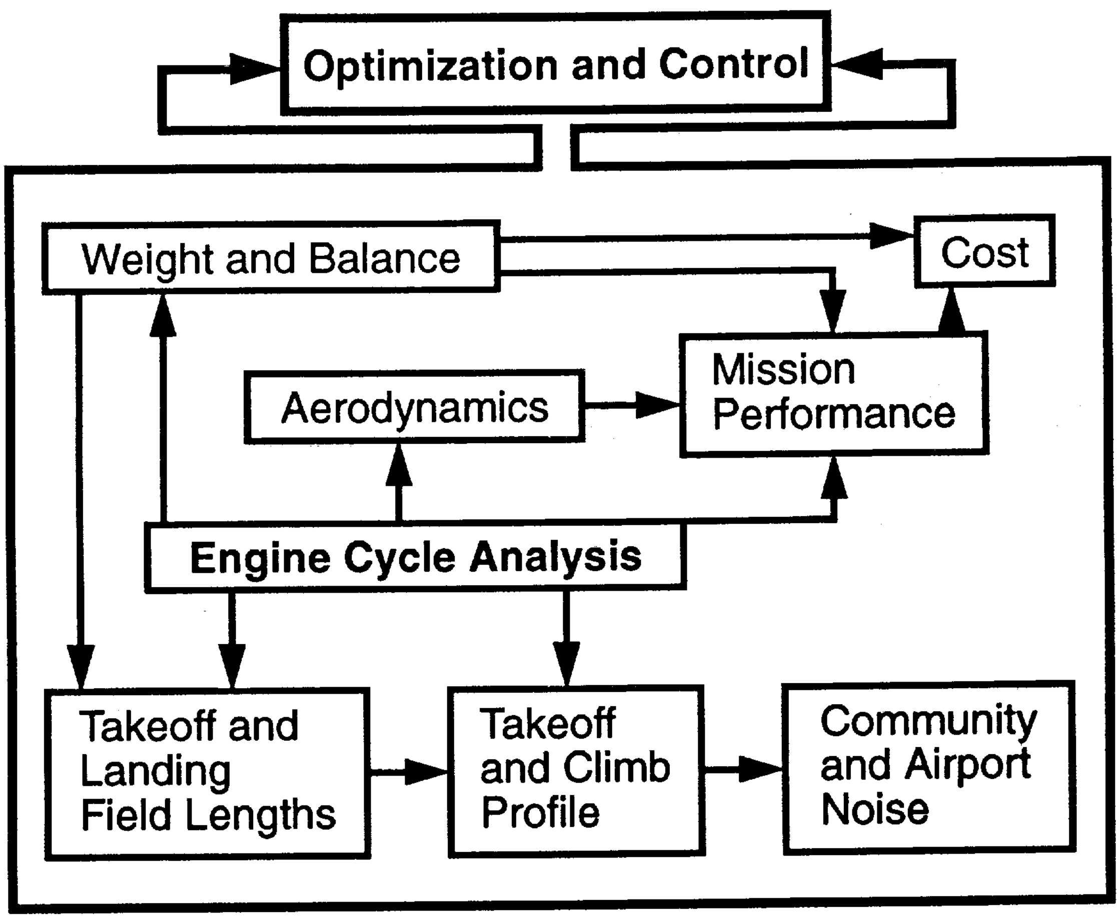


Figure 1.1 - Aircraft conceptual design system schematic.

analysis, including mission, FLOPS can do partial analyses, including weights, weights and aerodynamics, or propulsion only. The available aircraft design variables are aircraft takeoff gross weight, wing aspect ratio, takeoff thrust, wing area, wing taper ratio, wing sweep, and wing thickness to chord ratio. The available mission design variables are cruise Mach number and cruise altitude and the available engine cycle design variables are burner temperature, overall pressure ratio, fan pressure ratio, bypass ratio, and throttle ratio[1] or maximum allowable burner temperature. There are five available optimization algorithms in FLOPS, two quasi-second order methods of Davidon-Fletcher-Powell and Broyden-Fletcher-Goldfarb-Shano, the Polak-Ribiere variant of the conjugate gradient method, the first order steepest descent method, and a univariate search. The optimization is performed as a series of minimizations, called drawdowns, of the Fiacco-McCormick penalty function of the form:

$$F = \mathrm{OBJ} + \mathrm{R}_K \times \sum \frac{1}{G_i}$$

Here OBJ is the objective function, which may be aircraft takeoff gross weight, fuel burned, cruise Mach number multiplied by the lift to drag ratio, range, cost, specific fuel consumption, or total nitrous oxides emitted or a combination of these defined by user input weighting factors. G, the behavioral constraint parameters, are defined as:

$$G = \left(\frac{\text{value}}{\text{upper}\,limit} - 1\right) \text{or} \left(1 - \frac{\text{value}}{\text{lower}\,limit}\right)$$

On each successive drawdown R_K (the value of R for the K^{th} drawdown) may be reduced based on user input so that the behavioral constraints become less important. Some of the available constraints are lower limit on range, upper limit on approach speed, upper limit on takeoff field length, upper limit on landing field length, lower limit on missed approach climb gradient thrust, lower limit on second segment climb gradient thrust, upper limit on fuel volume, upper limit on jet velocity, and upper limit on emissions of nitrous oxides. The optimization algorithm to be utilized, the convergence criteria, the step size to be used for computing gradients, and more may all be controlled through user input.

The remainder of this chapter will briefly describe the primary technical modules in FLOPS, with the new cycle analysis capability described in more detail in chapter III.

Mission Analysis

Mission analyses can be performed for fixed takeoff gross weight or for fixed range. The mission

1. Throttle ratio is defined here as the ratio of the maximum allowable burner temperature divided by the design point burner temperature.

may be defined with as many as 40 segments using the segment types defined in table 1.1 There may

Segment	Primary Input
START	Starting Mach number and altitude.
CLIMB	Climb schedule number.
CRUISE	Cruise schedule number and total distance to end.
REFUEL	Fuel added and time required.
RELEASE	Weight released.
ACCEL	Engine power setting and ending Mach number.
TURN	Turn arc and engine power setting or turn acceleration.
HOLD	Cruise schedule number and time.
DESCENT	Descent schedule.
END	Ending Mach number and altitude.

Table 1.1 - Mission Segments

be as many as six different cruise schedules defined for the main mission. Cruise schedules may be for maximum specific range (velocity / fuel flow) or minimum fuel flow with Mach number and/or altitude varying, for maximum altitude at a fixed Mach number, for a fixed altitude and lift coefficient, or for maximum Mach number at a fixed or optimal altitude. Up to four climb schedules are allowed. Climb profiles may be explicitly defined or the optimum Mach number and/or altitude are computed for some combination of minimum fuel and time to distance or for some combination of minimum fuel and time to climb. For the mission analyses, tables of altitude, Mach number, fuel flow, and other relevant mission data versus weight for each of the cruise schedules are generated. These schedules cover the range from empty weight to takeoff gross weight and are updated as necessary. One of the cruise segments, the "*free*" segment, does not have a range associated with it. Mission analyses start at START and proceed to the *free* segment and then skip to END and proceed backwards to the *free* segment. The *free* segment is then flown as far as possible with the available fuel. The *free* segment is usually the highest altitude (figure 1.2). If any DESCENT segments are followed by CLIMB segments, all but one of those DESCENT segments will be instantaneous (i.e. zero fuel, time, and distance) and the *free* segment must be the final CRUISE segment. In order to perform any mission analyses, the program must have propulsion system performance data, weights data, and aerodynamic data.

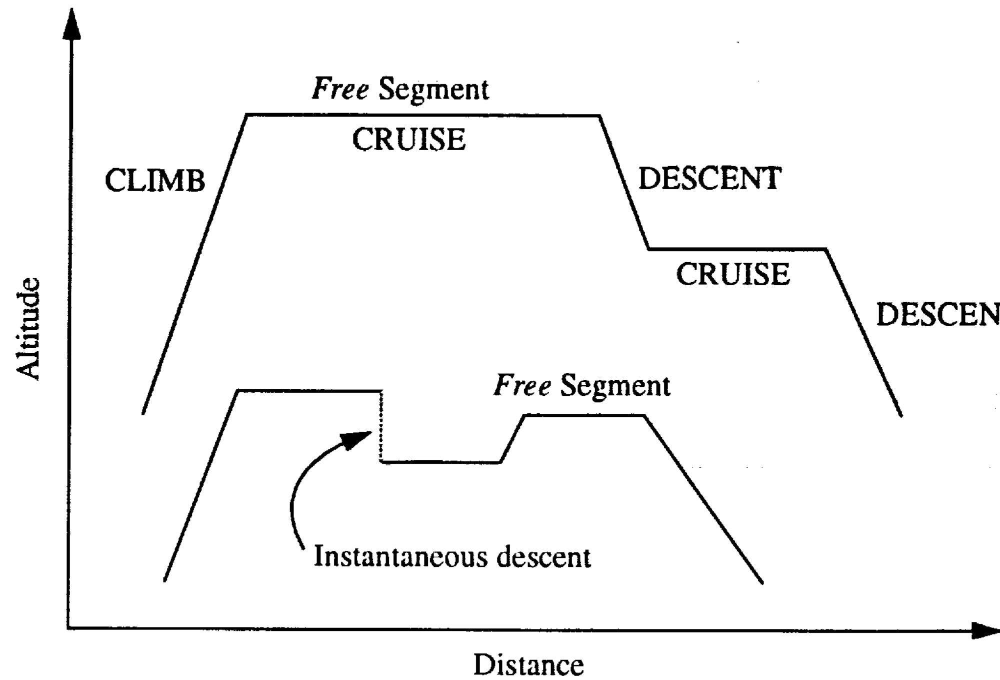


Figure 1.2 - Mission profile sketch.

Weights

Weights in FLOPS are generally computed using equations derived from a data base of existing aircraft. Weights are predicted for all components listed for each group shown in table 1.2. In addition

Group	Components
Structure	Wing, Horizontal tail, Vertical tail, Fuselage, Landing gear, Nacelle
Propulsion	Engines, Thrust reversers, Miscellaneous Systems, Fuel system
Systems & Equipment	Surface controls, Auxiliary power unit, Instruments, Hydraulics, Electrical, Avionics, Furnishings, Air conditioning, Anti-icing
Operational	Crew & Baggage, Unusable fuel, Engine oil, Passenger service, Cargo containers
Payload	Passengers, Baggage, Payload

Table 1.2 - Aircraft Component Summary

to the empirical equation, a more analytical method for predicting wing weight requiring a more detailed input description of the wing geometry is available. A more analytical method for predicting propulsion system weight will be described in the next chapter. User input can be used to scale or override any of the computed weights.

Aerodynamics

Drag polars are predicted using a modified version of the Empirical Drag Estimation Technique (EDET, ref. 13) with skin friction drag estimates based on the Sommer and Short T′ method (ref. 14). Alternatively, user defined drag polars may be input and scaled with variations in wing area and engine nacelle size. There are a variety of formats that may be used for introducing aerodynamic data including data which may be used to override the aerodynamic tables in EDET, scalable aerodynamic data, or coefficients for aerodynamic data in parabolic format. Scalable aerodynamic data requires wetted area and equivalent length data and two sets of wave drag polars, one for nacelles on and another for nacelles off. FLOPS then uses an unpublished method to account for the effects of changes in nacelle size and wing area independent of each other and of the fuselage. The parabolic format is used by FLOPS to generate tables of drag coefficient (C_d) as a function of Mach number and lift coefficient (C_L) and has the form:

$$C_{d_{J,I}} = C_{d_{min_I}} + C_{k_I} \cdot (C_{L_J} - C_{\mathrm{LB}_I})^2$$

The zero lift drag coefficient ($C_{d,min}$), the coefficient (C_k), and the minimum drag lift coefficient (C_{LB}) are all input as functions of Mach number and FLOPS fills in the drag table at the discrete lift coefficients $C_{L,J}$. Additional low speed aerodynamic data may be supplied when the detailed takeoff and landing module is used.

Takeoff and Landing

The takeoff and landing module calculates the takeoff and landing field length subject to all FAR Part 25 regulations including second segment climb thrust gradient and missed approach climb gradient (figure 1.3)[1]. For landing, the start of flare altitude is determined by iteration, such that the vertical acceleration and velocity are zero at the ground. During flare the aircraft is rotated at a constant rate and thrust is reduced so that idle thrust is reached at the ground, provided that the horizontal velocity remains above the lift off velocity and the rate of descent does not increase. The FAR landing field length is the actual landing field length divided by 0.6.

Balanced takeoff field length includes one engine out takeoff, all engines operating aborted takeoff, and one engine out aborted takeoff (figure 1.4). The critical engine failure speed, V_{ef} is

1. The climb gradients shown are for 4 engine transports and vary for other configurations.

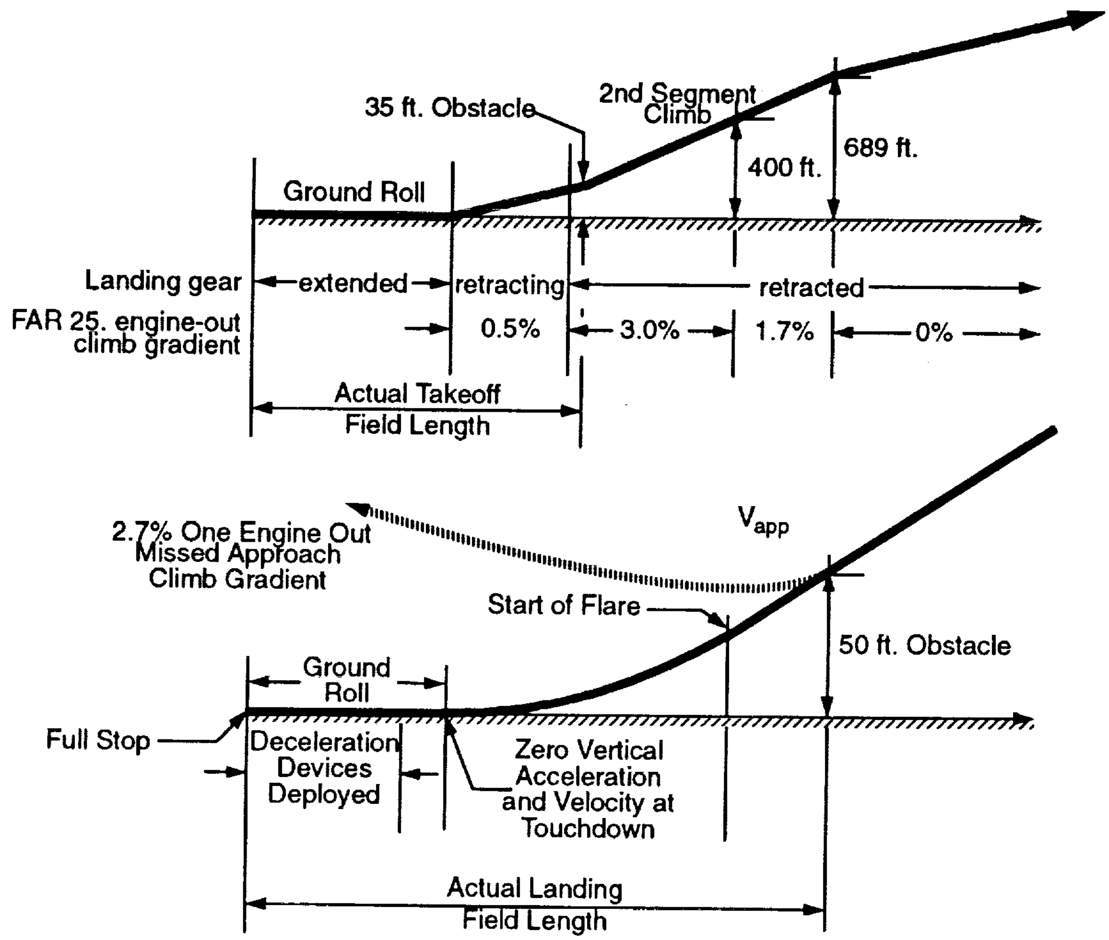


Figure 1.3 - Takeoff and landing profiles.

determined by iteration, such that the balanced field length calculated as shown for case A is equal to the maximum of the distances to full stop in cases B and C in figure 1.4. Since the aircraft must be able to rotate 5 knots early without increasing takeoff distance, if the critical engine failure speed is less than 5 knots from the rotation speed the rotation speed is reset to V_{ef} + 5 and the balanced field length is recomputed. The information generated here is used in generating detailed takeoff and climbout profiles and for predicting takeoff and landing noise.

Takeoff and Climb Profile

Detailed takeoff and climbout profiles may be generated for a variety of procedures, including varying flap settings and thrust levels. Detailed climbout profiles begin at brake release and consist of segments defined in table 1.3. Rotate, liftoff, and last segments are required and as many as seventeen additional segments may also be included. Each segment after the obstacle is defined by one of the constraints in table 1.4.The segment is terminated by reaching a specified velocity, time, altitude, or downrange distance. FLOPS uses the information generated here in calculating community noise

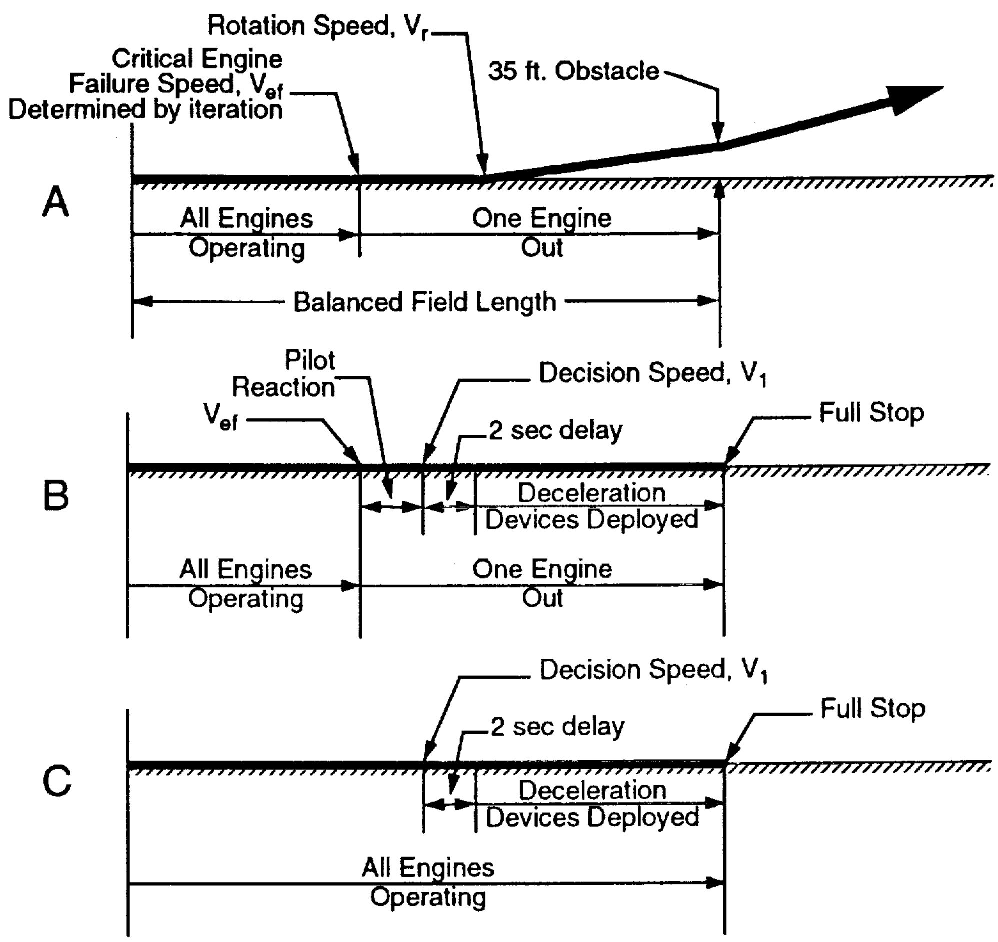


Figure 1.4 - Takeoff balanced field length.

Segment	Definition
ROTATE	Segment ends at start of rotation
LIFTOFF	Segment ends at liftoff
OBSTACLE	Segment ends at obstacle height
CHANGE	Segment ends to change parameters
CUTBACK	No distance segment to define required thrust setting for the next segment
LAST	Final segment to end climbout profile calculations

Table 1.3 - Climbout Segment Definitions

footprints.

