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ANALYSIS OF AN ADVANCED TECHNOLOGY SUBSONIC TURBOFAN INCORPORATING REVOLUTIONARY MATERIALS

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SUMMARY

Successful implementation of revolutionary composite materials in an advanced turbofan offers the possibility of further improvements in engine performance and thrust-to-weight ratio relative to current metallic materials. The present analysis determines the approximate engine cycle and configuration for an early 21st century subsonic turbofan incorporating all composite materials. The advanced engine is evaluated relative to a current technology baseline engine in terms of its potential fuel savings for an intercontinental quadjet having a design range of 5500 nmi and a payload of 500 passengers.

The resultant near optimum, uncooled, two-spool, advanced engine has an overall pressure ratio of 87, a bypass ratio of 18, a geared fan, and a turbine rotor-inlet temperature of 3085 °R. Relative to the baseline, the advanced engine yields a 22 percent improvement in cruise TSFC and a 36-percent reduction in engine weight. Together these improvements result in a 33-percent fuel saving for the specified mission.

Various advanced composite materials are used throughout the engine. For example, advanced polymer composite materials are used for the fan and the low pressure compressor (LPC). A Ti metal matrix composite is used for the high pressure compressor (HPC) to accommodate the higher operating temperatures. Ceramic composites are used for the combustor and both turbines.

The advanced engine's performance includes aggressive component efficiencies based on these new materials and structural changes such as swept fan and compressor blades, uncooled turbines, reduced hub tip ratios, higher blade loadings, reduced clearances, and three-dimensional design concepts.

INTRODUCTION

Over the past 40 yr, advanced turbine engines have been the pacing item in terms of the U.S. competitive edge in commercial aviation. During this period the specific fuel consumption (TSFC) of subsonic turbine engines has been reduced by about 40 percent. This reduction has been achieved through improvements in component aerodynamics, materials, and turbine cooling effectiveness. Improvements in materials and turbine cooling have resulted in the maximum turbine temperature being increased from 1000 to 2600 °F. Engine overall pressure ratios have increased from 5 to over 38 (refs. 1 to 3), and bypass ratios from 0 (turbojet) to 7 (turbofan). Advanced metallic materials have also allowed tip speeds and blade loading to be increased resulting in fewer but more efficient stages and lighter weight components. Composite materials are just beginning to be used in turbine engines and then only for nonrotating components.

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Successful implementation of revolutionary composite materials (e.g., polymer, metal matrix, and high-temperature nonmetallic composites) in an advanced turbofan offers the possibility of still further improvements in engine performance and thrust-to-weight ratios relative to current conventional materials (i.e., titanium, steel, and superalloys). Advanced composite materials with advanced structures will allow higher tip speeds and thinner blades resulting in fewer and more efficient stages. Advanced nonmetallic composites will allow turbines to operate uncooled at higher turbine inlet temperatures resulting in higher overall pressure ratios and bypass ratios, thereby improving performance. Lower material densities and, therefore, reduced blade weights will result in lower stresses and reduced engine weight. Advanced structures such as drum construction rather than disks will also result in lower engine weights.

The purpose of this study was to (1) determine the approximate cycle and configuration for a turbofan engine incorporating revolutionary all-composite materials, and (2) evaluate the potential fuel saving relative to an engine using current technology (current material) for a commercial subsonic transport mission. This was done by conducting both engine cycle and flowpath studies.

ANALYSIS

Mission

For this study, an intercontinental quadjet having a design range of 5500 nmi and a payload of 500 passengers was assumed. Engines having a thrust of about 10 000 lb at Mach 0.8 and 35 000 ft would be required. This size engine was, therefore, considered in the present study. Sensitivity factors for engine performance (TSFC) and weight were used to determine changes in fuel consumption.

Baseline Engine

The baseline engine used for the study is similar to the Maximum Efficiency Energy Efficient Engine of reference 4. It is a two-spool engine with the fan and the low pressure compressor directly driven by the low pressure turbine. The engine is based on current technology. Compressor pressure ratios are listed in table I along with the turbine rotor-inlet temperature. Air for cooling the turbine is extracted at the exit of the high pressure compressor. Turbine cooling requirements for the baseline engine are based on the method outlined in reference 5. Compressor exit bleed requirements for turbine cooling are based on values for cooling effectiveness assuming advanced convection cooling with trailing edge ejection. Current allowable bulk metal temperatures were used for the vanes (2200 °R) and blades (2100 °R). Based on the turbine stage cooling requirements, the stage efficiency was corrected accordingly.

Advanced Engines

Cycle study. - For the advanced engine, a cycle study was conducted to define an engine cycle based on thrust specific fuel consumption (TSFC) as the figure of merit. TSFC is influenced by engine cycle parameters such as overall pressure ratio (OPR), bypass ratio (BPR), turbine rotor-inlet temperature (T_{41}), component efficiencies, and component configurations. An in-house design-point study program (FACE) was used to determine design point TSFC's and to screen various separate flow turbofan configurations. Inputs for this program include fan and compressor pressure ratios, number of stages, type of compressor (axial or centrifugal), turbine rotor-inlet temperature, number of spools, number of turbine stages, and type of turbine cooling configuration.

