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Gas Turbine Characteristics for a Large Civil Tilt-Rotor (LCTR)

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Abstract

In support of the Fundamental Aeronautics Program, Subsonic Rotary Wing Project, an engine system study has been undertaken to help define and understand some of the major gas turbine engine parameters required to meet performance and weight requirements as defined by earlier vehicle system studies. These previous vehicle studies will be reviewed to help define gas turbine performance goals. Assumptions and analysis methods used will be described. Performance and weight estimates for a few conceptual gas turbine engines meeting these requirements will be given and discussed. Estimated performance for these conceptual engines over a wide speed variation (down to 50 percent power turbine rpm at high torque) will be presented. Finally, areas needing further effort will be suggested and discussed.

Nomenclature

C.G.	center of gravity, inches
eff	efficiency
fps	feet per second
ft	feet
hp	horsepower
HPC	high-pressure compressor
HPT	high-pressure turbine
lbm	pounds mass
LCTR	Large Civil Tilt Rotor
LP	low pressure
LPC	low-pressure compressor
LPT	low-pressure turbine
Max	maximum
N	actual speed, rpm
Nc	corrected speed, $N/\sqrt{\theta}$, rpm
PR	pressure ratio
PSFC	power specific fuel consumption, lbm/hr/hp
PT	power turbine
sec	second
T3	compression system exit temperature, °F
T4	combustor exit temperature, °F
Vtip	rotor tip velocity, feet per second
W	actual mass flow, lbm/sec
Wc	corrected mass flow, $W*\sqrt{\theta}/\delta$, lbm/sec
Wturb	power turbine actual mass flow, lbm/sec
δ	ratio of actual to standard pressure
θ	ratio of actual to standard temperature

Introduction

The NASA Heavy Lift Rotorcraft System Investigation (Ref. 1) identified a large tilt rotor as the best concept to meet the various airspace and other requirements for the future, short-haul regional market. This evolved into a conceptual vehicle designated as LCTR2 (Large Civil Tilt Rotor—iteration 2) (Ref. 2) as seen in Figure 1

This vehicle iteration was designed to carry 90 passengers at 300 knots with at least a 1,000 n mi range; powered by four turboshaft engines designed for 7,500 hp each. Other design features included a rotor tip speed of 650 ft/sec in hover and 350 ft/sec during cruise, enabled by a two-speed gearbox. This range of rotor tip speeds was needed to achieve the high level of performance and efficiency at two very different flight conditions. The rotor tip speed variation could theoretically be obtained using a variable diameter rotor or multiple-speed gearboxes (or a combination of these or other approaches); this work is focusing on achieving all speed variation from the engine. Although the exact requirements and characteristics for such a vehicle class are still being researched, performing analyses on a representative vehicle will help understand the sensitivities for such a design, help guide research efforts to reduce risks, and develop a suite of technologies from which this new vehicle class and capability can evolve and be developed. The final vehicle design could use one or a combination of these variable rotor tip speed concepts, determined from the vehicle's specific design and mission requirements and the state of these various required technologies.

Figure 1.—Conceptual view of LCTR2.

This report details gas turbine engine technology assumptions and analyses performed to estimate the engine parameters needed to obtain sufficient performance to meet operational goals for the proposed vehicle concept. This vehicle would use the turboshaft version of a gas turbine engine. The core gas turbine engine develops high energy (high temperature and pressure) gas to power a separate power turbine and shaft that supplies horsepower and torque to the main drive system. Initial cycle parametrics were performed to suggest gas turbine engine characteristics that would meet LCTR2 performance requirements. Based on these overall characteristics, thermodynamic engine analyses assuming a one-spool core (all compression on one shaft) and a two-spool core (with the compression split between two sets of compressors, each on their own shaft) were performed. As stated above, both configurations have a separate power turbine and shaft. For the thermodynamic analyses, compressor component performance maps were generated and used to estimate off-design performance at the hover and cruise flight points. At the cruise condition, gas turbine engine performance was estimated at 100, 75 and 50 percent power turbine rpm to quantify engine performance at reduced power turbine speeds. Results will be discussed and suggestions for further analysis will be given. Follow-on studies are underway to perform more detailed analyses of compression and turbine systems and component performance. It is expected that the more detailed component analyses will be incorporated and reported in subsequent studies and reports. It must be noted that this work is not expected to identify the specific gas turbine engine attributes and cycle definitions to meet all requirements. Its purpose is to further refine requirements and identify possible components, systems or subsystems that could enable such new classes of engine and vehicle designs and operations while also uncovering areas requiring further exploration and development.