Fixed thrust and constant velocity
Fixed thrust and constant flight path angle
Fixed thrust and fixed angle of attack
Fixed velocity and constant flight path angle
Fixed thrust and fixed cabin floor angle

Table 1.4 - Constraints for Climb Segments after the Obstacle

Community and Airport Noise

The takeoff and landing noise module is based on the FOOTPR program (ref. 15). Noise data can be generated at a matrix of user specified locations for use in generating takeoff and landing noise *footprints* or at the FAA certification points, takeoff, sideline, and approach. Noise sources include fan, primary and secondary jets, combustor, turbine, and airframe. Propagation of the noise sources to the observers includes the effects of atmospheric attenuation, ground attenuation and reflections, and shielding.

Cost

The cost analysis module, primarily developed for analysis of subsonic commercial transports, is described in detail in reference 16. The module combines a variety of cost models capable of predicting airframe research, development, testing, and evaluation (RDT&E, ref. 17), airframe production (ref. 18), engine RTD&E and production (ref. 19), and direct (ref. 20) and indirect (ref. 21) operating costs. These results are combined to produce life cycle cost and return on investment.

II. Cycle Analysis

The cycle analysis capability is based on the QNEP program described in reference 11. QNEP, for Quick NEP, is a simplified version of the Navy Engine Performance Computer program (NEPCOMP, ref. 9). QNEP was modified for FLOPS to improve reliability and precision and to reduce execution time. Because FLOPS is a preliminary analysis and design tool being used by a group with varied experience (many with only a cursory understanding of the propulsion system) other modifications were considered necessary, not only to prevent abuses of the new cycle analysis capability but also to make it as easy to use as possible. One such modification is the addition of a built-in control system which can be used to limit inlet exit (compressor inlet) corrected flow ($w\sqrt{\theta}/\delta$), compressor surge margin, compressor pressure ratio, compressor discharge temperature and/or pressure, and shaft horsepower for turboprops. The first three limits are used to keep the rotating components operating on the maps and the last three are there primarily to prevent component stresses and temperatures from exceeding realistic material limits. This chapter will begin with a brief overview of the QNEP engine cycle analysis program, followed by further details regarding modifications made in preparation for integrating the program into FLOPS. The engine cycles modeled in FLOPS will be described and comparisons of performance data with industry data and data generated using the Navy/NASA Engine Program (NNEP) will be presented. The chapter will conclude with a description of the interface between FLOPS and the cycle analysis module. The appendix of this paper is an excerpt from the FLOPS users guide. It contains further details on the cycle analysis module's input and operation.

Quick NEP Overview

QNEP uses one dimensional steady-state thermodynamic cycle analysis to predict design point and off-design performance for a variety of cycles. The engine cycle is defined by the logical connection of engine components (table 2.1) and off-design operation is governed by control components. Each of the flow-through components have one primary inlet and one primary exit flow station number which is used to define the flow path for the cycle. Heat exchangers have secondary (stream releasing heat) inlet and exit stations and secondary inlet and exit flows are permitted in duct components. Splitter and mixer components have secondary exit and inlet flow stations, respectively, and compressor and turbine components may have secondary exit and inlet stations, respectively. Compressors, turbines, and loads are connected using shaft components. Each shaft component can connect up to four components. Figure 2.1 illustrates one way in which these components might be connected. In addition to the configuration data that defines the basic cycle, there are up to fifteen design parameters (i.e. mass flow rates, component efficiencies, horsepower extraction, rotational

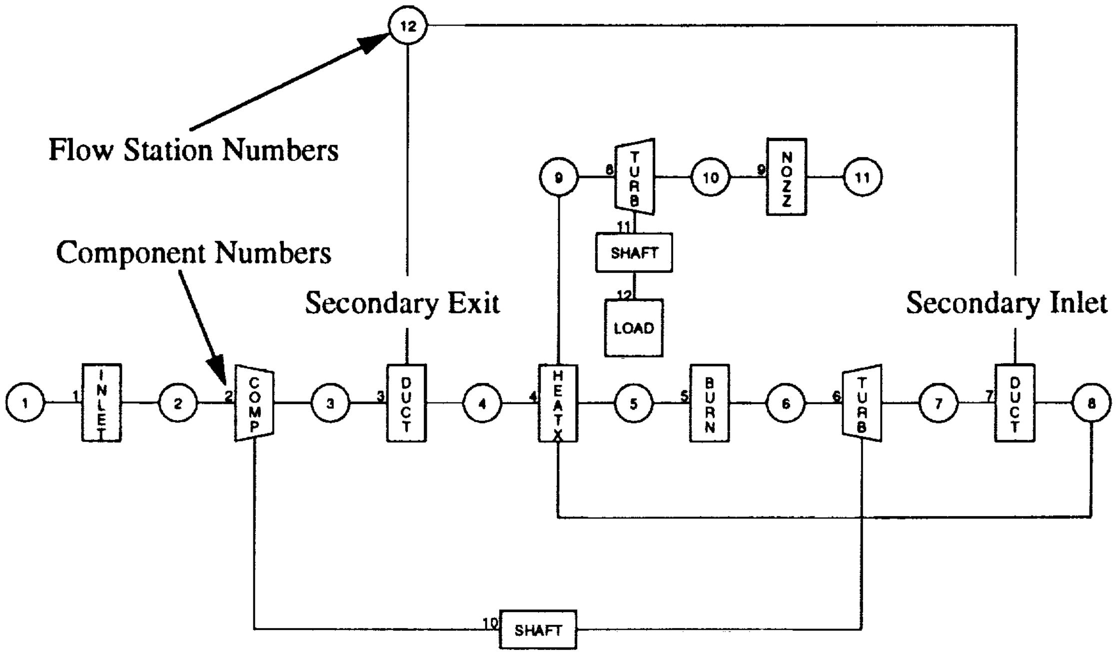


Figure 2.1 - Cycle schematic for a regenerative turboshaft.

speeds, pressure drops, and much more) for each of the components. The type of data depends on the component and is described in detail in reference 11. At the design point, flow-through components are sized and mass and energy are automatically conserved.

Off-design calculations require control components to ensure conservation of mass and energy. Control components do nothing more than link independent variables (component input quantities which can be varied) with the dependent variables (mass flow and energy imbalances) or errors. Controls are required to maintain continuity at the flow stations upstream of all compressors, turbines, and fixed throat area nozzles. Controls are required for all shafts to balance power, and mixer components use a control to match the static pressures of the primary and secondary streams. The mixer analysis is the simplified method of reference 11 and not the more thorough analysis that simultaneously solves the equations conserving mass, momentum, and energy as in reference 10. In the analysis, the static pressures of the two incoming streams are forced to be equal by use of a control component. During the solution process, the mixer exit static pressure is assumed to be the average of the two incoming streams. The solution is reached when the static pressures in the three streams are equal. Results from the program described in reference 10 suggest that the simplified method is adequate for the kinds of cycles considered for this work (figure 2.2). The mixer model will not model cases where one of the streams is supersonic. Heat exchanger components require a control for both design point and off-design performance to maintain conservation of energy. For example, if a fixed temperature rise is required for a given design, the user can allow the heat exchanger effectiveness to vary or possibly the ratio of the primary and secondary mass flow rates. Otherwise (and generally

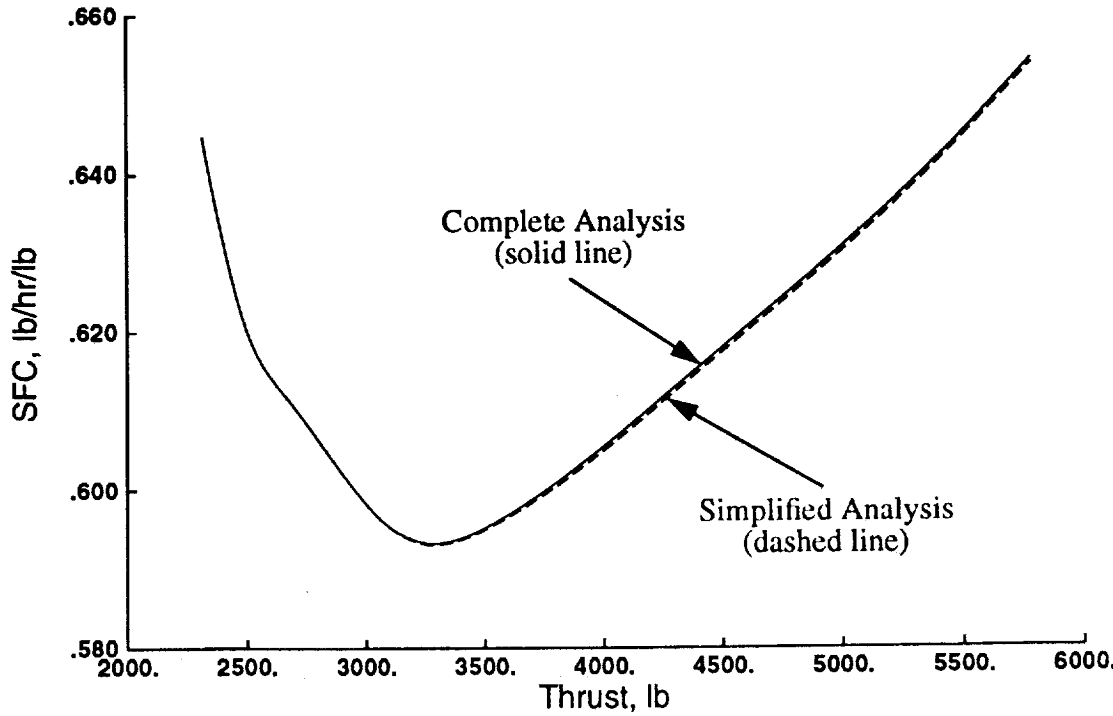


Figure 2.2 - Effect of Mixer Analysis Method for Subsonic Mixing Streams.

Component Type
Inlet
Duct, Burner, or Afterburner
Compressor or fan
Turbine
Heat Exchanger
Flow Splitter
Flow Mixer
Nozzle
Load
Shaft
Control

Table 2.1 - Table of Component Types

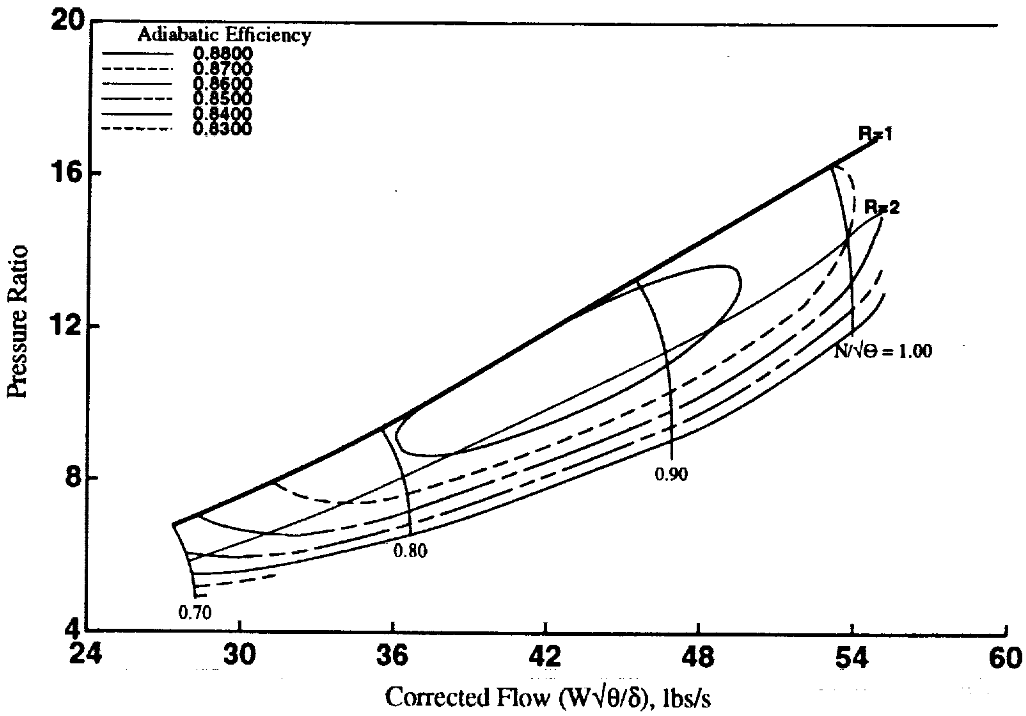


Figure 2.3 - Typical Compressor Map.

during off-design operation) the temperature rise itself can vary to maintain conservation of energy. Reference 9 provides a description of the overall solution procedure as well as detailed equations and methodology used for analysis of the individual components. Further details on control components and their use, as well as examples for a variety of cycles, can be found in references 9, 10, and 11.

In addition to controls, maps are used extensively to describe the off-design operating characteristics of many of the components. At the design point, scale factors are determined for each of the maps being used and these factors are then used to scale the data during off-design operation. Among the most important of these are compressor maps (figure 2.3). To properly interpolate on compressor maps, an artificial coordinate must be defined. Arbitrary lines, called R lines (two are shown), are drawn starting with and near parallel to the surge line (R=1 for the map shown) and must not intersect. A given R value, combined with a corrected rotational speed ($N/\sqrt{\theta}$), defines the compressor operating characteristics (pressure ratio, weight flow, and adiabatic efficiency). Other maps can be used to define turbine operating characteristics, inlet recovery and flow schedules, inlet and nozzle drag, duct and burner pressure drops, burner efficiency, and more. All component maps that may be used in the program and their formats are described in reference 11.

Quick NEP Modifications

The cycle analysis module in FLOPS is a modified version of QNEP. Besides the addition of the control system introduced at the beginning of this chapter, the rest of the changes are fairly minor and are only mentioned because they are not documented in reference 11. One modification, made to simplify access to inlet component data, requires that the first component (numerically) be the inlet. Table A-3 in the appendix lists the component input data that differ from the definitions defined in reference 11. Other input variables have been added to the engine cycle definition input data (namelist D) to allow simulation of the turbine bypass engine (TBE) and limited flexibility in the control system. These variables are defined in Table A-4 in the appendix. One of the more important component data variables, compressor design point adiabatic efficiency, is generally directly affected by the pressure ratio. To account for this, an empirical correlation, relating compressor adiabatic efficiency to pressure ratio and technology level, was incorporated into the cycle analysis module (figure 2.4).

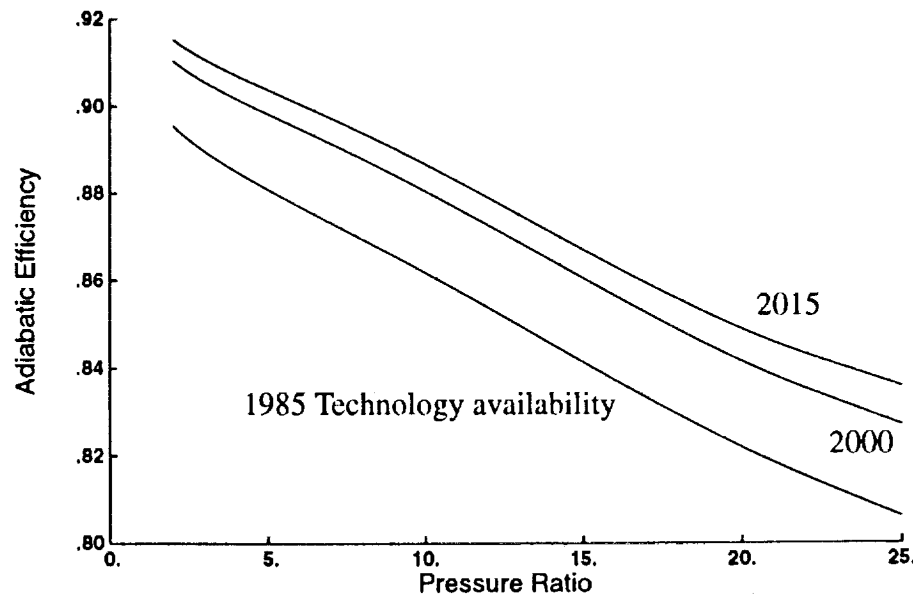


Figure 2.4 - Effect of technology availability and pressure ratio on compressor adiabatic efficiency.

A primary objective of this work is to demonstrate the capability to optimize engine cycle design variables along with aircraft design variables. Because of this, considerable attention was focused on reducing execution time and improving reliability without sacrificing accuracy. Significant reductions in execution time (about a factor of 20) were realized by linearizing the table interpolation scheme, by reducing the number of part power points calculated, and by improving various iterative calculations. The original table interpolation scheme used a modified version of the cubic spline method of reference 22 and accounted for nearly 60% of the total execution time. The modification had no significant impact on the accuracy of the results. The capability to fill in approximate part power data, given at least one complete set of part power data, had already been a part of FLOPS. Test cases indicated no significant impact on the results (less than 1% in magnitude and negligible impact on trends), provided that at least one full set of part power data was calculated for each Mach number (figure 2.5). Several iterative calculations were improved by making a more intelligent initial guess based on previously computed values, by using a higher order estimate from successive

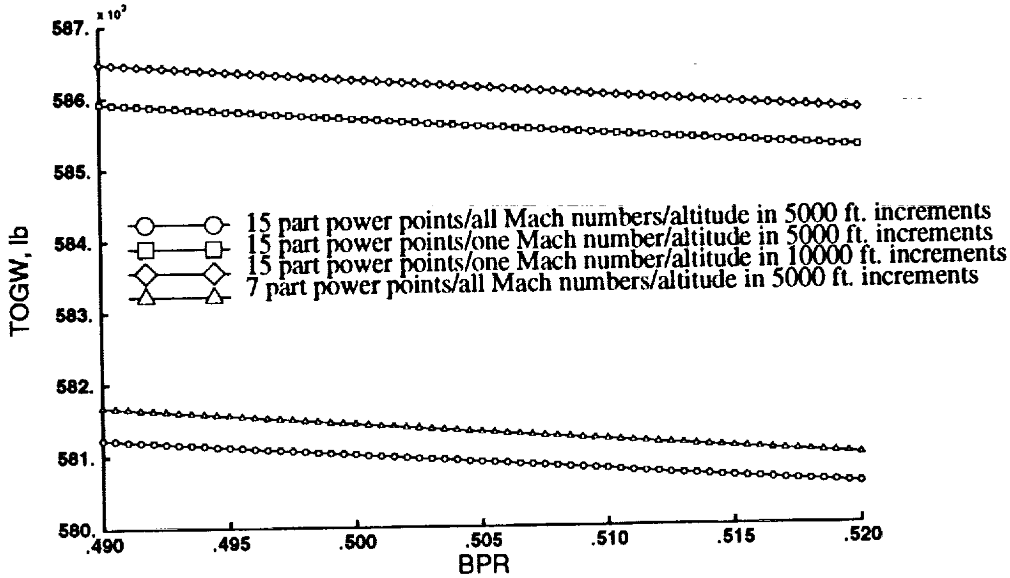


Figure 2.5 - Effect of Quantity of Engine Cycle Performance Data on Aircraft Takeoff Gross Weight.

approximations, and by instituting improved exception handling.

Once the cycle analysis module was integrated and preliminary optimization test cases were run, it became apparent that precision and fidelity were as important as speed. These effects were minimized in the cycle analysis module by lowering the default convergence criteria for off-design control errors from .001 to .000001. This low tolerance made it extremely difficult for the analysis to converge without compiling the program in double precision. Because problems with precision were also affected by choice of input and the variety of analyses in FLOPS, these aspects of the problem will be discussed further in chapter V.

Cycle Analysis Validation

Performance data, thrust and specific fuel consumption for an advanced subsonic separate flow turbofan and an advanced supersonic turbine bypass engine were computed using the cycle analysis module. For the subsonic test case, basic design point data, pressure ratios, burner temperatures, and thrust were known from industry sources and detailed data, such as compressor and turbine performance maps and efficiencies, burner efficiency, turbine cooling flow, and duct pressure drops had to be estimated. Data from the cycle analysis module are compared with industry data in figure 2.6. The objective in this case was to match industry data, so even though the default compressor and turbine maps were used the remaining detailed data were adjusted to achieve that match. For the Mach number 2.4 turbine bypass engine, data generated using the cycle analysis module are compared with data generated using NNEP (figure 2.7). In this case all the detailed data, including compressor and turbine maps, were available. The data shown are installed data and include both inlet and nozzle losses. The data generated by NNEP used the detailed procedure of reference 23 for

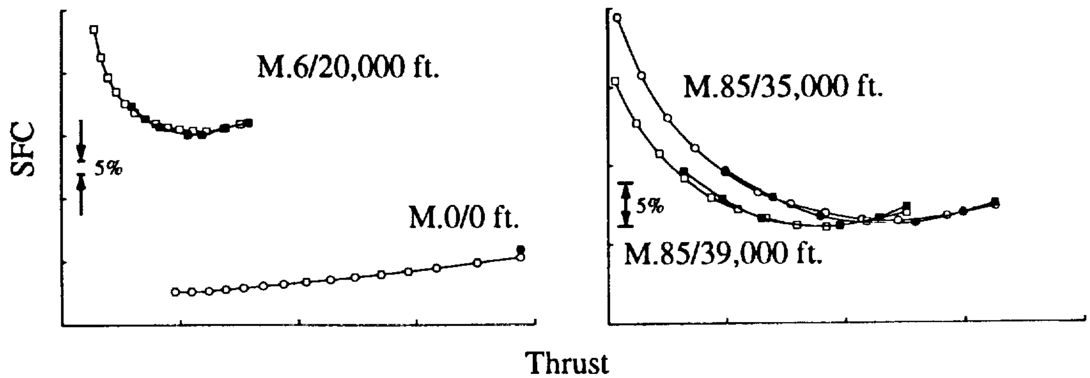


Figure 2.6 - Comparison of performance data from the cycle analysis module (open symbols) with industry data (solid symbols).