For the present study, two levels of compressor and turbine efficiencies were considered. One level represents current technology as opposed to the aggressive second level, which represents advanced technology. Based on discussions with NASA component personnel and a study conducted under NASA contract (ref. 4), higher component efficiencies due to the use of advanced composite materials were postulated based on components having thinner blades, higher tip velocities, uncooled turbines, improved clearance control, and reduced hub-tip ratios in addition to making more efficient use of advanced three-dimensional, CFM design technology. Efficiencies for the compressors and turbines are determined on the basis of stage pressure ratio and work factor ($gJ\ \Delta H/N/U_m^2$), respectively. Symbols are defined in appendix A. Compressor and turbine efficiencies are then corrected for size effects. Turbine efficiencies are also corrected to account for clearance and turbine cooling effects.

Advanced Components and Materials

Advanced materials currently being considered to allow future turbine engines to operate efficiently at high temperatures and pressures are shown in figure 1.

Compressors. - Both polymer and metal matrix composites (refs. 6 to 9) are candidates for fan blades and the front stages of a compressor. For the latter stages operating at higher temperatures metal matrix and intermetallics are possibilities. The rotor may consist of a drum for retaining the blades (outside the scope of the present study).

Combustor. - Advanced materials are also being considered for future turbine engine combustors. These materials included oxide dispersion strengthened (ODS) superalloys and nonmetallic composites, such as ceramic composites and carbon-carbon. These materials would enable the combustor to be operated at higher temperatures with little or no cooling.

Turbines. - The development of improved materials and better methods of cooling to attain higher turbine temperatures has been one of the primary methods for achieving more efficient and higher thrust-to-weight ratio engines. The potential use temperatures for various materials for both vanes and blades are shown in figures 2 and 3. For both vanes and blades, nonmetallic composites such as ceramic composites and carbon-carbon C-C (refs. 10 to 12) may permit use temperatures as high as 3460 and 4460 °R, respectively. With allowable use temperatures of this magnitude, there would be no need for turbine cooling.

Nozzles. - Advanced materials such as nonmetallic composites are also being considered for turbine engine nozzles.

Flowpath. - Based on the cycle studies, flowpaths for selected engines were then determined using NNEP (ref. 13) and the NASA weight code (ref. 14). Thermodynamic inputs required for the weight code are determined in NNEP. The weight program determines the weight of each component in the engine such as compressors, burner, turbines, frames, gearbox, and accessories. Component weights are determined on the basis of a preliminary design approach considering stress levels, maximum temperature, material density, geometry, stage loading, and hub-tip ratio. Based on the advanced materials of figure 1 selected for each component and the data presented in reference 4, the weight program input parameters for the baseline engine were updated to account for differences between the baseline turbofan and the advanced turbofan using advanced composite materials. The material for each component was selected on the basis of maximum component temperature. Relative to a current aluminum or magnesium gearbox housing, a metal matrix composite may result in a stiffer housing. However, for this study no weight advantage was considered.

Installed performance. - The uninstalled performances for the baseline and the advanced engines were corrected for nacelle drag. Based on the differences in installed TSFC and weight between the two engines, fuel savings were determined using mission sensitivity factors (ref. 15). These sensitivity factors were determined from a mission analysis of an intercontinental, turbofan powered transport having a range of 5500 nmi and a payload of 500 passengers.

RESULTS AND DISCUSSION

To minimize TSFC for an advanced turbofan engine providing a thrust of 10 000 lb at Mach 0.8 and 35 000 ft, the following parameters were considered: turbine rotor-inlet temperature (T_{41}), fan and compressor pressure ratios, overall pressure ratio (OPR), bypass ratio (BPR), and number of stages for a two-spool engine.

Effect of Advanced Engine Design Parameters on Performance

Turbine rotor-inlet temperature (T_{41}). - The level of turbine inlet temperature can affect not only engine performance (TSFC), but also engine size and, therefore weight. The effect of T_{41} on TSFC for a range of OPR's is shown in figure 4. For these advanced engines, the turbines are uncooled. The pressure ratio for the fan (1.55(bypass)/1.4(core)) and the low pressure compressor (LPC) were held constant while the pressure ratio of the high pressure compressor (HPC) was varied to achieve the specified OPR. The number of axial stages for the LPC and the HPC were fixed at 3 and 11, respectively. Although fewer stages for the HPC could be used for the lower pressure ratios, this would result in lower compressor efficiencies and, therefore, a higher TSFC. Two axial stage turbines were used for the high pressure turbine (HPT) and five for the low pressure turbine (LPT). Bypass ratio was optimized for each cycle with respect to TSFC. Turbine inlet temperatures (T_{41}) between 2760 and 3085 °R have a small effect on TSFC (fig. 4) due in part to the aggressive component efficiencies for the advanced engine. However, increasing the OPR from 40 (current technology) to 100 results in about an 8.5 percent decrease in TSFC. The effect of turbine temperature on specific thrust (based

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