Analysis Methodology

Engine system studies were performed to estimate the major gas turbine engine parameters that would meet performance requirements for the previously defined vehicle and mission. Engine power, weight and fuel consumption are important performance parameters that will help define possible engine configurations. As part of the parametric analysis, compressor pressure ratio (PR) was varied from 5 to 60, assuming a constant polytropic efficiency of 88 percent. Although turbomachinery efficiency would vary depending on engine size and configuration, that effect was deferred to later analyses. Combustor exit temperature was varied from 2000 to 3200 °F (in 400° increments). To get turbine cooling bleed estimates for the core turbines, the method of Gauntner (Ref. 3) was used, assuming metal temperatures of 2200 °F for the stator, 2100 °F for the rotor. These turbine metal temperatures are higher than present, small gas turbines to include the effects of incorporation of improved turbine material temperature capabilities and cooling techniques in these smaller turbine sizes. The power turbine was assumed to be uncooled. These should be reasonable temperatures and efficiencies for engines in the LCTR2-size class with entry-in-service in roughly the 2020 timeframe.

The object-oriented analysis framework, the Numerical Propulsion System Simulator (NPSS) (Ref. 4), was used to perform the gas turbine analyses. NPSS contains standard 0/1-D elements for the gas turbine components. These are configured into a representative steady-state, thermodynamic model. An example block diagram representative of a one-spool core, turboshaft is shown in Figure 2. Further elements are defined to drive specific parameters to desired values and insure continuity of mass, momentum and energy. After initial cycle parameters were determined, CMGEN (Ref. 5) was used to generate compressor performance maps that would be more representative of the flow-speed characteristics for a given compressor PR and size during the off-design analyses.

The gas turbine flow path and weight were generated using the WATE (Ref. 6) program. Using the output from NPSS (mass flows, temperatures, pressures, velocities, etc.) and further user input, WATE sizes the various mechanical and flow components for the gas turbine engine, determining materials, dimensions and weights for the different components represented. As part of the process, WATE also produces a graphical representation that can be used to check for reasonable component dimensions and ensure that there are no discontinuities or sharp turns in the gas flow path. The results of these analyses also form the basis for more detailed follow-on studies.

Results and Discussion

Initial Engine Parametrics

The preliminary analysis is instructive to suggest engine parameters to meet vehicle and mission requirements. Figure 3 shows calculated Power Specific Fuel Consumption (PSFC—lbm/hr fuel per hp produced) versus high pressure compressor (HPC) PR and combustor exit temperature (T4). Also included in the graph are areas representative of the performance regions for the Honeywell Aerospace T55 and GE Aviation

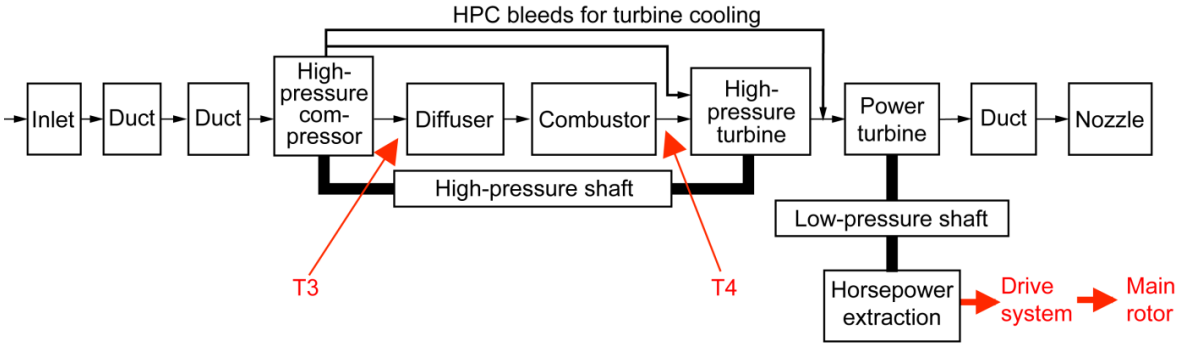

Figure 2.—Block representation of a one-spool core turboshaft gas turbine model.

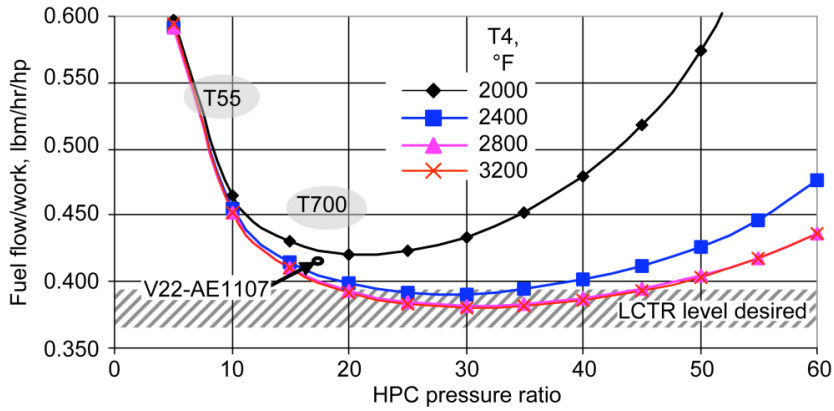

Figure 3.—Power Specific Fuel Consumption (PSFC, lbm/hr/hp) versus compressor pressure ratio and combustor exit temperature, T4.