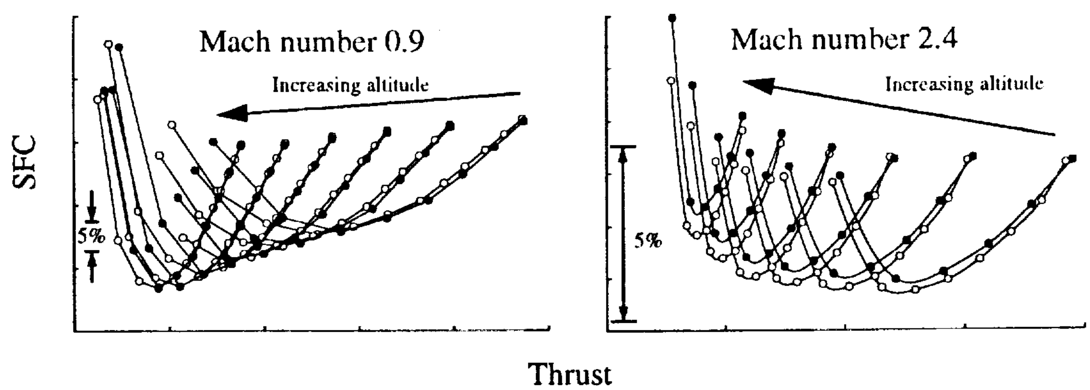


Figure 2.7 - Comparison of installed performance data from the cycle analysis module (open symbols) with data generated using NNEP (solid symbols).

predicting installation effects, while cycle analysis module used the simplified methods that are built into the program. While accurate agreement of results with industry data and well established methods is certainly important, it is equally important for this work that the method produce those results precisely, quickly, and reliably. To generate a full set of data, like that in figure 2.7, for Mach numbers 0 to 2.4 with a 2/10 increment, takes the cycle analysis module 77.1 seconds of computer time. Generating the same data using NNEP would require several hours of combined computer time and hands on restarts.

Integration of the Cycle Analysis Module

In integrating the cycle analysis module into FLOPS, emphasis was placed on the easy to use philosophy that went into the development of FLOPS since a majority of users will not be propulsion system specialists. In order to keep the quantity of user input to a minimum, a set of five files with different engine cycle definitions and component input data were prepared. In addition to the five default files, a user defined engine cycle definition data file may be used. A set of user input variables (see appendix) was selected which can be input along with the regular FLOPS namelist input file that will override some of the component data contained in the default files. Some of these variables were selected for their importance, as in the fundamental cycle design variables, and others such as power extraction and customer bleed were selected because they may depend directly on the aircraft size and type. The first time the cycle analysis module is accessed by FLOPS, the engine cycle definition file and a file containing the component maps are read into memory. Design point parameters from the engine cycle definition file are overridden by any data input with the FLOPS input data file and design point performance is predicted. Depending on user input, propulsion system weight may be predicted and off-design performance is computed over a range of Mach numbers, altitudes, and power settings as determined by user input. The integration of the cycle analysis module into FLOPS is illustrated schematically in figure 2.8. During optimization or parametric variation, performance weight and

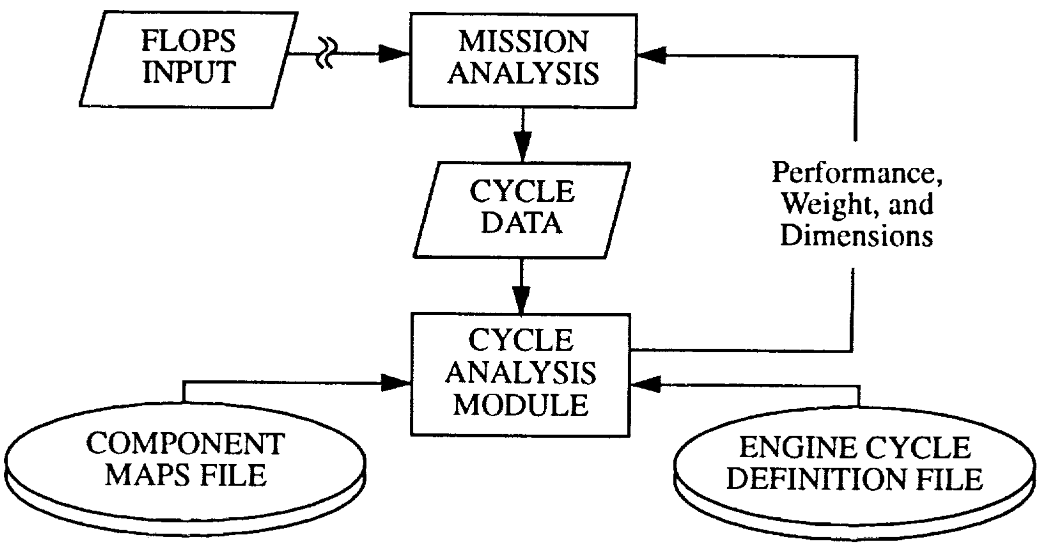


Figure 2.8 - Engine Cycle Analysis Module Schematic.

dimensions are updated whenever an engine cycle design variable changes. Generally, if only engine thrust changes, the performance data is not regenerated and propulsion system weight and dimensions are scaled by FLOPS.

III. Propulsion System Weight Analysis

The available procedures for predicting propulsion system weight and dimensions are often limited in scope, outdated, or are too difficult and require too much input for the casual user. While correlations are especially useful in optimization and preliminary analyses, they are typically developed from a data base of existing engines (ref. 24), and are therefore generally not adequate for advanced studies and especially for supersonic applications. Another procedure based on physical principles incorporates correlations for predicting the number of compressor and turbine stages based on cycle design parameters and thermodynamic data (ref. 25) and provides adequate weight and dimension trends to be useful in preliminary design studies. However, the method is limited to current technology subsonic turbofan engines and probably is not adequate for the very high bypass ratio turbofans currently being developed. Another method, based on preliminary mechanical design (ref. 26), was developed specifically for use with NNEP and adequately predicts weight and dimensions (±10%) for a suitably wide variety of cycles. Because of the similarities between NNEP and QNEP, this computer program would have been relatively easy to implement. However, the level of detailed input required to get the necessary level of accuracy would probably overwhelm the average FLOPS user. Therefore, a new capability for predicting propulsion system weight was developed for FLOPS.

The analysis uses the thermodynamic cycle data generated in the cycle analysis module and preliminary component design to estimate weight for current and advanced technology engines. The philosophy that went into the development of the prediction procedure, dictated by the FLOPS user community, required the amount of detailed input be kept to a minimum, while still maintaining a high level of realism and accuracy. Propulsion system weight is inherently a discontinuous function of nearly all the primary design variables, because the materials change as temperature or stress levels vary and the number of compressor and turbine stages can vary with pressure ratio, rotational speed, and allowable stress levels, among other parameters. Additionally, in a localized region of the design space, it is probably sufficient to scale a baseline engine weight with airflow (thrust level) or to use analytical expressions developed from a detailed analysis. For these reasons, it would be impractical and unrealistic to try to maintain the level of precision that was required in the cycle analysis, for the weight analysis.

Semi-empirical procedures to estimate preliminary weights for the entire propulsion system have been developed. A key element of the prediction procedure is a data base of material properties (density and usable stress as a function of temperature). Although it encompasses only four materials at present, aluminum 2124 alloy, titanium 6242 alloy, RENE 80 (a nickel based superalloy), and steel (ref. 27), the subroutine can be modified as data for additional materials are needed or become available. The material type is selected based on temperature and an optional minimum usable stress.

For example, at 300 °F aluminum is usable but not very strong. Thus if the usable stress is less than minimum requested value, data for the next material (titanium) are returned. The weight and dimensions for each component in the engine itself are predicted individually, but with consideration for adjoining components or components connected through a shaft. Though an integrated approach, where the entire bare engine is optimized for minimum weight would have been possible, it probably would not be practical since the level of detail, both in design variables and constraints, that goes into the current method is only a small fraction of what is required for a complete analysis. A computerized system for analysis and optimization of the entire propulsion system for minimum fuel consumption, weight, and cost is a challenge that is addressed in reference 8. The remainder of this chapter will describe methods used for predicting weight and dimensions for both nonrotating and rotating components in more detail with the results compared to industry data.

Nonrotating Components

The weight for all nonrotating components (except turbine and compressor stators) are based on surface area and weight per unit area (weighting factors). The weighting factor is based on the usable stress of the material and the tangential stress or *hoop stress* (σ_t) of a thin walled vessel (ref. 28):

$$\sigma_t = \frac{P \cdot r}{t}$$

The pressure (P) is based on the pressure at the sea level static design point with a correction applied to account for any increase in pressure at the maximum cruise Mach number. The radius (r) is the maximum radius of the duct and is based on inner and outer radii of the rotating components or flow areas as determined in the cycle analysis module. The material type, its density, and usable stress are based on temperature. The weighting factor is the material thickness (t) multiplied by the material density. Weight for additional components, such as burner manifold and nozzles and compressor and turbine frames, are based on the correlations in reference 26.

Inlets and Nozzles

The basic geometry for the inlet is based on the cruise Mach number, an input variable to set the inlet type (pitot, external compression, or mixed compression), and a curve fit of the geometries defined in reference 29. Nozzle geometry is based on the flow areas as determined through cycle analysis at the cruise Mach number. The weighting factors for inlets and nozzles are determined as above with additional factors applied depending on the degree of variable geometry that is used. Additionally, geometry and weighting factors for the inlet and nozzle may be input by the user.

The procedure for estimating inlet and especially nozzle weights is probably adequate for scaling purposes, provided that the weighting factors can be determined from a more detailed design. The procedure does provide a good external geometry that is reflected in the total vehicle aerodynamics. Calculations for two supersonic inlets are shown in figure 3.1. The weight and dimensions were all calculated using the default geometry and weighting factors and are compared with the study results of reference 30. Because of the variety of nozzle types, thrust reversing techniques, acoustic liners

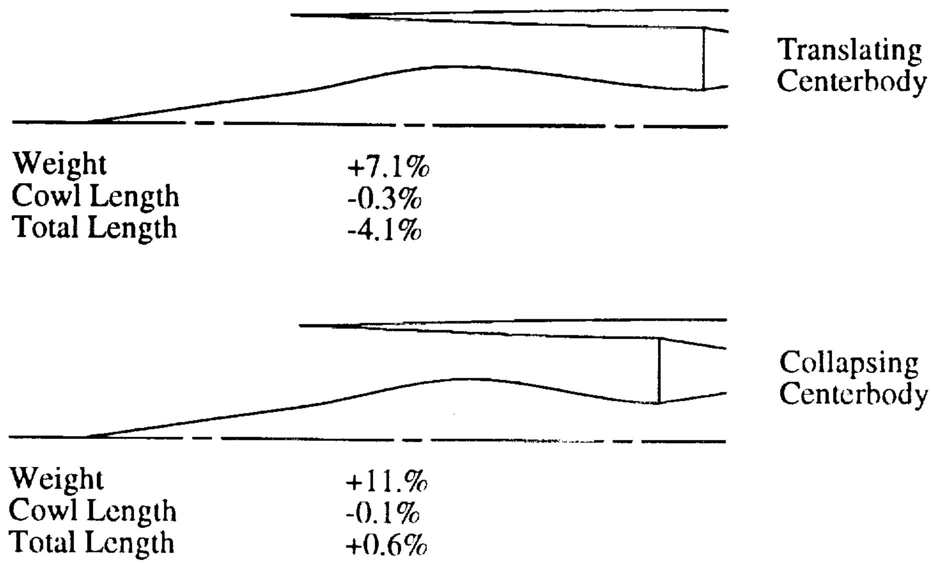


Figure 3.1 - Weight and dimension predictions for two Mach number 2.2 supersonic axisymmetric mixed compression inlets.

and other noise suppression devices currently being considered for supersonic cruise vehicles, nozzle weight and dimensions vary widely with the cycle design. Therefore, the nozzle weighting factor and length to diameter ratio is input, based on more detailed designs for the cycle being analyzed. As will be shown below, total propulsion system weight and dimensions for current and advanced subsonic engines agree well with industry data.

Rotating Components

Though the procedure used for predicting the weight and dimensions for rotating machinery is far more involved than the predictions for the nonrotating components, it is still a simplified analysis. The method, though adequate for the purpose of estimating weight, could not be used for predicting actual performance characteristics or detailed geometry. A detailed description of the aerodynamic and mechanical design considerations used in predicting numbers of stages and disk dimensions can be found in reference 27. This information, combined with some of the empirical correlations for estimating blade volumes, number of blades, and casing and hardware weights of reference 26 provide realistic weights and dimensions for the overall engine.

Compressors

Compressor aerodynamic analysis uses a repeating stage, repeating row, mean-line[1] analysis (ref. 27). The repeating stage criteria requires that the velocity vector at the rotor entrance be the same for each stage in the compressor ($\mathcal{V}_3$=$\mathcal{V}_1$ in figure 3.2). The repeating row criteria requires that the stator airfoils be a mirror image of the rotor airfoils (β_2=α_1 and β_1=α_2). With these simplifications, the velocity vector diagram for all stages in the compressor is shown in figure 3.2.

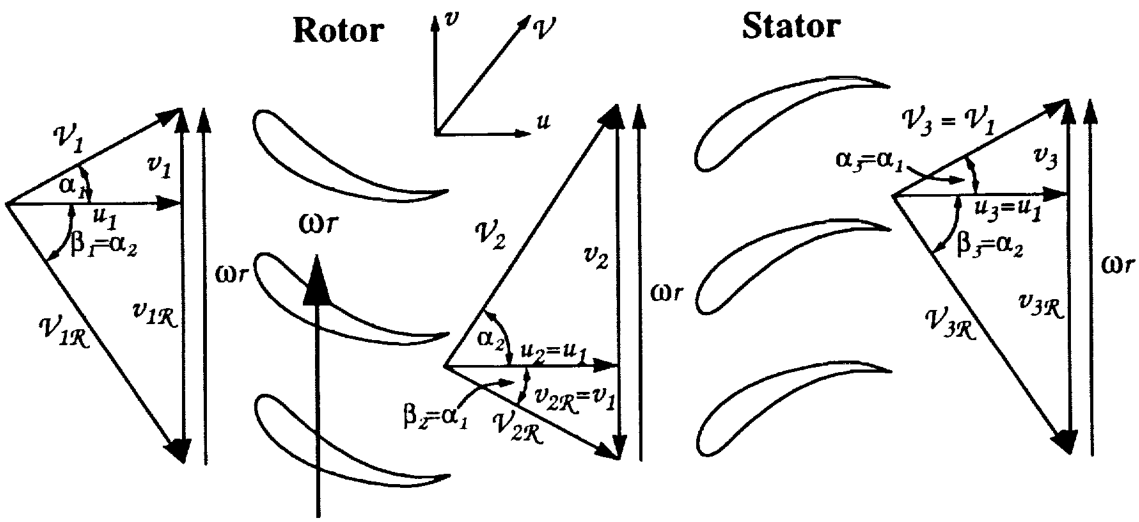


Figure 3.2 - Velocity diagram for repeating stage, repeating row compressor.

Another critical parameter used in the analysis is the diffusion factor (D). It is a parameter that characterizes the amount of deceleration (and associated static pressure rise) experienced by the flow over the airfoil upper surfaces. For the repeating stage, repeating row analysis it is defined as:

$$D \equiv 1. - \frac{V_3}{V_2} + \frac{|v_2 - v_3|}{2 \cdot \sigma \cdot V_2}$$

where σ or solidity is defined as the ratio of the airfoil chord length divided by the airfoil spacing. For typical compressor designs, as D increases the total pressure losses through the stage also increases, and these losses increase substantially for values of D much above 0.6. However, as D decreases the number of compressor stages must increase to achieve a desired pressure ratio. In a procedure developed for FLOPS, D is assumed to be a function of technology level. As technology level increases, improvements in airfoil design should allow higher diffusion factors (and hence fewer stages) without large total pressure losses. Another parameter, the compressor polytropic efficiency, is also characterized as a function of technology level in FLOPS, or it may be user defined. These two

1. One dimensional analysis of the flow at the average area.

parameters and an assumed solidity of one, combined with the mass flow and thermodynamic parameters from cycle analysis and the repeating stage, repeating row methodology, define the thermodynamic and geometric characteristics throughout the entire compressor for any given inlet Mach number and stator entrance flow angle, α_2. For the purposes of estimating compressor weight in FLOPS, an inlet Mach number of 0.5 is assumed and an initial value for compressor tip speed and hub to tip ratio are user defined for the first compressor component in the flow. The stator entrance flow angle is then allowed to vary to minimize the total compressor weight or some combination of weight and diameter for compressor components with one or two stages. The optimization scheme (ref. 31) uses the sequence of unconstrained minimization technique with a modified Kreisselmeier-Steinhauser function to handle constraints. For the initial stator entrance flow angle and throughout the iteration, a rotor inlet flow angle is calculated, thereby establishing the velocity vector diagram for however many stages may be required. Once this is established, stages are added one at a time until the overall pressure ratio meets or exceeds the required pressure ratio. For each stage along the way, a maximum total temperature is computed from which material, density, and usable stress are established. The number of rotor and stator blades and their volumes, estimated from the flow area and an empirical relationship for blade aspect ratio, are multiplied by the material density to give the weight for the blades. Dimensions, volume, and weight of the rotor rim and disk are then estimated, based on the assumption that the rim width is 4/10 of the stage length and it is 7/10 as high as it is wide, that the airfoils are not tapered, and that the airfoil centrifugal stress is evenly distributed on the hub. The nomenclature for both compressor and turbine rotors is illustrated in figure 3.3. The equations used in the analysis are derived in reference 27. Once the first compressor in the flow has been analyzed, any compressor component that is connected by a shaft is analyzed, followed by any connected turbine components. The hub to tip radius ratios for all compressor components downstream of the initial compressor are selected by the program such that any transition duct length is minimal. The rotational speed for all components connected to the first compressor on any given shaft is established by the analysis of the first compressor. For each compressor stage analyzed, there are several parameters that are checked and action is taken to correct any problems that may arise. These are listed in table 3.1. It should be noted that the action taken is one choice of many and that

Possible problem	Action taken
RPM > 15,000 rev/min	Increase hub to tip radius ratio
AN^2 > Input maximum Blade tensile stress > usable stress Wheel speed > realistic limit[a] Rotor disk width at the shaft > stage length	Reduce tip speed (and adjust hub to tip radius ratio if rotational speed has been fixed).
Blade height < 0.5 in	Adjust hub to tip radius ratio (and tip speed if rotational speed exceeds 15,000 rev/min)

Table 3.1 - Limits considered in the prediction of compressor weight.

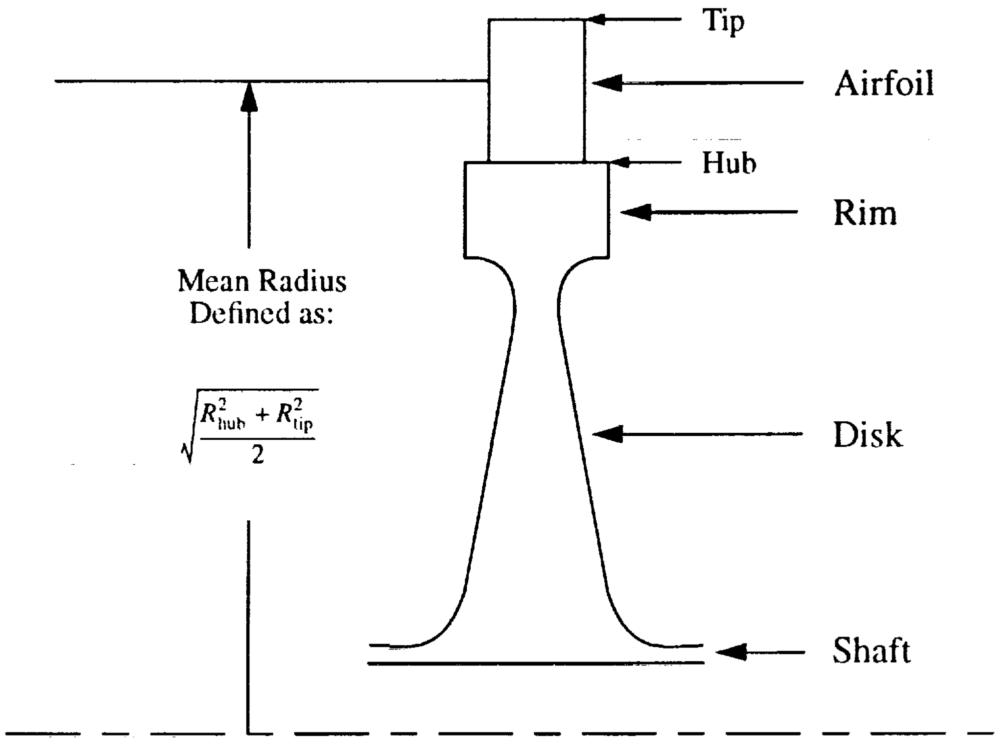


Figure 3.3 - Compressor and turbine rotor nomenclature.

a. The realistic limit depends on material density and usable stress and is used to prevent the disk thickness from getting unrealistically large.

any action taken here will impact the analysis of any connected components. A large reduction in rotational speed will severely impact the turbine design and for large high bypass ratio turbofans, a geared fan may be required. A truly integrated approach would account for this and consider, among other things, a higher strength material rather than a reduction in rotational speed.

Turbines

The basic procedure for predicting turbine weight is similar to the compressor analysis described above and again is based on the methods described in reference 27. However, since the number of turbine stages is typically far less than the number of compressor stages, the analysis for turbine components considers more design parameters. Also, although current advanced turbines include rotor-only stages and counter-rotating turbines, the method used here assumes that each stage consists of a stator followed by a rotor. Owing to the high cost of turbine airfoils (especially cooled airfoils), relative to compressor airfoils, first the number of stages and then weight is minimized. Given the inlet and exit conditions, an analysis is performed, assuming a single stage turbine. In the analysis the design variables are varied to minimize the total turbine weight. If a solution that meets the necessary constraints is found, the analysis is complete. If no solution is found, another stage is added and same procedure is repeated. The design variables are the rotor inlet Mach number, the rotor exit flow angle and the mean radius. Each of the design variables are subject to reasonable side constraints. The three behavioral constraints for each stage are: a user defined maximum allowable AN^2, a maximum allowable rotor exit relative Mach number (established internally as a function of technology level), and a maximum tip radius (established by rotating components already sized). Typically the stator exit Mach number is greater than one and the rotor exit relative Mach number is less than one to ensure that the choking in the stator controls the turbine mass flow rate. The material and usable stress at each stage are selected in the same manner as for compressor components. However, to account for turbine cooling effects, the minimum requested usable stress is used even though the available stress, based on the total temperature of the flow, may be less.

The same optimization scheme that is used in the compressor analysis is used here. However, unlike the compressor analyses' single design variable, there are two design variables per stage plus the mean radius and three side constraints per stage plus the maximum allowable tip radius for the turbine analysis. The optimization scheme is highly sensitive to the initial guess, the design vector scale factors, and the constraint values. Because of the variation in performance of the optimization scheme from one engine cycle to another, considerable effort was required to determine what scale factors and side constraint values worked fairly well for the variety of cycles of interest. In one case, increasing the maximum radius constraint by 10% produced a turbine that had a smaller maximum radius, 2 additional stages and 30% more weight than the original. Generally however, small variations in other miscellaneous optimization parameters, with the constraint values and scale factors set as they are, produced only a ±2-3% variation in total weight for a fairly wide variety of cycles.

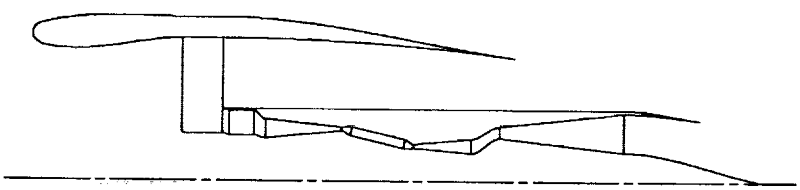
Additional Considerations

A more integrated approach, where all the rotating machinery connected by a shaft was optimized, was attempted with only limited success. Execution time was excessive and results were erratic. The erratic results in this approach, and possibly in the current analysis for turbines, is probably due to discontinuities caused by material changes and changes not influenced by the optimizer, such as those listed in table 3.1. Another approach, that should alleviate some of these problems, would be to sub-optimize a single stage at a time using the exit conditions as input for the next stage. The primary optimizer would then control such things as tip speed, mean radii, and constraints for all components connected by a shaft. This approach should reduce the burden on the optimization scheme, but may have an adverse impact on the design of stages downstream. Because results from the current method are considered adequate, this approach was not attempted.

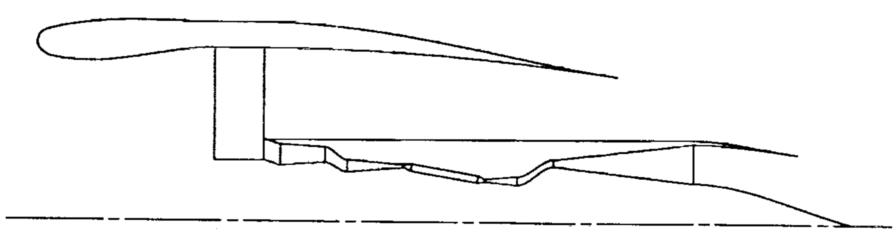
In spite of the extremely limited number of design variables, real world constraints, and materials that are considered, the method developed predicts weights and dimensions for a variety of cycles with sufficient accuracy for the kinds of preliminary analyses that are generally conducted with FLOPS. A few of the real world constraints that are not considered directly are, airfoil stresses due to bending moments, vibration, or foreign object damage; aeroelastic effects or flutter; temperature gradients; thermal and torsional stresses; fatigue and corrosion; and cost. Indirectly, they are considered in the selection of a usable stress that is significantly below the ultimate tensile strength for the material and the generalization that cost and weight are directly proportional. The method is limited to axial flow turbomachinery, current and near term advanced technology power plants, and does not predict propeller, heat exchanger, or flow inverting valve weight and dimensions.

Validation

Preliminary engine weights were predicted for a current technology advanced subsonic transport engine, a 1995 engine in service (EIS) advanced subsonic transport engine, and four supersonic transport study engines. These six cycles were selected because cycle design and weight data were available. The two subsonic transport engines are relatively high bypass ratio, two spool (two shafts), separate flow turbofans. Available data and predictions include nacelle weight and dimensions and results relative to industry data are summarized in figure 3.4. Weight and dimensions predicted for the



Weight	+1.2%	
Maximum Diameter	+6.7%	1987 EIS
Total Length	+7.8%	

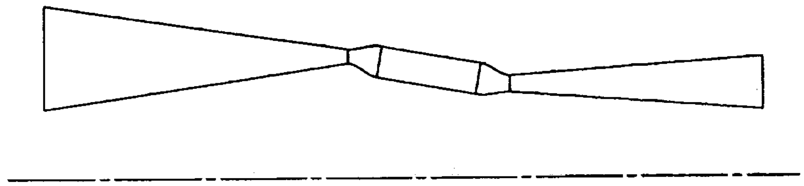


Weight	-0.5%	
Maximum Diameter	-6.9%	1995 EIS
Total Length	+1.7%	

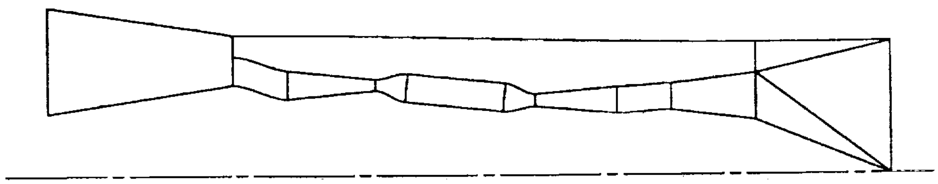
Figure 3.4 - Weight and dimension predictions for current and advanced technology subsonic transport engines.

four supersonic transport cycles are summarized relative to industry data in figure 4.5. Each of the cycles are designed for Mach number 2.4 cruise. The first is a turbine bypass engine[1], and the remaining three are mixed flow turbofans with bypass ratios ranging from 0.4 to 1.1.

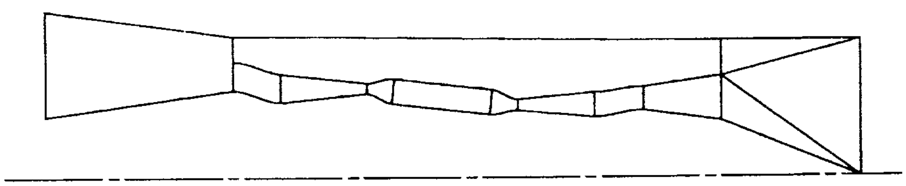
1. The industry prediction for this engine included the weight of engine controls and accessories. For comparison with the method developed a penalty of 10% was assumed.



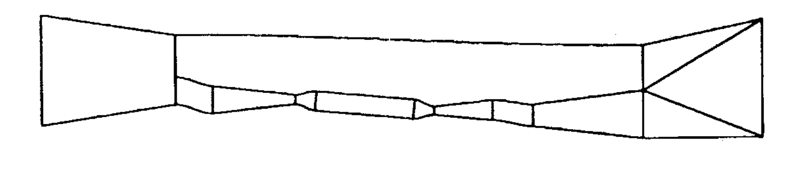
Weight	-0.0%	2005 EIS TBE
Maximum Diameter	-1.3%	
Total Length	+12.%	



Weight	-6.8%	2005 EIS 0.40 BPR MFTF
Maximum Diameter	-2.5%	
Total Length	-0.5%	



Weight	-0.2%	2005 EIS 0.68 BPR MFTF
Maximum Diameter	-3.8%	
Total Length	+1.2%	



Weight	-9.0%	2005 EIS 1.13 BPR MFTF
Maximum Diameter	+0.7%	
Total Length	-5.0%	

Figure 3.5 - Weight and dimension predictions for advanced technology supersonic transport engines.

IV. Optimization

It has been found throughout the course of this work that optimization involving a multitude of disciplines, with each discipline often having some degree of suboptimization, is difficult. Generally when an optimization scheme is presented, its utility is demonstrated using problems with algebraic expressions for objectives and constraints (ref. 31, 32, and 33). Though the design of the experiment method used in reference 7 could be used with the system that has been developed, it cannot capture all the complex interactions between the design variables and constraints that have been observed in this work. With the new cycle analysis capability, the design variables that may be considered during optimization in FLOPS are listed in table 4.1, the constraints are listed in table 4.2, and the objective function has the form:

$$\begin{aligned} OBJ = {} & W_1 \cdot \text{Weight} + W_2 \cdot \text{Fuel} + W_3 \cdot \text{Mach}(L/D) + W_4 \cdot \text{Range} + \\ & W_5 \cdot \text{Cost} + W_6 \cdot \text{NOx} + W_7 \cdot \text{SFC} \end{aligned}$$

where W_i are user input weighting factors. Specific fuel consumption (SFC) is used if an engine cycle is to be optimized (SFC is minimized) for an input cruise Mach number and altitude independent of any aircraft concept.

With the cycle analysis module integrated into FLOPS, initial optimization test cases were run, and contour plots were generated. The results were not promising. A large step size (10%) for calculating gradients was required to approach what appeared to be a minimum gross weight and the contour plots suggested that the integrated system of analyses was producing erratic results. This was confirmed with one dimensional parametric variations. Figure 4.1 shows the improvement that was made in behavior of the results over time. Finding the causes for the erratic results and finding a remedy was a difficult problem. Discontinuities were found to be caused by selection of input, problems with the analyses methodology, errors in the analyses, tolerances used in suboptimization, and even the amount of propulsion system performance data generated. Parametric variations were performed for all the major design variables in FLOPS to ensure that there were no unexplained discontinuities in aircraft takeoff gross weight. Tolerances throughout the program were adjusted to minimize discontinuities in the analyses. If there is an error in the analysis, FLOPS will find it and exploit it.

Input

During optimization in FLOPS, the selection of input can have a major impact on optimization

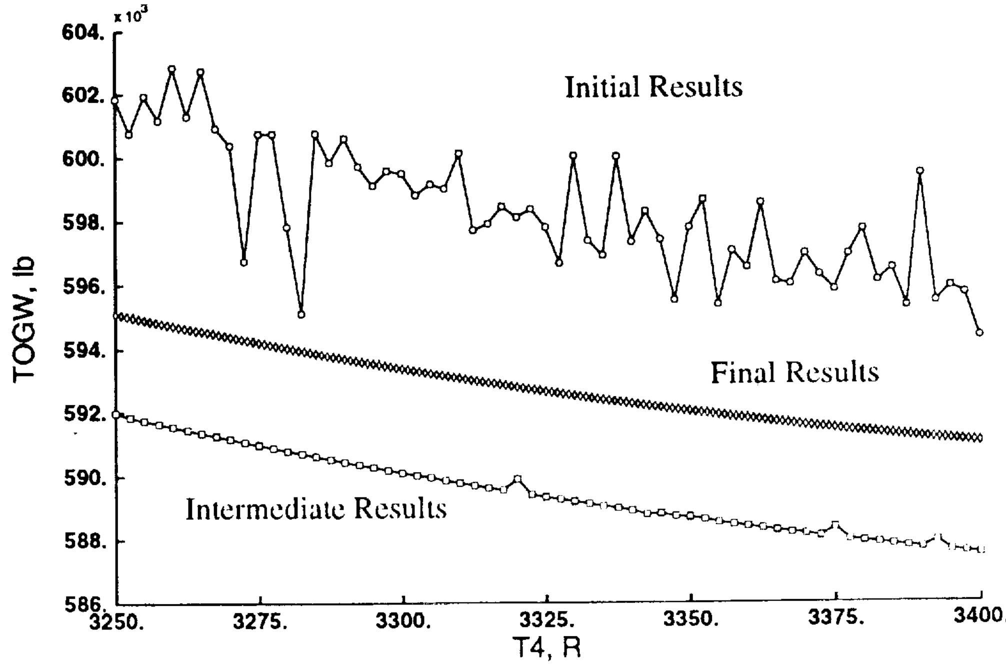


Figure 4.1 - Improvement in continuity of aircraft takeoff gross weight.

results. While its impact on the total aircraft takeoff gross weight is minimal, the selection of the appropriate climb profile can be critical during optimization (fig. 4.2). Any input options that will

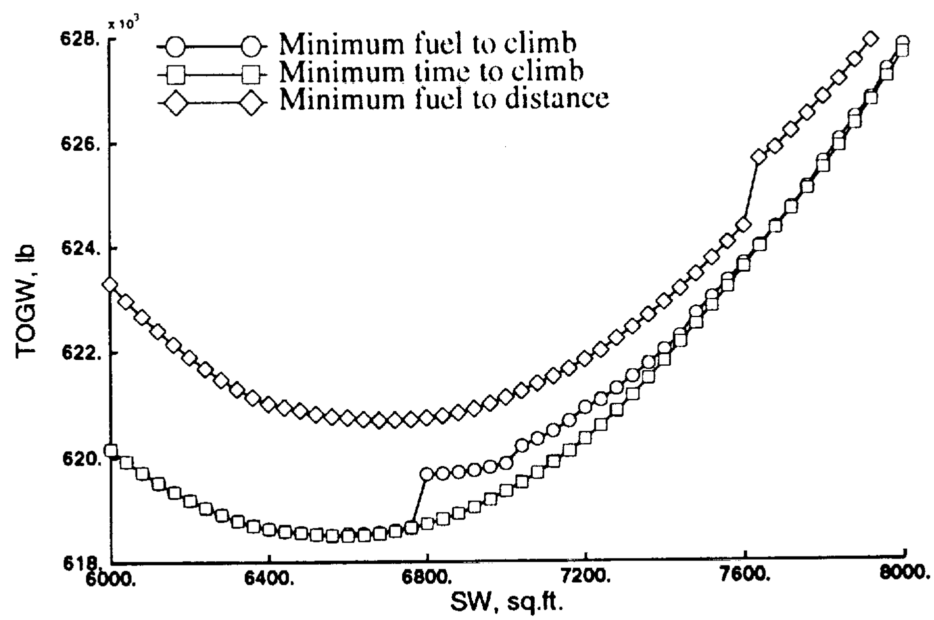


Figure 4.2 - Effect of climb profile option on continuity of supersonic aircraft takeoff gross weight versus wing area for a Mach number 2.4 transport.

Aircraft ramp weight
Wing aspect ratio
Maximum rated thrust per engine
Wing area
Taper ratio of the wing
Quarter-chord sweep angle of the wing
Wing thickness-chord ratio
Cruise Mach number
Maximum cruise altitude
Engine design point burner temperature
Engine overall pressure ratio
Engine fan pressure ratio
Engine bypass ratio
Engine throttle ratio[a]

Table 4.1 - List of Aircraft and Engine Cycle Design Variables

a. Defined as the maximum burner temperature divided by the design burner temperature.

cause discontinuities, such as allowing an afterburner during climb if needed (or part power settings if possible for minimum fuel climb profiles), should be avoided during optimization. Instead such a case should be run with the afterburner on at all times during climb. Since the fuel burned during climb is generally a small fraction of the total fuel burned, the impact on the engine cycle selection should be minimal. Once the engine cycle has been established, the aircraft may be resized through further analysis. In this example the aircraft is flying through the transonic region. In such a problem the minimum fuel to distance optimization in FLOPS actually results in a heavier aircraft because for climb optimization, FLOPS pursues a local rather than global minimum and the time (and distance) taken to traverse the transonic region is large.

Selection of the optimization algorithm, the step size used for computing gradients, the maximum allowable step size used in a one-dimensional search, and the initial design can be difficult. Generally the quasi-second order methods produced the lowest value of the objective function provided that the initial design is reasonable; otherwise, any of the methods could produce the lowest value for the objective function. Though the univariate search algorithm is least likely to fail to converge and generally provided the lowest value for the objective function when the quasi-second order methods

Lower limit on range
Upper limit on approach speed
Upper limit on takeoff field length
Upper limit on landing field length
Lower limit on missed approach climb gradient thrust
Lower limit on second segment climb gradient thrust
Upper limit on internal fuel
Upper limit on compressor discharge temperature
Upper limit on compressor discharge pressure
Upper limit on maximum sea level static jet velocity
Lower limit on maximum sea level static specific thrust
Upper limit bypass area / core area for MFTF's
Upper limit on nitrous oxide emissions

Table 4.2 - List of Aircraft and Engine Cycle Constraints

fail, the conjugate gradient and steepest descent algorithms have produced lowest value for the objective in some cases. It is recommended that at least two algorithms be used for the first run of a given case, and depending on computer resources and time, all 5 algorithms are recommended. In a set of fifteen test cases run using three aircraft, seven different design missions, and two different engine types, the Broyden-Fletcher-Goldfarb-Shano algorithm produced the lowest value for the objective in 5 out of 12 cases. In another set of 10 runs (five 5-variable and five 6-variable), the number of analyses required for a solution to be found for each of the methods was averaged for each of the five runs. The results of both studies are summarized in table 4.3. The objective in all twelve of the cases was to maximize range for a fixed aircraft takeoff gross weight. Though the resulting ranges for each of the five algorithms were all within at least 3.6% of each other in all twelve cases, the engine cycle design variables varied by as much as 19%.