T700 families of turboshaft engines, the version of Rolls Royce AE1107 in the Bell Boeing V-22 aircraft (all from Ref. 7) and the estimated level of PSFC used in the LCTR2 system studies. With the assumed technology levels, there is a minimum in the PSFC curves at a HPC PR of 30 (for values of T4 around 2800 to 3200 °F—both curves fall almost on top of each other). To meet the LCTR2 design of 7,500 hp per engine in that region of T4 and HPC PR would require an airflow of approximately 30 lbm/sec. Therefore, based on the technology assumptions, the engine parameters (at the sea level static design) used for subsequent analyses were: airflow of 30 lbm/sec, overall pressure ratio of 30, and T4 of 3000 °F. Table 1 compares the major parameters for this gas turbine to the Rolls Royce AE1107 (the engine in the Bell Boeing V-22 rotorcraft, a modern turboshaft engine in a slightly smaller power class). To meet the PSFC requirements, the notional study engine will need to operate at temperatures and pressures significantly higher than those found in present turboshaft engines, but at levels already in modern, large engines, while maintaining compressor and turbine efficiency and performance at the required much smaller airflows and blade sizes. A few conceptual engine configurations were investigated and will now be discussed in concert with their representative engine thermodynamic and flow path arrangements.

TABLE 1.—GAS TURBINE ENGINE PARAMETERS

Parameter	AE1107	Notional engine
Horsepower	6,000	7,500[a]
Weight, lbm	971	1,000[a]
Airflow, lbm/sec	35.3	30
PSFC, lbm/hr/hp	.426	.37[a]
Overall pressure ratio	16.7	30
Compressor exit temperature, T3, °F	810	1099
Combustor exit temperature, T4, °F	2200	3000
Corrected flow:		
Compressor entrance	35.5	30
Compressor exit	3.2	1.4

[a]Parameters are from Reference 2.

One-Spool Core Engine

The first engine to be modeled was a one-spool core (with a free turbine on a second spool) with an all-axial compressor. This allowed the engine flow path to be laid out and to generate compressor component performance maps to do an initial check on off-design performance. Compressor maps were generated at the design pressure ratio and flow. Two mission profiles were evaluated, each with a hover condition and a range of conditions at cruise. Gas turbine performance was calculated at the key points (hover and initial cruise power), to verify sufficient engine horsepower was available at full power turbine rpm (hover) and at

100 percent, 75 percent and the reduced power turbine mechanical rpm identified for optimum rotor performance at cruise. Results are given in Table 2 detailing the flight condition and pertinent turbomachinery operating characteristics. This preliminary analysis indicated that the gas turbine was able to achieve horsepower requirements for all flight points except one. There was insufficient engine horsepower for the mission 2 initial cruise point (operating at higher than the 300 knot requirement at reduced rotor / power turbine rpm). However, the engine was able to meet vehicle power requirements after about 1/3 of cruise fuel was consumed, reducing the vehicle weight and power needed. It was also found that at the cruise condition, 100 percent power turbine mechanical speed actually results in an aerodynamic over speed as a result of the drop in ambient temperature with altitude. This suggests that even if a multispeed gearbox was used, power turbine rpm could need to be reduced for cruise operation.

TABLE 2.—OPERATION OF ONE-SPOOL CORE TURBOSHAFT ENGINE OVER REFERENCE MISSION PROFILES AT VARIOUS POWER TURBINE RPMS

Altitude, ft	Speed, knots	Vtip, fps (rotor)	ΔT, °F	Meet hp?	Core rpm (actual/design)	HPC Nc	HPT Nc	T4, °F	PT Nc	HP / Wturb[a]	Wc-PT (lbm/s)
0	0	650	+27	Yes	1.02	99.2	100.4	3091	98.5	261.7	12.54
Mission 1											
2,000	0	650	+45	Yes	0.99	95.6	101.0	2864	101.8	215.9	12.16
(100% PT rpm) 28,000	303.4	350	0	Yes	0.90	98.0	99.9	2367	111.7[b]	210.2	12.60
(75% PT rpm)				Yes	0.90	98.1	100.3	2352	84.1[b]	208.4	12.86
(50% PT rpm)				Yes	0.91	99.3	100.5	2405	59.9	198.9	13.21
Mission 2											
5,000	0	650	+36	Yes	0.99	97.0	100.6	2870	101.9	227.1	12.31
(100% PT rpm) 28,000	330	350	0	Yes	0.92	99.1	99.7	2464	109.7[b]	228.7	12.71
(75% PT rpm)				Yes	0.92	99.4	100.2	2459	82.5[b]	225.5	13.00
(50% PT rpm)				*NO*	*0.93*	*100.0*	*100.4*	*2480*	*59.1*	*208.8*	*13.26*
(50% PT rpm) End 28,000	330	350	0	Yes	0.92	99.9	100.4	2472	59.2	207.9	13.26