The size of the increment used for each of the design variables when gradients are computed may be introduced as absolute increments or as a fraction of the initial design variable. Generally a finite difference step size of 0.1% of the initial design variables works well. In some of the mixed flow turbofan cases using a slightly smaller step size (0.05%) on the design burner temperature and overall pressure ratio produced marginally better overall results (<0.5%) with considerable improvement in the consistency in the trends of the design variables for a series of cases. A small maximum allowable step size for one-dimensional searches tends to prevent the case from failing and generally requires

	Algorithm				
	Davidon-Fletcher-Powell	Broyden-Fletcher-Goldfarb-Shano	Conjugate Gradient (Polak-Ribiere)	Steepest Descent	Univariate Search
Number of times the lowest value of the objective was found.	1	5	3	0	3
Number of times the program failed to converge.	2	1	0	1	1
Average of the number of analyses required for convergence (4 design variable cases / 5 design variable cases).	81 ----- 190	80 ----- 243	88 ----- 156	92 ----- 169	195 ----- 216

Table 4.3 - Performance comparison of optimization algorithms.

more analyses. A large maximum allowable step size may lead to unrealistic designs by overshooting the target by a considerable margin. Failed cases can result when part-power points in the cycle analysis module fail to converge. This can occur when an unrealistic cycle (i.e. high bypass ratio, high fan pressure ratio, and low burner temperature) is being generated.

Validation

Considerable effort went into confirming that the optimization procedures were, in fact locating the minimum aircraft takeoff gross weight. To accomplish this, a series of 16 test cases were run using an advanced technology Mach number 2.4 high-speed civil transport concept with advanced technology mixed flow turbofans having bypass ratios in the range of 0.1 to 2.0. To keep the problem manageable, only five of the fourteen possible design variables were used (table 4.4). The aircraft characteristics and mission are summarized in table 4.5. Each of the cases was run with the same initial guess for wing area, thrust[1], burner temperature, and overall pressure ratio. There were two initial guesses for fan pressure ratio, one for the higher bypass ratios and one for the lower bypass ratios. Since the maximum allowable burner temperature was fixed, variations in design burner temperature are essentially variations in throttle ratio. Based on analyses using the detailed method for predicting propulsion system weight described in chapter IV, an analytical expression for

1. At the higher bypass ratios, the initial guess for thrust and wing area had to be increased manually to get a valid initial design.

Maximum rated thrust per engine
Wing area
Engine design point burner temperature
Engine overall pressure ratio
Engine fan pressure ratio

Table 4.4 - List of Aircraft and Engine Cycle Constraints Considered for Optimization Validation

• Mach 2.4 supersonic transport
• 5,500 n.mi. range (25% subsonic)
• 250 passengers
• Mixed Flow Turbofan Compressor Discharge Temperature Limit of 1710 °R Maximum Burner Temperature of 3560 °R Constraints
• Approach Speed upper limit of 140 kts. Takeoff and Landing Field length of 10,300 ft. Available fuel volume 500 fpm rate of climb capability Takeoff and Landing noise are not considered

Table 4.5 - Aircraft Configuration and Mission Characteristics

propulsion system weight as a function of bypass ratio and airflow was developed. This analytical expression was used for these optimization test cases. Since bypass ratio was not a variable for these cases, the thrust level (proportional to airflow) is the only design variable that will affect the propulsion system weight. The rationale for this study was that if the trends in aircraft takeoff gross weight and the design variables versus bypass ratio were reasonable, then an optimal or nearly optimal design had been found for all cases that followed that trend. If a design did not follow the trend, then a local optimum may have been found. For all cases the Broyden-Fletcher-Goldfarb-Shano method was used twice, with the second run using the results of the first. If the design variables or the objective function changed with respect to the results of the previous run by more than 1%, then that case was rerun. The results of that study are shown in figure 4.3. There was some difficulty with the 1.8 and 2.0 bypass ratio cases in that the aircraft encountered difficulty during climb. The 0.4 and 1.2 bypass ratio cases were evaluated in more detail. The design variables were varied ±5% from the optimum value. The results shown in figures 4.4 and 4.5 indicate that a minimum aircraft takeoff gross weight was located, subject to takeoff field length and approach speed constraints. Also shown in the figures are compressor discharge temperature trends. Compressor discharge temperature (CDT)

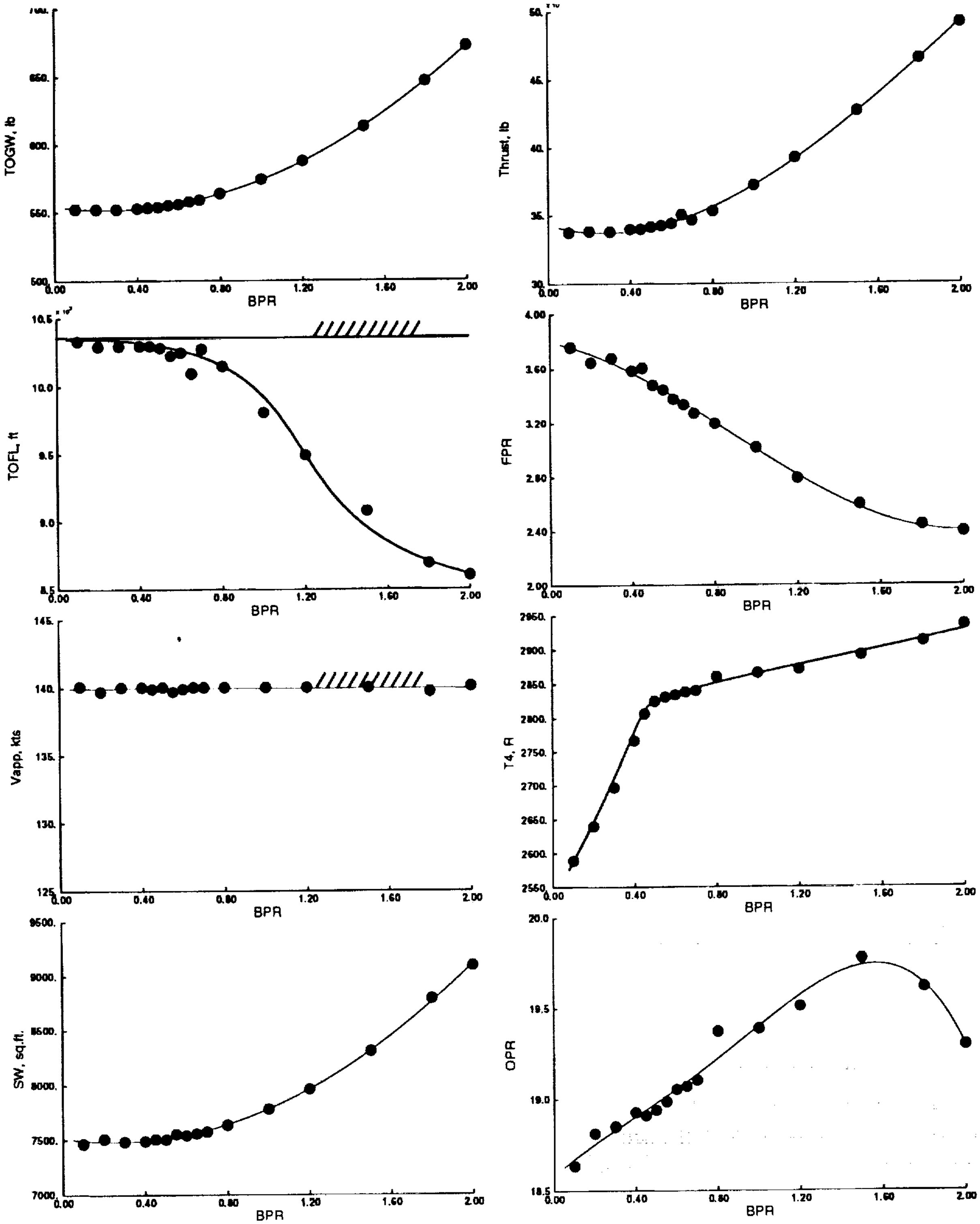


Figure 4.3 - Results for 16 five-variable optimization test cases.

is limited by engine control, not the optimizer. This is why the limit is never exceeded as overall pressure ratio increases or as design burner temperature decreases.

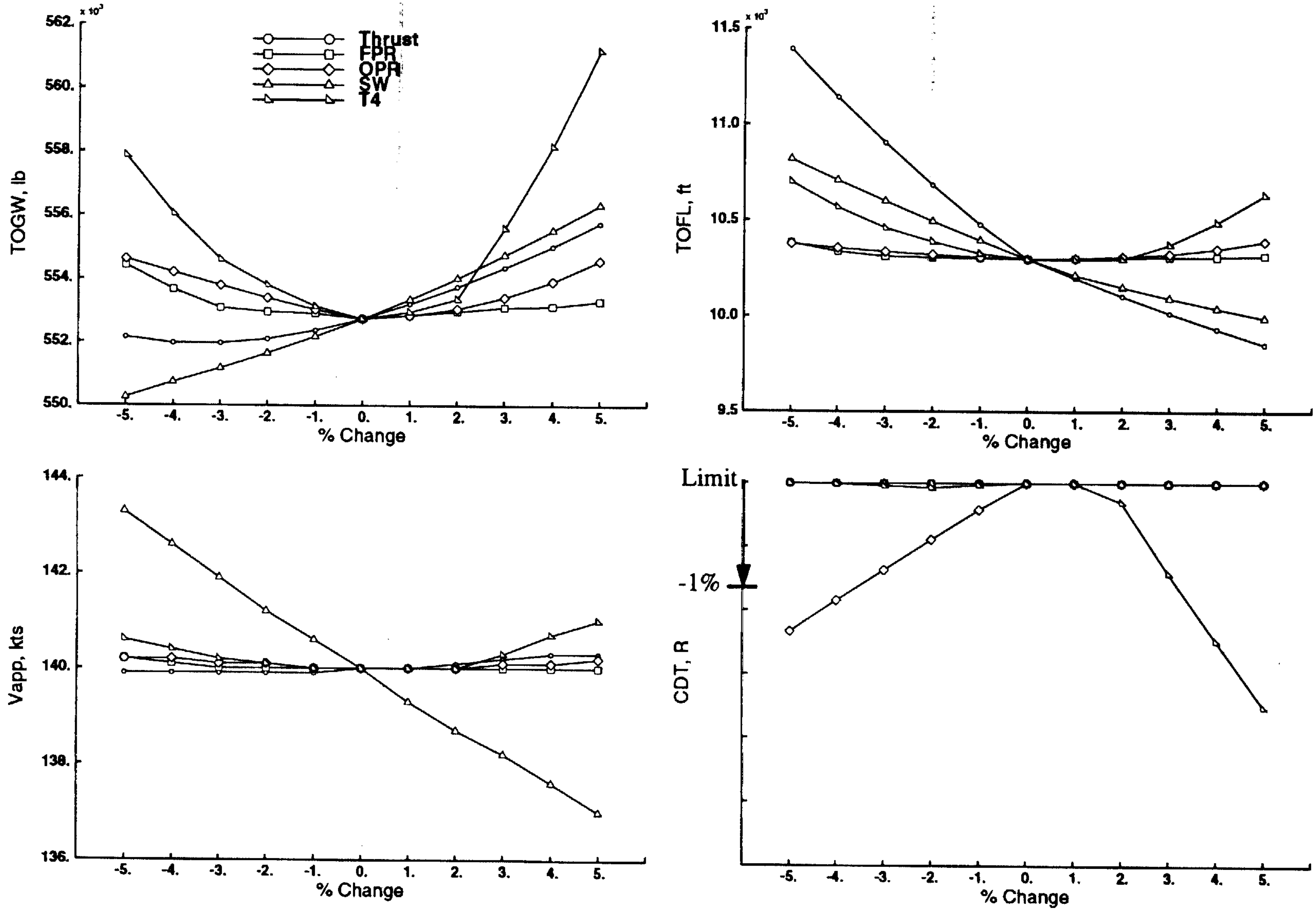


Figure 4.4 - Sensitivity of selected parameters to the design variables for the 0.4 bypass ratio mixed flow turbofan optimization test case.

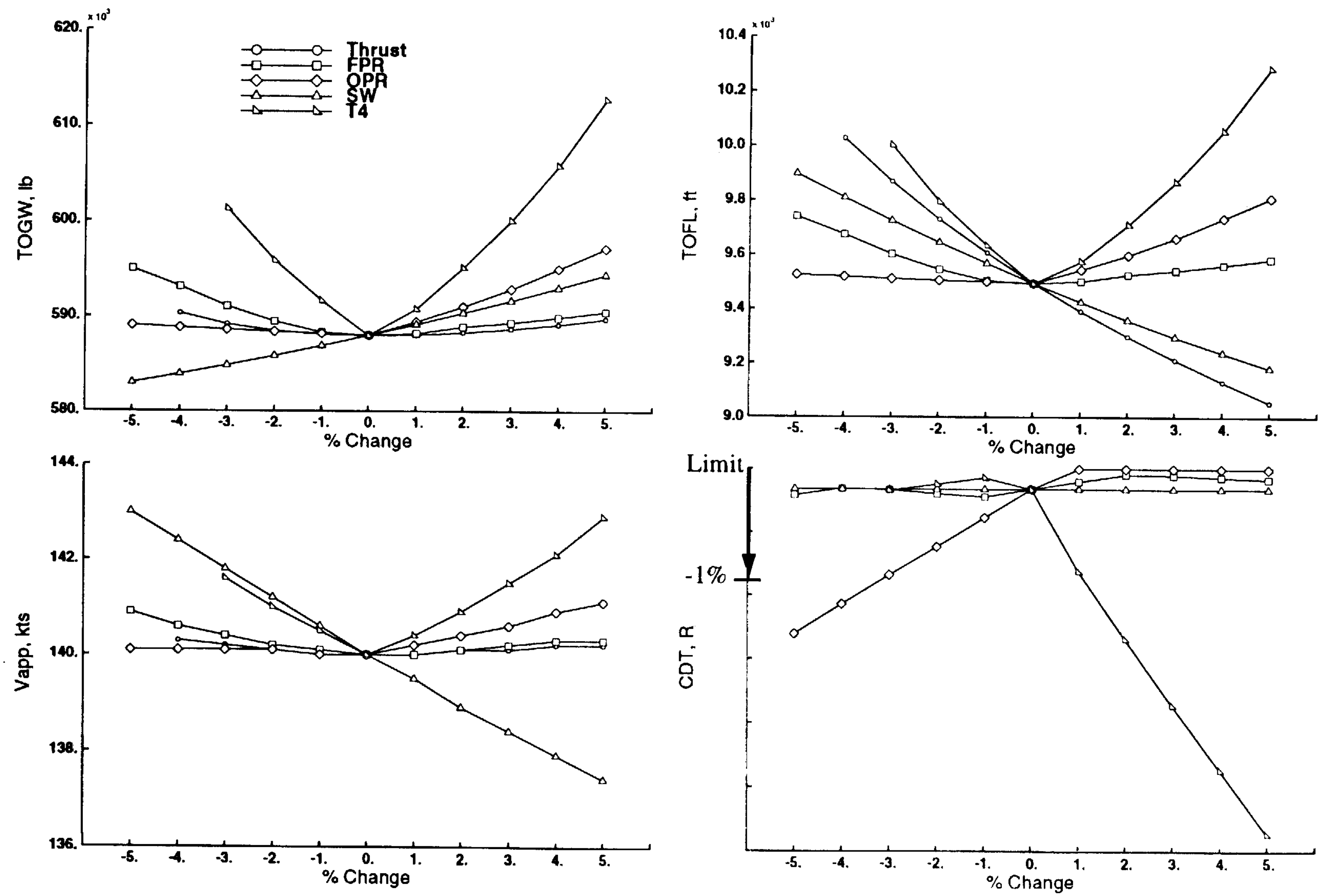


Figure 4.5 - Sensitivity of selected parameters to the design variables for the 1.2 bypass ratio mixed flow turbofan optimization test case.

V. Application and Results

The purpose of this chapter is to illustrate the application of this work to current research in the area of high-speed civil transport (HSCT) aircraft. The competing demands on both the propulsion system and the airframe for high thrust-to-weight and low noise is one of the program's biggest challenges. There are currently at least five different propulsion system concepts that are being considered. Early in the program the mixed flow turbofan was not considered a viable candidate. The primary candidates were the high specific thrust turbine bypass engine with a mixer ejector nozzle that entrains large quantities of ambient air at takeoff, variable cycle engines, and valved engines that would convert from low specific thrust for takeoff to high specific thrust for supersonic cruise. More recently, due to weight, complexity, or performance penalties associated with many of these cycles or noise suppression devices, the mixed flow turbofan is being reconsidered. Each concept has numerous design variables and constraints. The cycle analysis module is capable of modeling two of the cycles that are being considered: the mixed flow turbofan and the turbine bypass engine.

Unlike a typical subsonic transport engine, the engine that powers any future HSCT will likely operate at or near its maximum operating condition throughout most of the mission. The compressor discharge temperature and turbine inlet temperatures will exceed the limits of any current commercial engine. Though turbine and compressor airfoil life is a concern, the temperature limits that are being used in HSCT cycle studies in industry and at NASA are considered achievable for a year 2005 entry into service. The impact of these two limits, for a mixed flow turbofan installed on a Mach 2.4 HSCT, will be assessed with FLOPS. In a second example, engine cycle optimization will be applied to a low sonic boom configuration that flies overland at Mach 1.6 and at Mach 2.0 overwater. Because the low boom configuration has been designed and shaped to produce an "acceptable" pressure signature at the ground, the only parameters that are allowed to vary are the engine thrust and engine cycle design variables. Comparisons will be made between this aircraft and an aircraft of the same takeoff gross weight that flies overland at Mach 0.9 and overwater at Mach 2.4. The impact of the engine cycle and the length of the overland segment on the maximum range will be presented for both of these aircraft.

Each of the segments in all mission analyses are optimized for the given engine cycle performance data. Climb segments are optimized for minimum time to climb. Both cruise segments (overland and overwater) are optimized for maximum specific range and descent is at optimum lift-drag ratio influenced by flight idle fuel flow. All the cycles used in these studies have 1 pound/second of customer bleed and 200 HP power extraction. The inlet pressure recovery and nozzle internal performance are the same for all cycles. Because variations in nacelle size can have a significant impact on total skin friction drag, the aircraft skin friction drag is predicted by FLOPS. The remaining aerodynamic characteristics (wave drag and drag due to lift) for all configurations used in these studies were estimated outside of FLOPS.

Propulsion System Weight

The propulsion system weight is an extremely important parameter. An additional 1000 lbs of propulsion system weight (less than 2%) translates into an additional 3000 lbs of additional aircraft takeoff gross weight. In comparison, variations as large as 10% in some of the engine cycle design variables change the aircraft takeoff gross weight very little. Due to the uncertainty in the propulsion system weight and the need for a continuous analytical function for optimization, simplified equations for inlet, engine, and nozzle weights were used for all analyses. The weight for the inlet is a function of airflow only (fig. 5.1). The nozzles are mixer ejector nozzles and the weight is a function of airflow and sea level static jet velocity (fig. 5.2). The increase in nozzle weight with increasing jet velocity is due to the increase in the amount of ambient air that must be entrained during takeoff to meet the FAR part 36, Stage 3 noise constraints. The engine weight is a function of airflow and bypass ratio (fig. 5.3). The total propulsion system weight also includes engine firewall, mounts, and controls. The simplified equations were developed from a limited database of industry predictions for engine cycles designed for 2005 EIS and a 2.4 cruise Mach number. For the case where the maximum cruise Mach number is 2.0, the overall pressure ratio is generally higher than for the Mach 2.4 cases, and the engine weight predictions for these cases may be optimistic. However, the inlet weight predictions are relatively conservative for the Mach 2.0 cases.

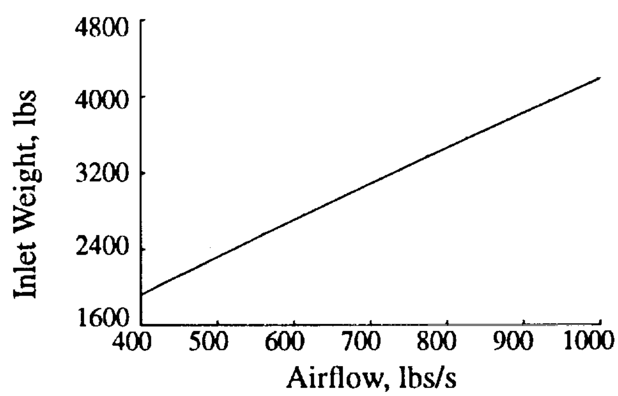


Figure 5.1 - Inlet weight versus airflow.

5000 n.mi. Mach 2.4 Baseline

Currently in the NASA High-Speed Research (HSR) program, the primary focus is on Mach 2.4 cruise aircraft capable of 5000 n.mi. range, of which approximately 25% is subsonic overland flight. Economic studies and technology availability estimates indicate Mach 2.4 to be the most promising cruise Mach number for a year 2005 entry-into-service (EIS) viable vehicle and it is presently the upper limit being studied in the HSR program (ref. 34). The aircraft and mission characteristics are summarized in table 5.1.