[a]Actual power turbine horsepower /mass flow (hp/lbm/sec)
[b]Multispeed gearbox required to match rotor and power turbine speeds)

The operating points for the core turbomachinery and power turbine are shown in Figures 4 to 6. As can be seen in Figure 4 and Figure 5, the core compressor and turbine operate in a fairly narrow band over the flight profile. As shown in Figure 6, the power turbine has a much larger variation in its rpm. Although power turbine corrected flow varies less than 6 percent from its sea level design value, its pressure ratio increases about 15 percent from the full speed rpm at hover to the part speed rpm of the cruise. As shown in Table 2, the amount of engine power required per pound of actual power turbine mass flow is fairly constant for both missions at both flight points. This was not unexpected, since flight altitude and speed are determined by the engine power available (among other factors), with actual engine airflow and power falling with the increase in altitude and the accompanying drop in ambient density. However, it was hoped that power requirements per lbm airflow would drop to allow power turbine rpm to decrease as the mission progressed from hover to cruise, such that the main rotor rpm could also be reduced to maintain high rotor efficiency. For this analysis, at reduced rotor and power turbine rpm, the loss of turbine efficiency is compensated by an increase in pressure ratio to maintain power production. Maintaining a constant power turbine horsepower output per lbm airflow with an almost 50 percent reduction in rpm without loss of efficiency requires turbine design unique to this vehicle and mission class. A preliminary analysis looking at variable vanes and blades to meet this requirement is given in Reference 8. One such design concept includes power turbine variable incident nozzles, incident tolerant blades and additional stages. Whatever the design solution, further analysis is needed for the power turbine to verify its performance at almost constant power to weight flow over such a range of rotational speeds while still meeting engine weight goals.

Engine gas flow path dimensions and weights were generated assuming an all axial compression system; output from the WATE analysis is shown in Figure 7. It resulted in an engine plus accessories total weight of 946 lb (about 5 percent lighter than assumed in the LCTR2 study). At this engine airflow and horsepower class with one spool, all axial compression results in 10 stages to achieve the desired compression (tip speeds are below 1,000 ft per second for the last 4 stages). This specific application also results in very small blade heights for the latter stages (less than 1 in. for the last 4 stages, the last stage blade height is only 0.57 in.). This size blade could be a challenge for achieving and maintaining efficiency over the life of the engine. Further discussion on this particular design challenge is given in Reference 9. The power turbine WATE results assume typical turbine stage design. If additional power turbine stages using incident tolerant blades (and other variability) were needed to maintain good operation over its large speed range, these factors could increase power turbine weight.

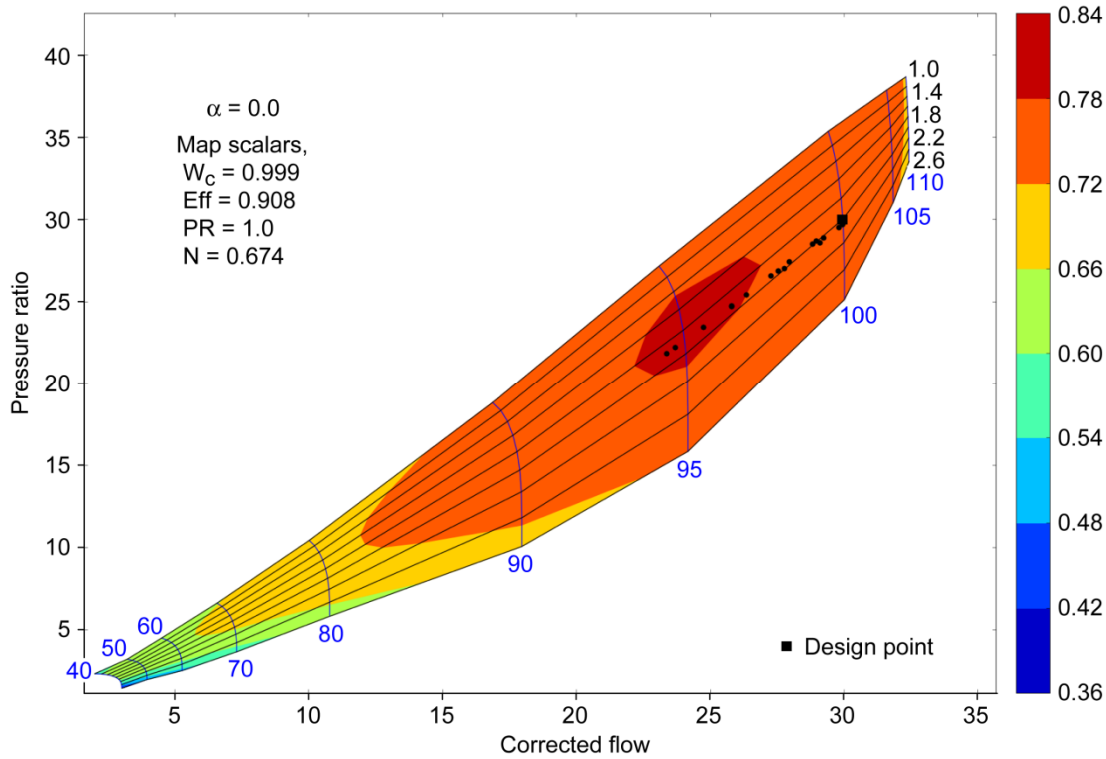

Figure 4.—One-spool core turboshaft engine compressor performance map and operating points.

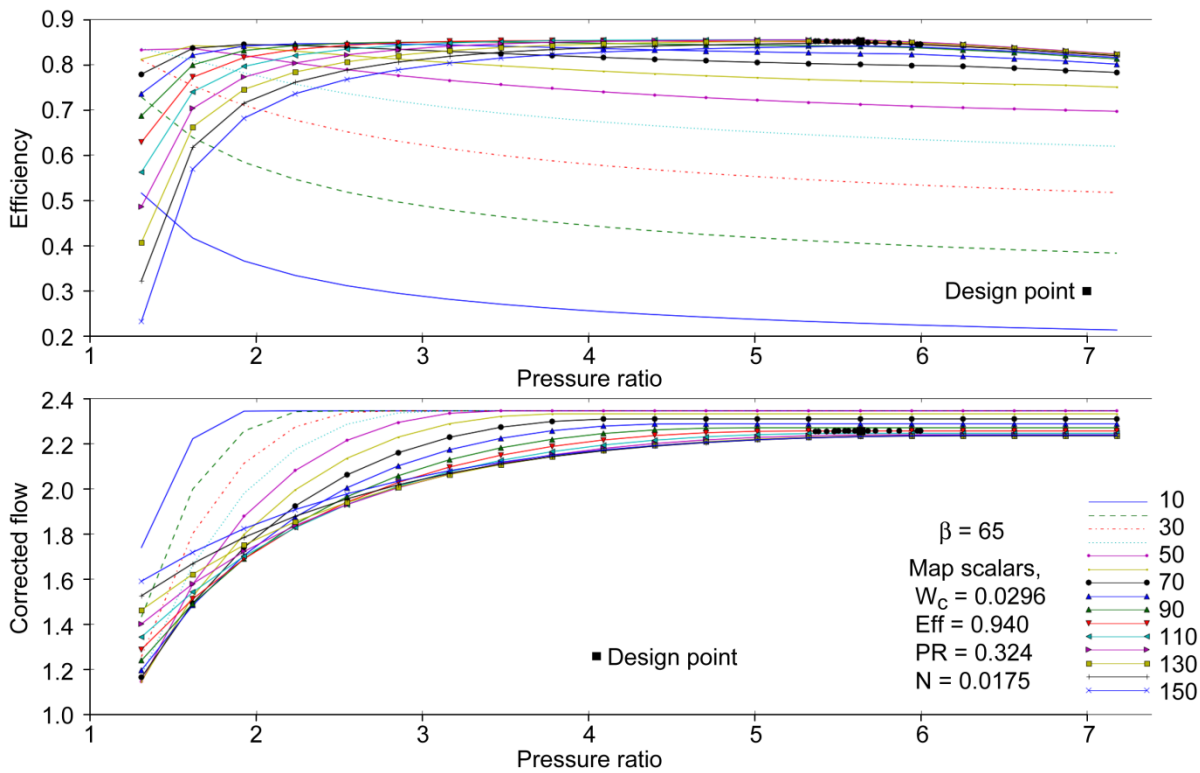

Figure 5.—One-spool core turboshaft engine turbine performance maps and operating points.