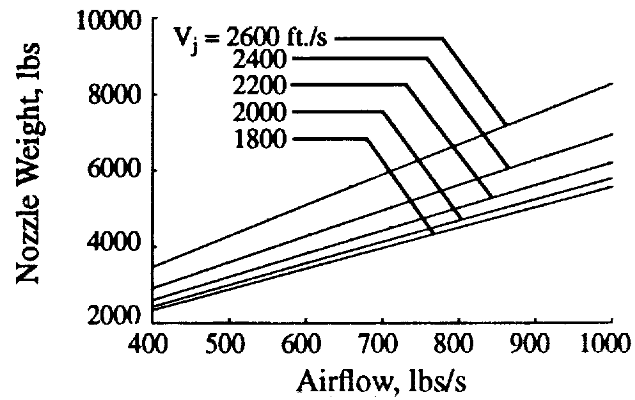


Figure 5.2 - Nozzle weight versus airflow and jet velocity.

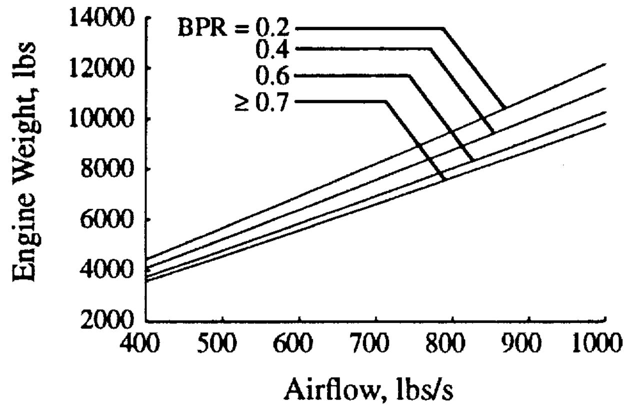


Figure 5.3 - Engine weight versus airflow and bypass ratio.

The six design variables that were considered for this aircraft concept are bypass ratio, burner temperature, fan pressure ratio, overall pressure ratio, takeoff thrust, and wing area. Thirteen 5-variable cases were run at discrete bypass ratios ranging from 0.1 to 1.2 and one case was run where all 6 design variables were allowed to vary. In all these cases the objective was to minimize the aircraft takeoff gross weight. Except for available fuel volume in the two highest bypass ratio cases (1.0 and 1.2), the takeoff field length constraint was the only active constraint in the final results. Aircraft takeoff gross weight versus bypass ratio is shown in figure 5.4 for the thirteen 5-variable optimization cases. All the design variables followed reasonable trends versus bypass ratio, except for considerable scatter in the design burner temperature and the overall pressure ratio. However, for all bypass ratios, the burner temperatures were all within 1.5% of each other and the overall pressure ratios were all within 4.5%. It should be noted here that, based on the assumptions for propulsion system weight used for these analyses, bypass ratios between 0.3 to 0.6 would warrant further study since they are all within approximately 3000 lbs of each other or within 2% of the total propulsion system weight.

Not surprisingly, the 6-variable optimization case resulted in an optimal bypass ratio near 0.45. Once the optimum had been found, several two-dimensional parametric variations were performed. Plots were generated with contours of aircraft takeoff gross weight, and constraint lines for the 11,000 ft. takeoff field length and available fuel volume. On each of the contour plots shown, the optimum as

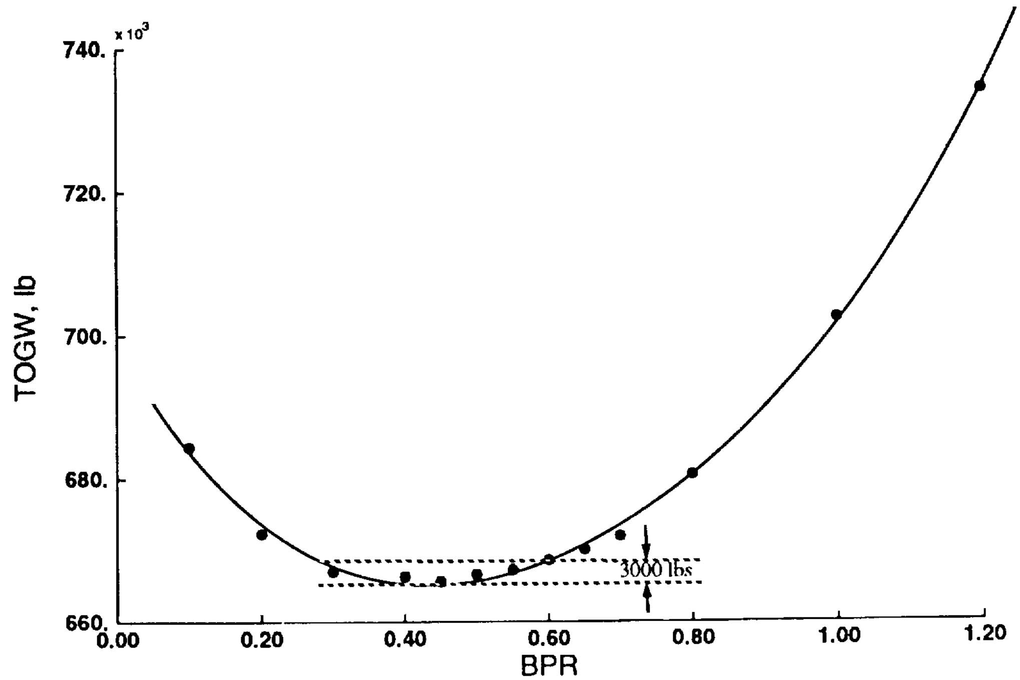


Figure 5.4 - Optimum aircraft takeoff gross weight versus bypass ratio.

• Mach 2.4 supersonic transport
• 5,000 n.mi. range (25% subsonic)
• 305 passengers
• Mixed Flow Turbofan Compressor Discharge Temperature Limit of 1710 °R Maximum Burner Temperature of 3560 °R
• FAR 36, Stage III Noise Nozzle suppression and weight increase with jet velocity Actual takeoff and sideline noise are not estimated
• Constraints Approach Speed upper limit of 160 kts. Takeoff and Landing Field length of 11,000 ft. Available fuel volume 500 fpm rate of climb capability

Table 5.1 - 5000 n.mi. Aircraft Configuration and Mission Characteristics

found by the optimizer in FLOPS is indicated on each figure. The first of these (fig. 5.5) is the type of contour plot that is typically used for sizing the wing area and engine size given a set of propulsion system performance characteristics. The next figure shows how variations in thrust and design burner

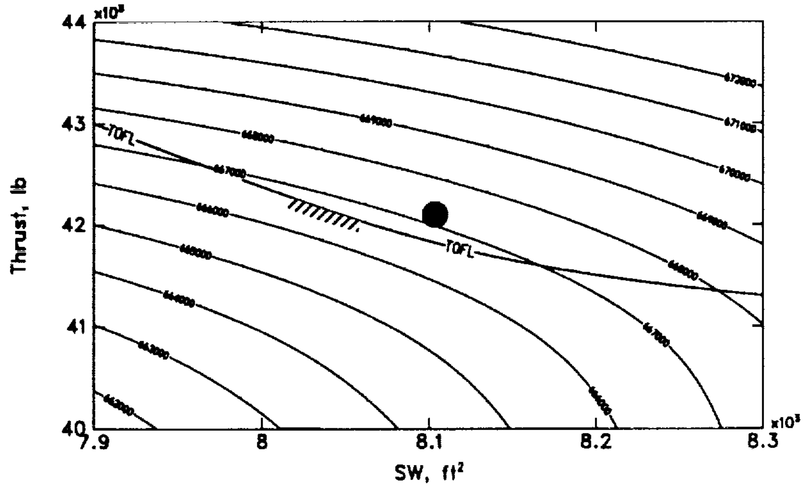


Figure 5.5 - Aircraft takeoff gross weight versus engine size and wing area.

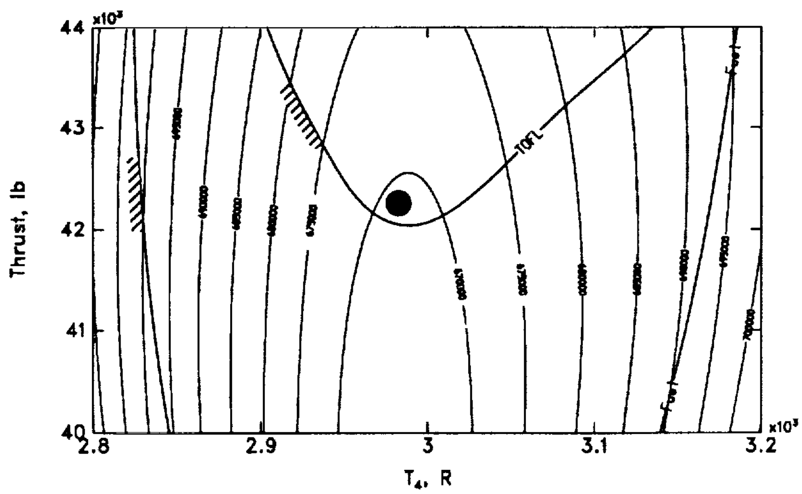


Figure 5.6 - Aircraft takeoff gross weight versus engine size and design burner temperature.

temperature affect the aircraft takeoff gross weight (fig. 5.6). Variations in design burner temperature are effectively variations in throttle ratio. As design burner temperature goes down, throttle ratio goes up. Below the optimum, the compressor discharge temperature is at its maximum value of 1710 °R, and performance is degraded since the cycle analysis module forces the engine to be throttled back so that the limit is not exceeded. For design burner temperatures above the optimum, performance is degraded because those engine cycles are not operating at the fullest potential allowed by technology. The next figure exhibits the same basic characteristics, though the penalty is not as severe (fig. 5.7). Here the compressor discharge temperature is near the limit at the optimal bypass ratio (where the bypass ratio is matched to the optimum fan pressure ratio) and at the limit for bypass ratios above and below the optimum. Figure 5.8 clearly shows an optimum combination of overall pressure ratio and design burner temperature. In this case the compressor discharge temperature is at its limit at a design burner temperature near 3000 °R for the high pressure ratios and as pressure ratio decreases, the

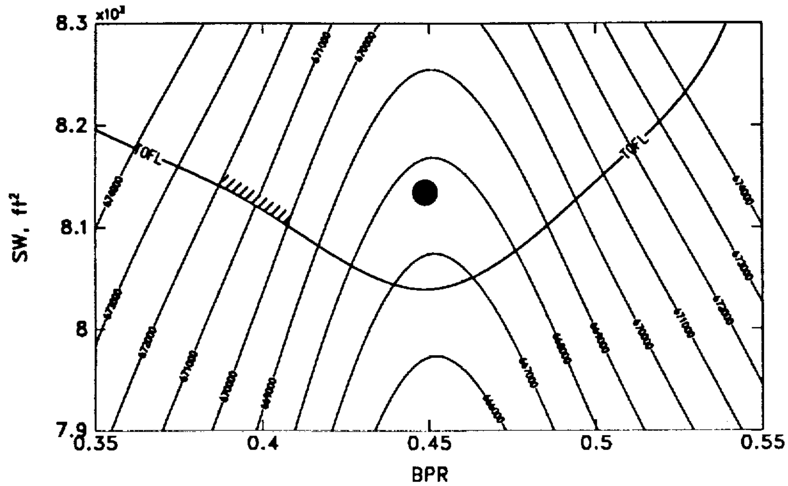


Figure 5.7 - Aircraft takeoff gross weight versus wing area and design bypass ratio.

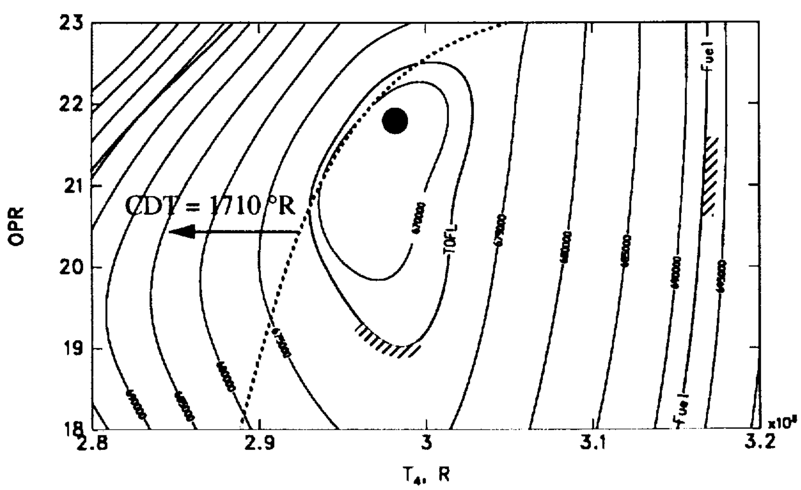


Figure 5.8 - Aircraft takeoff gross weight versus overall pressure ratio and design burner temperature.

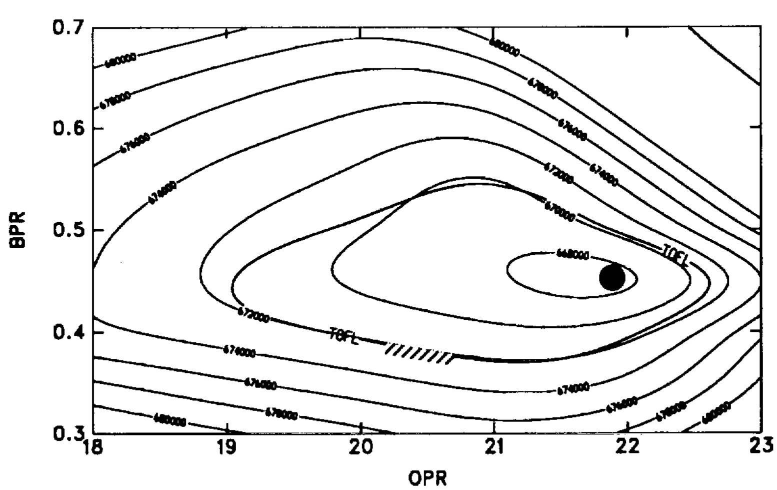


Figure 5.9 - Aircraft takeoff gross weight versus bypass ratio and overall pressure ratio.

design burner temperature that matches the limit also decreases. This characteristic is sketched in as a dashed line on figure 5.8. The next two figures, fan pressure ratio versus overall pressure ratio (fig.

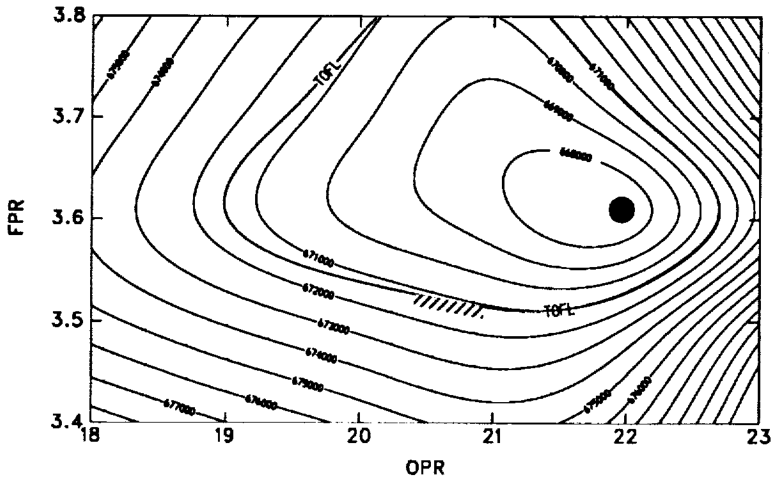


Figure 5.10 - Aircraft takeoff gross weight versus fan pressure ratio and overall pressure ratio.

5.9) and bypass ratio versus overall pressure ratio (fig. 5.10), both show similar characteristics because the compressor discharge temperature limit is being met at the higher overall pressure ratios and very near the limit at the optimum.The final figure for this case was generated using the same input that was used for figure 5.10 except that the compressor discharge temperature limit in the cycle analysis module was set unrealistically high (fig. 5.11). In this case, contours of constant compressor discharge temperature at Mach 2.4 and 65,000 ft. are plotted and, given that the remaining design variables remain unchanged, the impact of this limit on the aircraft takeoff gross weight is clearly illustrated. Though the compressor discharge temperature can be passed to the optimizer as a

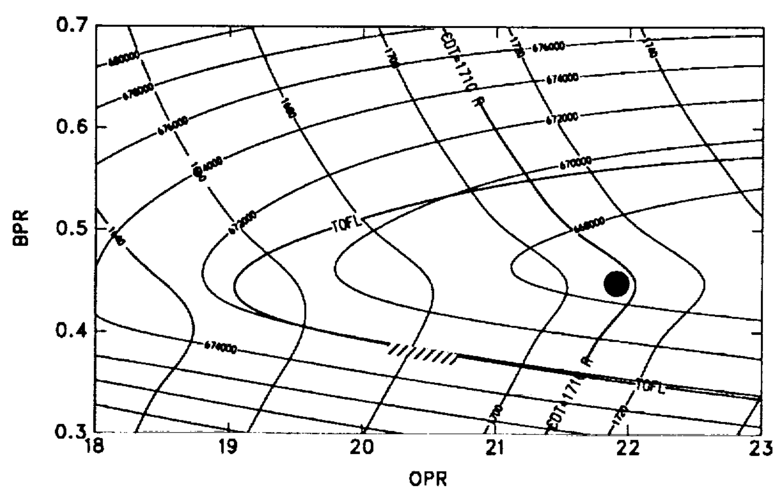


Figure 5.11 - Aircraft takeoff gross weight and compressor discharge temperature versus bypass ratio and overall pressure ratio.

constraint variable, this is not recommended since only one point in the flight envelope is checked whereas in the cycle analysis module the limit is enforced at all points. Typically the maximum compressor discharge temperature will occur at the maximum Mach number with some variation with altitude.

All these results serve to further emphasize the program's capability to locate an optimum. The contour plots that have been generated provide additional information that might be considered in more detailed studies. For example, from figure 5.11 it can be seen that the overall pressure ratio can be reduced from near 22 to near 20 with an aircraft takeoff gross weight penalty of only 2000 lbs. A detailed compressor design may reveal that one fewer stage is required for an overall pressure ratio near 20. Savings in weight, combined with savings in cost could potentially more than overcome the 2000 lb gross weight penalty.

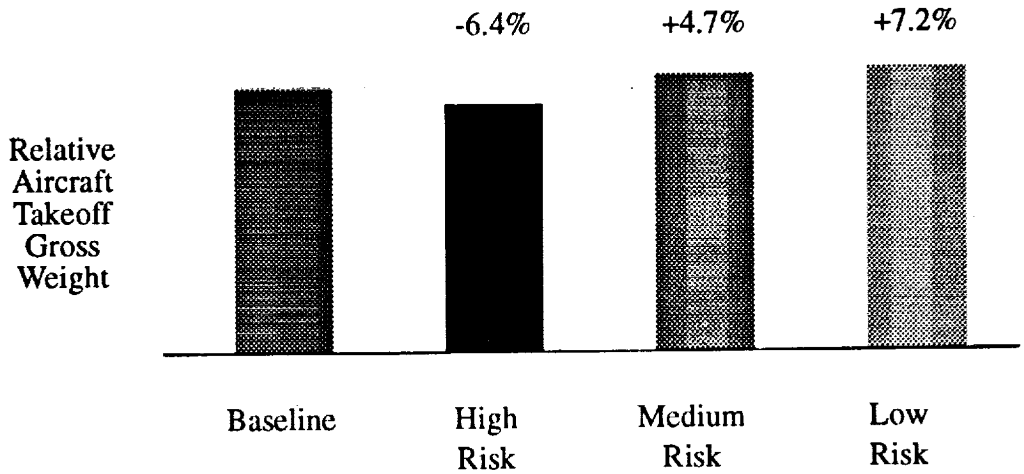
Risk Assessment

This section is titled risk assessment instead of technology assessment because perturbations on compressor discharge and turbine inlet temperatures are relatively small and would likely have the largest impact on compressor and turbine airfoil life rather than the actual materials and weight. The baseline case is the result of the six variable optimization case described in the previous section and uses the temperature limits and turbine cooling flow requirements that are characteristic of what is currently being used in industry and NASA in the HSR program. To achieve the required life of 9000 hours for the hot rotating components, advances in materials, manufacturing techniques, and turbine cooling techniques will likely be required for a 2005 EIS propulsion system. The penalty for not meeting or for relaxing the goals and the benefit from exceeding the goals can be predicted in FLOPS. Three additional 6-variable optimization cases were run: *high risk*, *medium risk*, and *low risk*. The only variations in the input were on the maximum allowable compressor discharge temperature (CDT), the maximum allowable burner temperature (T_4), and the amount of turbine cooling air which changes the turbine inlet temperature (T_{41}). The results are summarized on figure 5.12.

These results clearly show the impact that advanced technologies could have on any future high-speed civil transport. As expected, bypass ratios and overall pressure ratios increased with increasing risk, and compressor discharge temperatures were at the limit for all cases. Though the benefits associated with meeting or exceeding the goals are clear, the *medium risk* case may warrant further study. The 4.7% takeoff gross weight penalty associated with moderate reductions in turbine inlet and compressor discharge temperatures has to be weighed against potential reductions in propulsion system weight, maintainability, and cost. A more detailed study would require the consideration of rotor airfoil and disk life, maintenance and manufacturing costs, and ultimately direct operating cost. This is beyond the scope of this work.

Optimization for Maximum Range

The final section of this chapter will compare two different supersonic cruise aircraft. Both aircraft have a takeoff gross weight of 650,000 lbs and carry 250 passengers. The baseline aircraft has aerodynamic and design characteristics that are similar to aircraft used in the previous cases. It cruises at Mach 0.9 overland and at Mach 2.4 overwater. The aircraft and mission characteristics are summarized in table 5.2. The second aircraft is a low sonic boom concept designed to cruise at Mach 1.6 overland and at Mach 2.0 overwater. Its characteristics are summarized in table 5.3. The wing and fuselage of the low sonic boom aircraft have been shaped and sized to produce a specific pressure signature at the ground and any variation in vehicle weight or wing area would change that signature. As a result, the only design variables are engine thrust and the engine cycle design variables. The



	Baseline	High Risk	Medium Risk	Low Risk
Gross Weight, lb	667,000	624,000	699,000	715,000
CDT Limit, °R	1710	1760	1610	1610
Maximum T_4, °R	3560	3560	3410	3260
Turbine Cooling Air	23%	12%	20%	20%
Maximum T_{41}, °R	3420	3495	3294	3153

Figure 5.12 - Effect of engine cycle temperature limits on aircraft takeoff gross weight.

• Mach 2.4 supersonic transport (Mach 0.9 overland)
• 650,000 lb Takeoff Gross Weight
• 250 passengers
• Mixed Flow Turbofan Compressor Discharge Temperature Limit of 1710 °R Maximum Burner Temperature of 3560 °R
• FAR 36, Stage III Noise Nozzle suppression and weight increase with jet velocity Actual takeoff and sideline noise are not estimated
• Constraints Approach Speed upper limit of 160 kts. Takeoff and Landing Field length of 11,000 ft. Available fuel volume 500 fpm rate of climb capability

Table 5.2 - 250 Passenger Aircraft Configuration and Mission Characteristics

• Mach 2.0 supersonic transport (Mach 1.6 overland)
• 650,000 lb Takeoff Gross Weight
• 250 passengers
• Mixed Flow Turbofan Compressor Discharge Temperature Limit of 1710 °R Maximum Burner Temperature of 3560 °R
• FAR 36, Stage III Noise Nozzle suppression and weight increase with jet velocity Actual takeoff and sideline noise are not estimated
• Constraints Approach Speed upper limit of 160 kts. Takeoff and Landing Field length of 11,000 ft. Available fuel volume 500 fpm rate of climb capability

Table 5.3 - Low Sonic Boom Aircraft Configuration and Mission Characteristics

objective for both of these aircraft concepts was to maximize range. The effect of the design length of the overland segment on the total range and on the cycle selection will be shown as will the effect of flying off-design overland segments.

Current industry studies are designing aircraft and engine cycles for a fixed design mission. However, the selection of the cycles studied in more detail will be based on the economics of an aircraft flying an off-design mission. Though the penalty appears to be small (~1-4% depending on the aircraft) provided that the overwater segment is at least 50% of the total, some cycles may be penalized more than others. In particular those cycle types with specific fuel consumption (SFC) characteristics typical of the mixed flow turbofan (MFTF), where the minimum SFC is at or near the maximum power point, may be penalized more than those cycles with the minimum SFC at part power (fig. 5.13). For this example FLOPS was used to optimize cycles and engine size for both configurations, and wing area for the baseline configuration. Five runs, at different design overland segment lengths, were made using the mixed flow turbofan model on each aircraft concept. One run using the turbine bypass engine was made for a 1500 n.mi. overland design segment for both of the aircraft. The turbine bypass engine used the same compressor discharge and burner outlet temperature limits as the MFTF's. For the baseline configuration, the optimum bypass ratios increased from 0.48 to 0.70 as the length of the design overland segment increased. For the low-boom configuration, the bypass ratios remained relatively constant (near 0.90), except for the 4500 n.mi. design overland segment where the resulting bypass ratio was near 1.1. The resulting designs were then run at various off-design overland segments. The results are summarized in figures 5.14 and 5.15. The solid symbols are the results using the turbine bypass engine and the odd symbol (open for the TBE and solid for the MFTF's) in each curve indicates the design overland range.

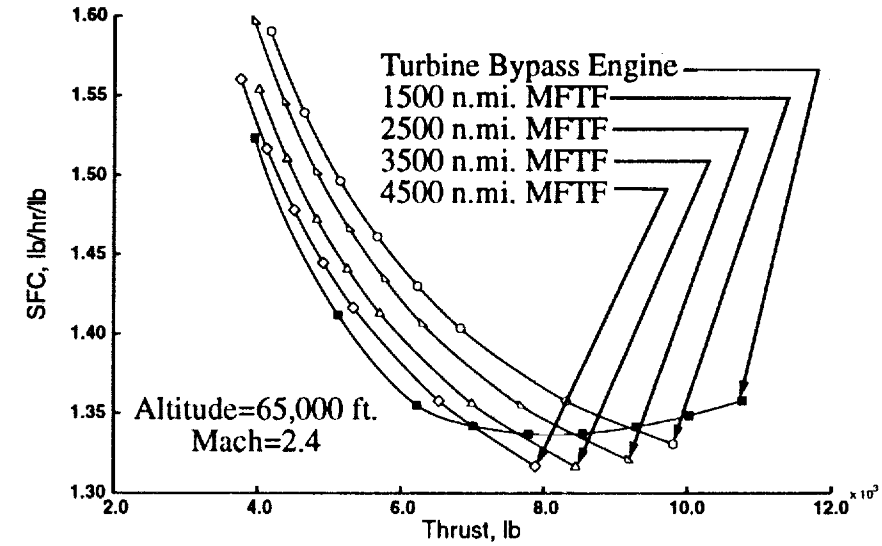


Figure 5.13 - Specific fuel consumption versus thrust for four mixed flow turbofans and a turbine bypass engine.

The fact that the design mission (odd symbol) is at the maximum range for each of the design overland distances suggests that the optimizer is working. As expected, bypass ratios increased with increasing design overland distances for the baseline case. Except for the 4500 n.mi. case, bypass ratios were fairly constant for the low boom concept. This is probably because the variation between the two different cruise Mach numbers is small. All cycles for both concepts exhibited increasing overall pressure ratios and design burner temperatures, with increasing design overland segments. As expected, fan pressure ratios matched the reverse of the bypass ratio trends, and compressor discharge temperatures were near the limit in all cases. For the low boom concept, the only design variables are in the propulsion system, and its effect on the aircraft is evident. Results show that for a 1500 n.mi. overland mission, the low boom concept has a 5% total range penalty relative to the baseline. However, depending on the cycle that is selected, the penalty for flying off-design overland segments can be far more severe for the low boom concept than for the baseline.

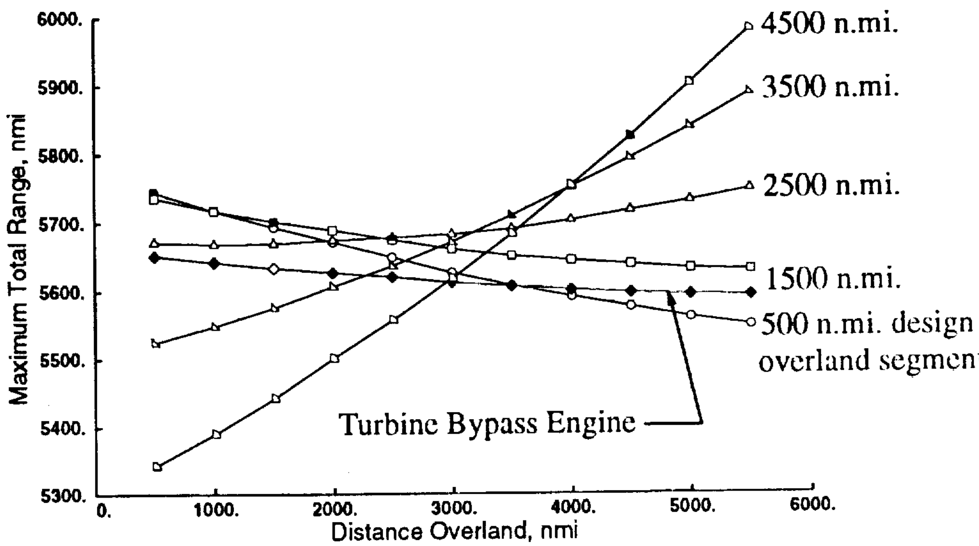


Figure 5.14 - Total range versus distance overland (Mach 0.9) for a 250 passenger 650,000 lb aircraft equipped with engine cycles optimized for various design overland segments and a maximum cruise Mach number of 2.4.

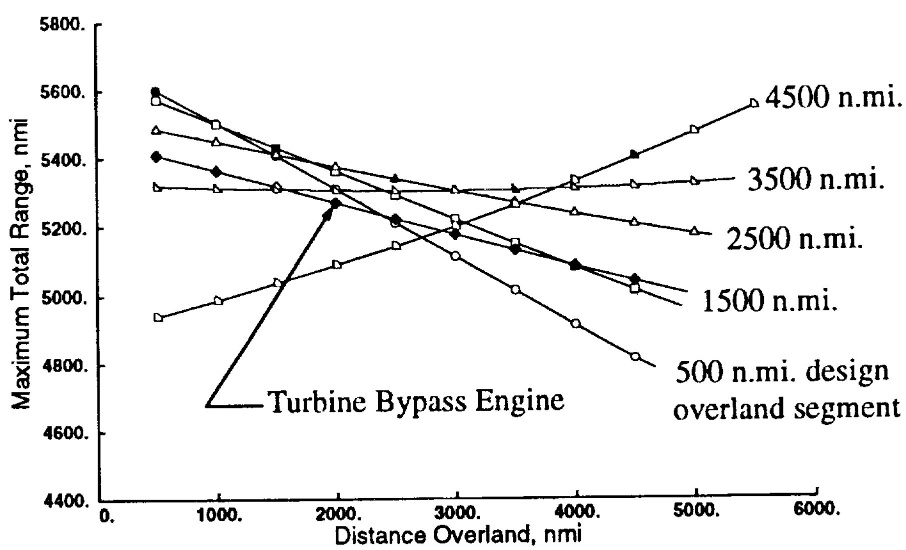


Figure 5.15 - Total range versus distance overland (Mach 1.6) for a 250 passenger 650,000 lb low sonic boom aircraft equipped with engine cycles optimized for various design overland segments and a maximum cruise Mach number of 2.4.

VI. Concluding Remarks

A method for conceptual aircraft design that incorporates direct optimization of major engine design variables for a variety of cycle types has been developed. The method should improve the lengthy screening process currently involved in selecting an appropriate engine cycle for a given application or mission. The new capability will allow environmental concerns such as airport noise and emissions to be addressed early in the design process. The ability to rapidly do optimization and parametric variations using both engine cycle and aircraft design variables, and to see the impact on the aircraft and not just specific fuel consumption and thrust, should provide insight and guidance for more detailed studies. A method for predicting propulsion system weight with minimal user input has been developed and incorporated into the program, and plots of the engine and nacelle can be generated. Though the inlet and nozzle weight predictions may require more detailed input based on a more detailed design for a given application, the predicted bare engine weights were shown to agree reasonably well with industry predictions and data.

While the optimization algorithms in the Flight Optimization System all work fairly well, careful analysis of the results is required. All the results presented in this paper were generated with at least 6 runs. The first run used five different optimization algorithms and one final run was made using the Broyden-Fletcher-Goldfarb-Shano algorithm with the best result from the 5 initial runs as a starting point. Additional runs were made if there was a significant change in the results. Careful analysis of the results included performing parametric variations and generating contour plots. Another method employed in checking the final results was to perform parametric optimizations. In these cases one of the design variables is parametrically varied and the remaining variables are used for optimization. The final results are then compared relative to one another. Any inconsistencies in the objective or the design variables may indicate that there is a problem.

With the new capability that has been incorporated, the Flight Optimization System is now an extremely powerful tool for preliminary analyses of not only advanced aircraft concepts, but of propulsion systems as well. Application of the program to supersonic cruise aircraft has been demonstrated for three problems: optimization for minimum gross weight, optimization for maximum range, and technology risk assessment. The engine cycle thermodynamic data necessary for predicting noise can be generated. Nitrous oxides emissions indices can be estimated and total nitrous oxides emitted for a given mission can be computed. The sensitivity of aircraft takeoff gross weight to propulsion system component efficiencies and customer bleed and power extraction can be easily and rapidly predicted. In cases where manufacturer propulsion system performance data or data developed from more detailed analyses are unavailable, preliminary analyses for a wide variety of aircraft concepts and missions are made easier in that preliminary data that fits the application can be generated.

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Appendix - Cycle Analysis Module Users Guide

The ENGGEN program is a stand-alone version of the Flight Optimization System (FLOPS) engine cycle analysis module. The input consists of Namelist $ENGINE, which contains data normally read into FLOPS in Subroutine CYINIT. Namelist $ENGDIN may optionally be included ahead of namelist $ENGINE to set specific points where data are to be computed. Additionally, if nacelle weight or dimensions are to be computed, namelist $NACELL must be input. This program is not capable of performing parametric variations on engine cycle parameters. The data files that are used by the cycle analysis module are defined in table A-1.

FILE	USAGE
Unit 3	Primary engine cycle input For IENG = 0, User defined (see IFILE in Namelist $ENGINE) For IENG = 1, Named "TURJET" For IENG = 2, Named "TFNSEP" For IENG = 3, Named "TFNMIX" For IENG = 4, Named "TURPRP" For IENG = 5, Named "TBYPAS"
Unit 4	Optional engine cycle output file (see OFILE Namelist $ENGINE, default name = "ENGOUT")
Unit 5	Standard input file
Unit 6	Standard output file
Unit 10	Optional engine cycle analysis debug file named "DEBUG"
Unit 11	Optional external engine deck (Default name = "ENGDEK") If being input - EIFILE, Namelist $ENGDIN If being generated - EOFILE, Namelist $ENGINE
Unit 12	Engine component map tabular data file - TFILE, Namelist $ENGINE (Default name = "ENGTAB").

Table A-1. - Input file unit numbers and use in the cycle analysis module.

FILE	USAGE
Unit 13	Optional output data for noise prediction.
Unit 14	Optional engine / nacelle PostScript plot file - PLTFIL, Namelist $ENGINE (Default name = "ENGPLT").
Unit 15	Temporary scratch file used in generating above.

Table A-1. - Input file unit numbers and use in the cycle analysis module. (Concluded)

The order in which the data is input is shown in table A-2.

Namelist $ENGDIN (optional)
Namelist $ENGINE
Namelist $NACELL (if NGINWT ≠ 0)

Table A-2. - Namelist Input Order

Input Data Description

Namelist $ENGDIN

Only those variables applicable to the stand alone module are shown. additional variables are used by FLOPS.

Name	Description
EMACH(I)	Array of Mach numbers in descending order at which engine data are to be generated (Default computed internally, Maximum = 20, Minimum = 2, Do not zero fill)
ALT(J,I)	Arrays of altitudes in descending order, one set for each Mach number, at which engine data are to be generated (Default computed internally, Maximum = 15 per Mach number, Minimum = 2 per Mach number, Do not zero fill). Altitudes and numbers of altitudes do not have to be consistent between Mach numbers

Namelist $ENGINE

Some variables in this namelist are logical, others are not (i.e. integer, real, or character type).

Name	Description
IENG	Engine cycle definition input file indicator = 0, User defined engine cycle (See IFILE below) = 1, Turbojet (IFILE = 'TURJET', Default) = 2, Separate flow turbofan (IFILE = 'TFNSEP') = 3, Mixed flow turbofan (IFILE = 'TFNMIX') = 4, Turboprop (IFILE = 'TURPRP') = 5, Turbine bypass (IFILE = 'TBYPAS')
IFILE	Name of cycle definition file. Used only if IENG = 0, but there must be an external file with the correct name (See IENG above) available (No default).
TFILE	Name of the file containing component map tables (Default = 'ENGTAB'). This is a required file.
IPRINT	Engine cycle analysis printout control. Printout is on file OFILE (See below). = 0, Important warning messages only = 1, Normal output (Default, 200 - 1000 lines) = 2, Plus component and station data at each full throttle point (2000 - 3500 lines) = 3, Plus component and station data at each part power point and engine component tabular data (2500 - 25000 lines, depending on ITHROT below) = 4, Plus convergence history (5000 - 35000 lines)
NPRINT	Noise data print control. = 0, no printout (default) = 1, print noise data file to file named ANOPP =-1, print compressor component operating line on normal output file if IPRINT > 0.
OFILE	Name of engine cycle analysis printout file (Default = 'ENGOUT'). If OFILE = 'OUTPUT', printout will be put on the standard output file (Unit 6). If IPRINT = 0 (See above), OFILE is set to 'OUTPUT' automatically.

Name	Description
GENDEK	If .TRUE., engine data will be saved on the file designated by EOFILE (below) as an Engine Deck for future use (Default = .FALSE.)
EOFILE	Name of output Engine Deck for GENDEK = .TRUE. (Default = 'ENGDEK', See EIFILE in Namelist $ENGDIN)
ITHROT	Controls frequency of part power data generation = 0, Computed at each Mach-altitude combination = 1, Computed only at the maximum altitude for each Mach number (Default) = 2, Computed only once, at the maximum altitude for the maximum Mach number Values of 1 or 2 will save over half of the engine generation cpu time with little impact on results, but IFILL must be > 0 in Namelist $ENGDIN.

The following 3 variables control the number of part power throttle settings generated. Since the mission analysis module can only use 16, it is recommended that the engine cycle analysis module be used to generate up to 15 and that IDLE > 0 in Namelist $ENGDIN be used to generate flight idle.

Name	Description
NPAB	Maximum number of afterburning throttle settings for each Mach-altitude combination (Default = 0)
NPDRY	Maximum number of dry (non-afterburning) throttle settings (Default = 15, NPAB + NPDRY .LE. 30)
XIDLE	Fraction of maximum dry thrust used as a cutoff for part power throttle settings (Default = .05)
NITMAX	Maximum iterations per point (Default = 50)

Cycle Design Point Data

Name	Description
DESFN	Engine design point net thrust, lb (Default = 10000.)
XMDES	Engine optimization point Mach number (Default = 0.) XMDES and XADES are used for "PROPulsion Only" analyses and do not apply when running program ENGGEN.

Name	Description
XADES	Engine optimization point altitude, ft (Default = 0., see also XMDES above) If XADES < 0., it is interpreted as the negative of the design point dynamic pressure (psf), and the altitude is back-calculated with a minimum of 0.

The following 5 variables are overridden if comparable data is input in Namelist $CONFIN.

Name	Description
OPRDES	Overall pressure ratio (Default = 15.0)
FPRDES	Fan pressure ratio (Default = 1.5, turbofans only)
BPRDES	Bypass ratio (Turbofans only, Default is computed based on OPRDES, FPRDES, TTRDES, XMDES and ALDES). For turbine bypass engines BPRDES must be input and is defined as the fraction of compressor exit airflow that is bypassed around the main burner and the turbine.
TETDES	Engine design point turbine entry temperature, °R (Default = 2500.)
TTRDES	Engine throttle ratio defined as the ratio of the maximum allowable turbine inlet temperature divided by the design point turbine inlet temperature. If TTRDES is greater than TETDES, it is assumed to be the maximum allowable turbine inlet temperature. (Default = 1.0)

Other Engine Configuration Definition Data

Name	Description
HPCPR	Pressure ratio of the high pressure (third) compressor. (Only used if there are three compressor components.)
ABURN	True if there is an afterburner (Default = .FALSE.)
DBURN	True if there is a duct burner (Separate flow turbofans only, Default = .FALSE.) ABURN and DBURN cannot both be true.
EFFAB	Afterburner/duct burner efficiency (Default = .85)
TABMAX	Maximum afterburner/duct burner temperature, °R (Default = 3500.)