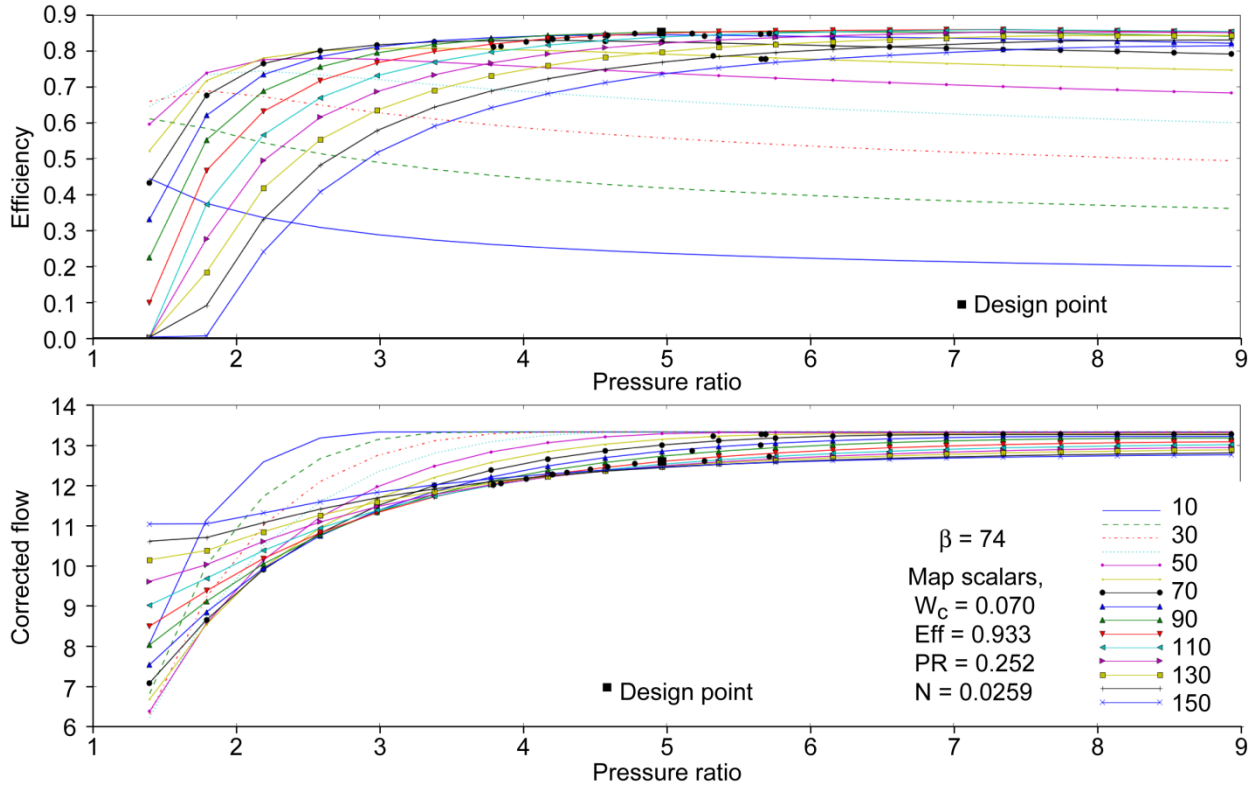

Figure 6.—One-spool core turboshaft engine power turbine performance maps and operating points.

Weights		Dimensions	
Bare engine weight	895.7	Engine length	63.1
Accessories weight	50.0	Engine pod C. G.	21.9
Engine weight	945.7	Engine max diameter	24.0
Inlet and nacelle weight	96.4	Nacelle max diameter	28.7
Total engine pod weight	1042.1	Total engine pod length	69.3

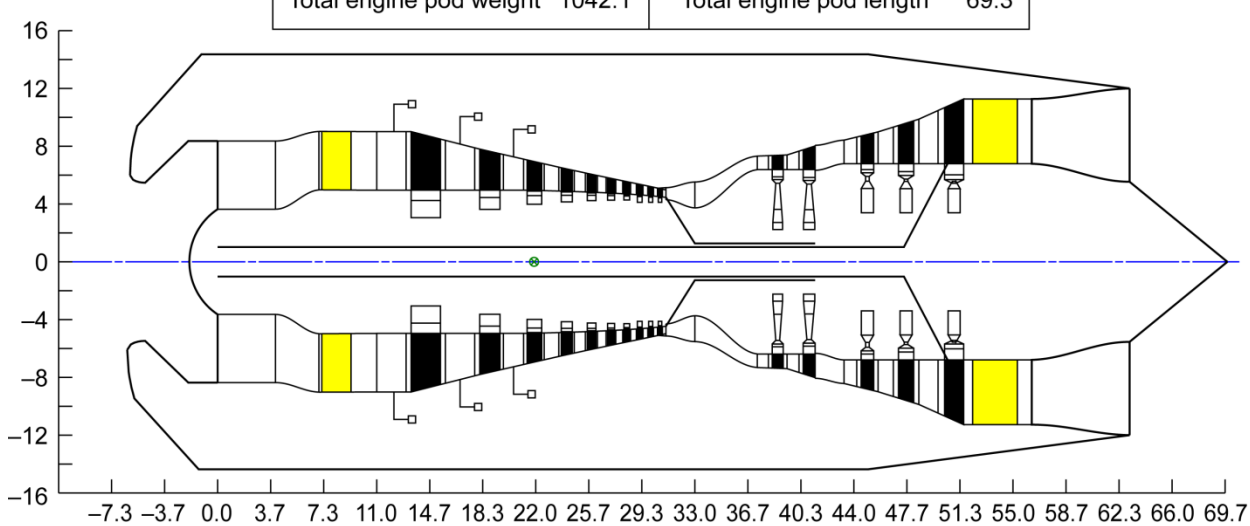

Figure 7.—One-spool core turboshaft engine WATE output.

Two-Spool Core Engine

Noting the challenging compressor design for a one-spool core, a two-spool core (with a free power turbine on a third spool) was modeled. A block representation is given in Figure 8. Compression was split with roughly equal enthalpy change for the low and high pressure compressors, which resulted in a low pressure compressor (LPC) pressure ratio of 9.2 and a high pressure compressor (HPC) pressure ratio of almost 3.3 (maintaining the same engine overall pressure ratio of 30). Results are given in Table 3 detailing the flight condition and pertinent operating characteristics for the low pressure spool turbomachinery and power turbine. This preliminary analysis indicated that the two-spool core turboshaft engine could achieve the horsepower requirements for all flight points (and power turbine mechanical rpms). Operation of the high pressure spool is essentially at constant corrected rpm and correct mass flow rates (less than 2.3 percent variation in the compressor and less than 1 percent in the turbine aerodynamic values). The high pressure spool had a very minor physical over-speed (1 percent above the design value) at hover, and was only operating at 90 to 97 percent of the physical design speed for cruise (the difference in corrected or aerodynamic versus physical rpm is from the reduction in component entrance temperatures going from hot-day hover to standard-day altitude conditions).

The operating points for the low pressure spool turbomachinery and power turbine are shown in Figures 9 to 11. As seen in Figure 9, the low pressure compressor corrected speed varies a maximum of 7 percent from the design condition; this maximum point is a result of the engine matching response to the hover condition on a hotter than average day (45 °F hotter at 2,000 ft altitude). As shown in Figure 10, there is also essentially no variation in the low pressure turbine corrected mass flow (<1 percent), although its pressure ratio varies slightly (–4 to 6 percent from the design value). The operation of the power turbine is very similar between this two-spool core engine (Figure 11) and the one-spool core version (Figure 6). This was expected with the similarity in vehicle power requirements and gas properties (mass flow, temperature and pressure) provided by the core of each engine; therefore, the previous analysis still applies.

TABLE 3.—OPERATION OF TWO-SPOOL CORE TURBOSHAFT ENGINE OVER REFERENCE MISSION PROFILES AT VARIOUS POWER TURBINE RPMS

Altitude, ft	Speed, knots	Vtip, fps (rotor)	ΔT, °F	Meet hp?	LP shaft rpm (actual/design)	LPC Nc	LPT Nc	T4, °F	PT Nc	HP/ Wturb[a]	Wc-PT (lbm/s)
0	0	650	+27	Yes	1.02	99.2	100.1	3103	98.3	263.9	11.96
Mission 1											
2,000	0	650	+45	Yes	0.97	93.6	99.5	2826	102.3	208.5	11.51
(100% PT rpm) 28,000	303.4	350	0	Yes	0.89	96.7	99.9	2319	112.7[b]	201.8	11.95
(75% PT rpm)				Yes	0.89	96.9	100.6	2295	85.1[b]	200.2	12.18
(50% PT rpm)				Yes	0.91	98.4	101.4	2333	60.8	190.5	12.54
Mission 2											
5,000	0	650	+36	Yes	0.97	95.3	99.7	2817	102.7	218.5	11.66
(100% PT rpm) 28,000	330	350	0	Yes	0.91	98.0	100.1	2413	110.8[b]	219.5	12.06
(75% PT rpm)				Yes	0.91	98.5	100.8	2397	83.5[b]	216.3	12.32
(50% PT rpm)				Yes	0.93	100.3	101.6	2455	59.4	205.6	12.62
(50% PT rpm) End 28,000	330	350	0	Yes	0.92	99.2	101.5	2397	60.1	199.1	12.60

[a]Actual power turbine horsepower /mass flow (HP/lbm/sec)
[b]Multispeed gearbox required to match rotor and power turbine rpms.