Name	Description
VEN	True if the exhaust nozzle has a variable flow area (Default = .FALSE.) The nozzle flow area is automatically allowed to vary for cases when the afterburner or duct burner is on.
COSTBL	Customer high pressure compressor bleed, lb/sec (Default = 1.)
HPEXT	Customer power extraction, hp (Default = 200.)
WCOOL	Turbine cooling flow as a fraction of high pressure compressor mass flow. The cooling flow defaults to the value in the engine cycle definition file. If WCOOL is input greater than or equal to zero the default will be overridden.
FHV	Fuel heating value, btu/lb (Default = 18500.)
DTC	Deviation from standard day temperature in °C The deviation, as used in the cycle analysis module, is DTC at sea level and varies to zero at ALC (see below). The design point is at standard temperature. (See also DTC in TOLIN and MISSIN. These temperature deviations are independent and default to zero.)
ALC	The altitude at which DTC (see above) becomes zero. (Default = 10000. ft.)
YEAR	Technology availability date used to estimate compressor polytropic efficiency (Default = 1985.)
BOAT	True to include boattail drag (Default = .FALSE.)
AJMAX	Nozzle reference area for boattail drag, sq ft. Used only if BOAT = .TRUE. Default is the largest of 1) 1.1 times the inlet capture area 2) Nozzle exit area at the inlet design point 3) Estimated engine frontal area 4) Estimated nozzle entrance area or if nacelle weight and geometry calculations are performed (see NGINWT below) AJMAX is set to the nacelle cross-sectional area at the customer connect. or if AJMAX is less than zero, the cruise design point nozzle exit area multiplied by the absolute value of AJMAX is used as the reference.
SPILL	True to include spillage and lip drag in engine performance data (Default = .FALSE.)

The next 5 variables are used only if SPILL = .TRUE.

Name	Description
LIP	Compute inlet cowl lip drag (Default = .FALSE.)
BLMAX	Inlet bleed flow fraction of total flow at the inlet design point (Default = .016 * AMINDS**1.5)
SPLDES	Inlet design spillage fraction (Default = .01)
AMINDS	Inlet design Mach number (Default = XMMAX)
ALINDS	Inlet design altitude, ft (Default = AMAX)

The next 2 variables are used only for turboprops (IENG = 4)

Name	Description
ETAPRP	Maximum propeller efficiency (Default = 0.840). The actual propeller efficiency is based on an internal schedule of efficiency versus Mach number with the maximum efficiency (ETAPRP) occurring at a Mach number of 0.80.
SHPOWA	Design point shaft horsepower divided by the design point core airflow, HP/(lb/sec) (Default = 60).

The next 6 variables define the Mach-altitude array points at which engine performance.data is to be computed unless EMACH and ALT are input in Namelist $ENGDIN.

Name	Description
XMMAX	Maximum Mach number (Required)
AMAX	Maximum altitude, ft (Default computed from XMMAX and QMIN)
XMINC	Mach number increment (Default = .2)
AINC	Altitude increment (Default = 5000.)
QMIN	Minimum dynamic pressure, psf (Default = 150.)
QMAX	Maximum dynamic pressure, psf (Default = 1200.)

The following five variables are used in engine behavioral constraints. (See Namelist $SYNTIN, G (8 to 12) respectively.) In addition, CDTMAX and CDPMAX are used during the cycle analysis as constraints on engine operation at all points in the flight envelope unless LIMCD is input as zero.

Name	Description
CDTMAX	Maximum allowable compressor discharge temperature, °R (Default = 99999.).
CDPMAX	Maximum allowable compressor discharge pressure, psi (Default = 99999.).
VJMAX	Maximum allowable jet velocity, ft/sec (Default = 99999.)
STMIN	Minimum allowable specific thrust, lb/lb/sec (Default = 1.)
ARMAX	Maximum allowable ratio of the bypass area to the core area of a mixed flow turbofan (Default = 99999.)
LIMCD	Switch to use the compressor discharge temperature and pressure limits only for optimization. = 0, values at the cruise design Mach number and altitude may be used as constraints during optimization = 1, limits enforced at all points in the flight envelope (Default)

The remaining variables may be used to predict engine weight and dimensions.

Name	Description
NGINWT	Switch for engine weight calculations = 0, none (Default) = 1, engine only = 2, engine and inlet = 3, engine, inlet, and nacelle = 4, engine, inlet, nacelle, and nozzle > 9, to use equations Use the negative value to calculate the weight for the initial design and then scale engine weights and dimensions with airflow.

Name	Description
IWTPRT	Printout control for weight calculations. Printout is on file OFILE. = 0, No output. = 1, Print component weights and dimensions (Default). = 2, Print component design details. = 3, Plus initial and final optimization data. = 4, Print component details at each iteration.
IWTPLT	PostScript plot control for engine (and nacelle) schematics on file PLTFIL (See below, default = 0). If nacelle weight is not calculated (see NGINWT above) it will not be plotted. If the negative value is input, only the final design will be plotted. = 0, No plot (Default). = 1, One correct aspect ratio plot of engine and nacelle (one page per design). = 2, One correct aspect ratio plot of engine only and one of engine and nacelle (two pages per design). = 3, 1 with full page plot (two pages per design). = 4, 2 with full page plot (four pages per design).
PLTFIL	Name of the PostScript plot file (Default = ENGPLT).

The following four variables are used if NGINWT is non-zero

Name	Description
GRATIO	Ratio of the RPM of the low pressure compressor to the rpm of the connected fan (Default = 1).
UTIP1	Tip speed of the first compressor (or fan) in the flow. Default is based on YEAR, engine type, and other design considerations.
RH2T1	Hub to tip radius ratio of the first compressor (or fan) in the flow. Default is based on YEAR, engine type, and other design considerations.
IGV	Flag for compressor inlet guide vanes. = 0, None (default). = 1, Fixed. = 2, Variable. Use negative 1 or 2 for no IGV on the fan.

Name	Description
TRBAN2	Maximum allowable AN^2 for turbine components. The input value is the actual maximum divided by 10^{10}. AN^2 is the flow area in square inches multiplied by the rotational speed squared and has units of $in^2 * RPM^2$. The default is based on year.
TRBSTR	Turbine usable stress lower limit, psi. Normally when component weights are predicted, the usable stress is a function of operating conditions. For turbine components, this can be unusually low because cooling effects are not accounted for. (Default = 15000.)
CMPAN2	Maximum allowable AN^2 for compressor components. The input value is the actual maximum divided by 10^{10}. AN^2 is the flow area in square inches multiplied by the rotational speed squared and has units of $in^2 * RPM^2$. The default is based on year.
CMPSTR	Requested compressor usable stress, psi. This forces a change in compressor material when the current (lower temperature) material starts to run out of strength as temperature increases. (Default = 25000.)

Namelist $NACELL

Namelist $NACELL if required only if nacelle weight and dimensions are to be computed. Most of the input is geometry (refer to figure A-1) and the remaining input is used in predicting weight.

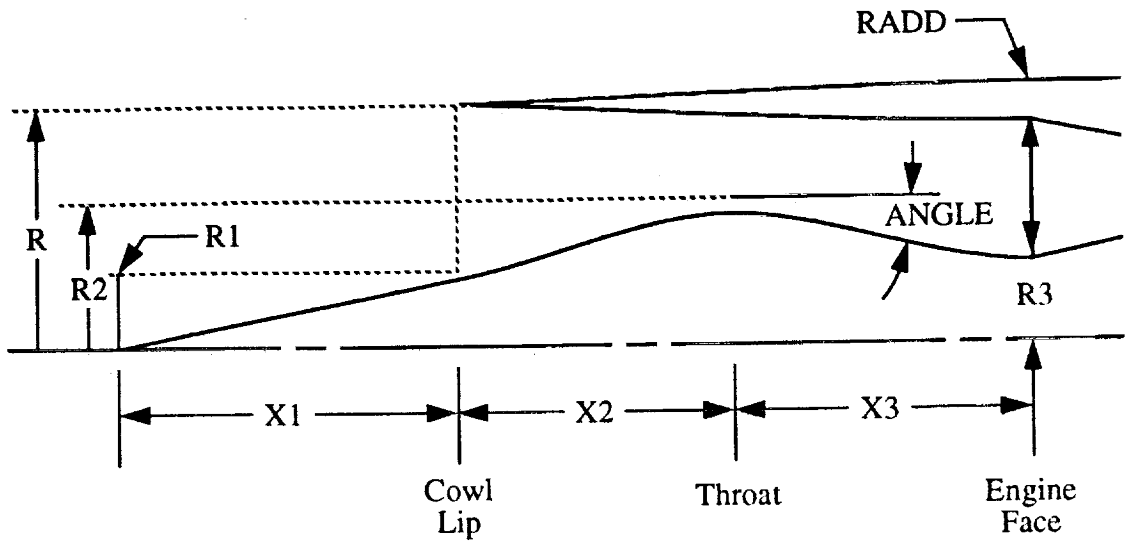

Figure A-6 - Inlet geometry definition.

Where:

R	Inlet capture radius (height for 2D)
R3	Compressor hub radius (zero for 2D)
X3	Subsonic Diffuser length based on R2, R3, and ANGLE.

The remaining variables may be input. The default values are based on a Mach number 2.4 axisymmetric translating centerbody inlet and are used only if MIXED (see below) is greater than zero.

Name	Description
X1R	X1 / R (Default = 2.06). If IVAR (see below) = -1, X1R is the cowl length divided by the inlet capture radius. Use IVAR=-1 for subsonic nacelles.
X2R	X2 / R (Default = 1.58)

Name	Description
R1R	R1 / R (Default = .354)
R2R	R2 / R (Default = .585)
ANGLE	Average angle of the subsonic diffuser portion of the inlet between the throat and the engine face (Default = 10 degrees).
CLANG	Cowl lip angle (Default =0.). Generally only used for external compression inlets.
MIXED	Inlet compression type indicator. = -1, inlet geometry is based solely on the geometry variables described above (Default). = 0, inlet geometry is based in the internal geometry data base for external compression inlets and the given inlet design Mach number. = 1, inlet geometry is based in the internal geometry data base for mixed compression inlets and the given inlet design Mach number.
RADD	Distance from the engine compressor tip to the exterior of the nacelle (Default = 3. in). If RADD < 1. the added radial distance is RADD times the compressor tip radius.
XNLOD	Nozzle length / diameter (Default is computed).
XNLD2	Fan nozzle length / height (Default is computed).
INAC	Nacelle type indicator = 0, none (Default) = 1, axisymmetric = 2, two-dimensional (use -2 if two or more engines are to be podded together) = 3, two-dimensional inlet / axisymmetric nozzle (use -3 if two or more engines are to be podded together) = 4, two-dimensional bifurcated inlet / two- dimensional nozzle (use -4 if two or more engines are to be podded together) = 5, two-dimensional bifurcated inlet / axisymmetric nozzle (use -5 if two or more engines are to be podded together)

Name	Description
IVAR	Variable geometry switch used to estimate weight factor WTCB1 described below. =-1, fixed geometry inlet (no centerbody) (See X1R above.) = 0, fixed geometry inlet (with centerbody) = 1, translating centerbody, (Default) = 2, collapsing centerbody = 3, translating and collapsing centerbody
NVAR	Variable geometry switch used to estimate weight factor WTNOZ described below. = 0, fixed geometry nozzle, (Default) = 1, variable area throat = 2, variable area exit = 3, variable area throat and exit = 4, fixed geometry plug core nozzle and fixed geometry fan nozzle (typical subsonic transport installation).

The following weighting factors are multiplied by the surface area of the applicable inlet section to predict inlet weight. The defaults are based on the internal materials data base and the maximum cruise Mach number.

Name	Description
WTCB1	Weighting factor for the inlet centerbody up to the throat.
WTCB2	Weighting factor for the inlet centerbody from the throat to the engine face.
WTINT	Weighting factor for the internal cowl up to the engine face.
WTEXT	Weighting factor for the external nacelle.
WTNOZ	Weighting factor for the nozzle.

The remaining variables are only used for 2D nacelles (INAC = 2).

Name	Description
H2W	Inlet height to width ratio for 2D inlets (Default = 1.0).

Engine Cycle Analysis Methodology in FLOPS

A new engine cycle analysis module has been developed for the Flight Optimization System (FLOPS). The module is primarily based on QNEP (ref. A-1, for further information, see refs. A-2 and A-3). This module is now part of the production version of FLOPS. It is also available as a stand-alone program ('ENGGEN').

Engine Design Point

The engine design point is defined at sea level static (Mach number 0.1 at sea level for turboprops). The design point variables are overall pressure ratio (OPRDES), fan pressure ratio (FPRDES), bypass ratio (BPRDES), turbine entry temperature (TETDES), throttle ratio (TTRDES) and thrust (DESFN). Fan pressure ratio is only used for mixed and separate flow turbofans. Bypass ratio is only used for mixed and separate flow turbofans and turbine bypass engines. If the engine being modeled is a turbine bypass engine, the bypass ratio (BPRDES) is defined as the fraction of compressor exit airflow that is bypassed around the main burner and the turbine. Even though throttle ratio is not used at the design point, it is still considered a design variable since it affects off design performance. The design thrust is the maximum dry (non-afterburning) thrust at the design point. Any input value of THRSO (namelist $WTIN) will be overridden by the maximum sea level static thrust regardless of the engine cycle selected, not necessarily the design thrust (DESFN).

Off Design Operation

During off design operation, the engine is run with the maximum allowable turbine entry temperature (TETDES*TTRDES). Then, if any constraints (see below) are violated at any point in the flight envelope, the turbine entry temperature is reduced until no constraints are violated. For turbine bypass engines, instead of reducing turbine entry temperature, engine inlet airflow is reduced. During off design operation all the cycles, except the TBE, use engine inlet airflow as a variable to "match" the engine. The TBE maintains maximum airflow by allowing the turbine bleed bypass air (TBB) to vary. As the TBE is throttled back the TBB goes to zero. When this occurs the TBB is fixed at zero and the engine inlet airflow is allowed to vary.

Engine Off Design Constraints

1) The engine corrected flow must not exceed the corrected flow at the design point, CWDES. CWDES is an internal variable established during the design point. Corrected flow is defined as:

$$W \cdot \frac{\sqrt{T/T_{\text{ref}}}}{P/P_{\text{ref}}}$$

where: W is the mass flow, T the temperature and P the pressure (all at the engine face) and the reference conditions are standard day sea level conditions.

Also, the engine corrected flow must not exceed the corrected flow as defined in the first table in the engine component map tabular data file (TFILE, Namelist $ENGINE, Default name = "ENGTAB") This table contains the ratio of the corrected flow of the inlet to the engine design corrected flow as a function of Mach number. If the ratio is greater than 1.0, this same ratio is used as a multiplier on the maximum allowable overall pressure ratio (see constraint 4). Also, if the ratio is greater than 1.0, the first constraint may be violated. For turbine bypass engines, the engine corrected airflow is set to the value obtained from this table, and this constraint is automatically satisfied.

2) If and only if IFAN is set to some nonzero value in the engine cycle definition $D namelist input file corresponding to some compressor component number, then the design point surge margin for that compressor component (usually a fan) is used as a constraint.

3) The compressor exit temperature must not exceed the maximum allowable temperature, CDTMAX. CDTMAX is input and default is large (see also LIMCD in namelist $ENGINE).

4) The compressor exit pressure must not exceed the maximum allowable pressure, CDPMAX. CDPMAX is input and default is large.

5) The overall pressure ratio must not exceed the design overall pressure ratio, OPRDES. OPRDES is input.

6) For turboprops the shaft horsepower must not exceed the shaft horsepower at the design point, SHPDES. SHPDES is an internal variable established during the design point.

Constraints 1, 2 and 5 are primarily there to ensure that compressor components do not operate outside the bounds of the compressor maps as defined in the engine component map tabular data file (TFILE, Namelist $ENGINE, Default name = "ENGTAB"). Constraint 2 is available in case an inlet with known flow handling capabilities is available. The remaining constraints (3, 4 and 6) are there to ensure that components do not melt, blow up or break.

Namelist $D

Namelist $D is not input directly into FLOPS. Instead it is read from the engine cycle definition file named in namelist $ENGINE with the variable IFILE. Most of the input definitions are defined in reference A-1. Data definitions that are not defined or have been changed for this application, are defined in tables A-3 and A-4. Also, the corrected flow schedule that must be the first table in the component map tables file "TFILE" is now corrected flow versus Mach, not versus corrected temperature, as documented in reference A-1 Also, the output data variable DATOUT5 for compressor components has been changed to compressor surge margin from referred speed scale factor. Data defined in reference A-1 that are no longer available are IPRINT, TITLE, ALT, NM, XMA, NP, TDEL, PUNCH0, and ENDRUN.

Component Type	CDAT Number	Definition
Inlet	14	Corrected flow at the exit station. (Used internally with the inlet flow schedule.)
Duct	10	Fraction of burner inlet air not heated.
Compressor	12	Adiabatic efficiency at the design point, except for: = 0; the adiabatic efficiency at the design point is a function of the design point pressure ratio and the technology parameter *YEAR*. = -1; the maximum adiabatic efficiency on the map is set as a function of design point pressure ratio and *YEAR* (as above), and the design point efficiency is based on the design point *R* value and $N/\sqrt{\theta}$. < -.3 and > -1.; CDAT12 is the negative of the polytropic efficiency at the design point. < .3 or > -.3; CDAT12 is added to the efficiency as computed for CDAT12 = 0.
	15	Compressor polytropic efficiency. (Used internally for weight calculations.)
Nozzle	11	Calculated exit static temperature (internal use only).

Table A-3. - Redefined component input data in namelist D.

Variable	Definition
IFNCON	Component number of the control used for setting the design point net thrust.
IABCON	Component number of the control that satisfies the error upstream of the primary nozzle. This control is deactivated for operation where the nozzle area is allowed to vary (i.e. afterburner on).
IFAN	Component number of the compressor which will have its surge margin limited to its design point surge margin (usually the first compressor component in the flow).
AMAXN1	Multiplier on the design nozzle throat area. The resulting area is used as the maximum allowable area.
N1AC1	Component number of the control that is to be deactivated if the primary nozzle throat area exceeds the maximum.
N1AC2	Component number of the control that is to be activated if the primary nozzle throat area exceeds the maximum.
ICTBE	Component number of the control that allows the turbine bypass bleed flow to vary.
JTBE	Component number of the duct from which the turbine bypass bleed is extracted. A nonzero value for JTBE indicates that the cycle being modeled is a turbine bypass engine.

Table A-4. - New Data in Namelist D.

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13. ABSTRACT *(Maximum 200 words)*

A method for conceptual aircraft design that incorporates the optimization of major engine design variables for a variety of cycle types has been developed. The methodology should improve the lengthy screening process currently involved in selecting an appropriate engine cycle for a given application or mission. The new capability will allow environmental concerns such as airport noise and emissions to be addressed early in the design process. The ability to rapidly perform optimization and parametric variations using both engine cycle and aircraft design variables, and to see the impact on the aircraft, should provide insight and guidance for more detailed studies.

This paper begins with a brief description of the aircraft performance and mission analysis program and the engine cycle analysis program that were used in this work. A new method of predicting propulsion system weight and dimensions using thermodynamic cycle data, preliminary design, and semi-empirical techniques is introduced. Propulsion system performance and weights data generated by the program are compared with industry data and data generated using well established codes. The ability of the optimization techniques to locate an optimum is demonstrated and some of the problems that had to be solved to accomplish this are illustrated. This paper concludes with results from the application of the program to the analysis of three supersonic transport concepts installed with mixed flow turbofans. The results from the application to a Mach 2.4, 5000 n.mi. transport indicate that the optimum bypass ratio is near 0.45 with less than 1% variation in minimum gross weight for bypass ratios ranging from 0.3 to 0.6. In the final application of the program, a low sonic boom fixed takeoff gross weight concept that would fly at Mach 2.0 overwater and at Mach 1.6 overland is compared with a baseline concept of the same takeoff gross weight that would fly Mach 2.4 overwater and subsonically overland. The results indicate that for the design mission, the low boom concept has a 5% total range penalty relative to the baseline. Additional cycles were optimized for various design overland distances and the effect of flying off-design overland distances is illustrated.

14. SUBJECT TERMS
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