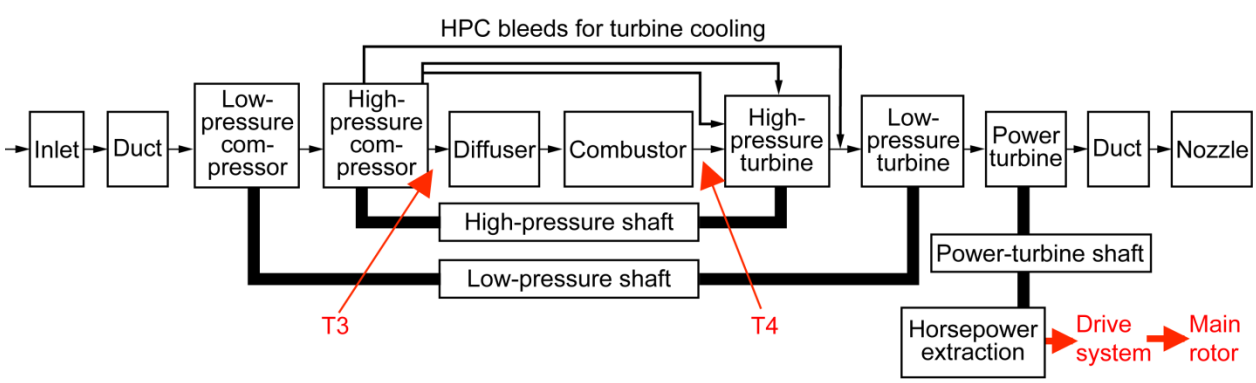

Figure 8.—Block representation of a two-spool core turboshaft gas turbine model.

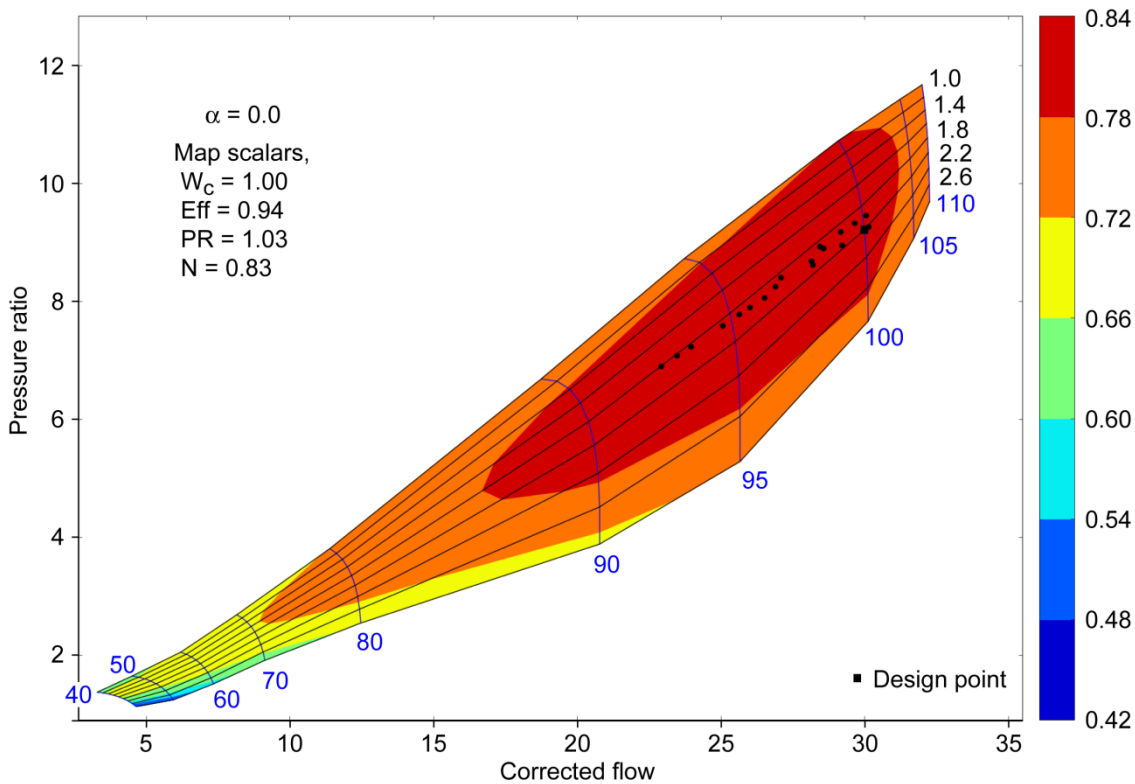

Figure 9.—Two-spool core turboshaft engine low-pressure compressor performance map and operating points.

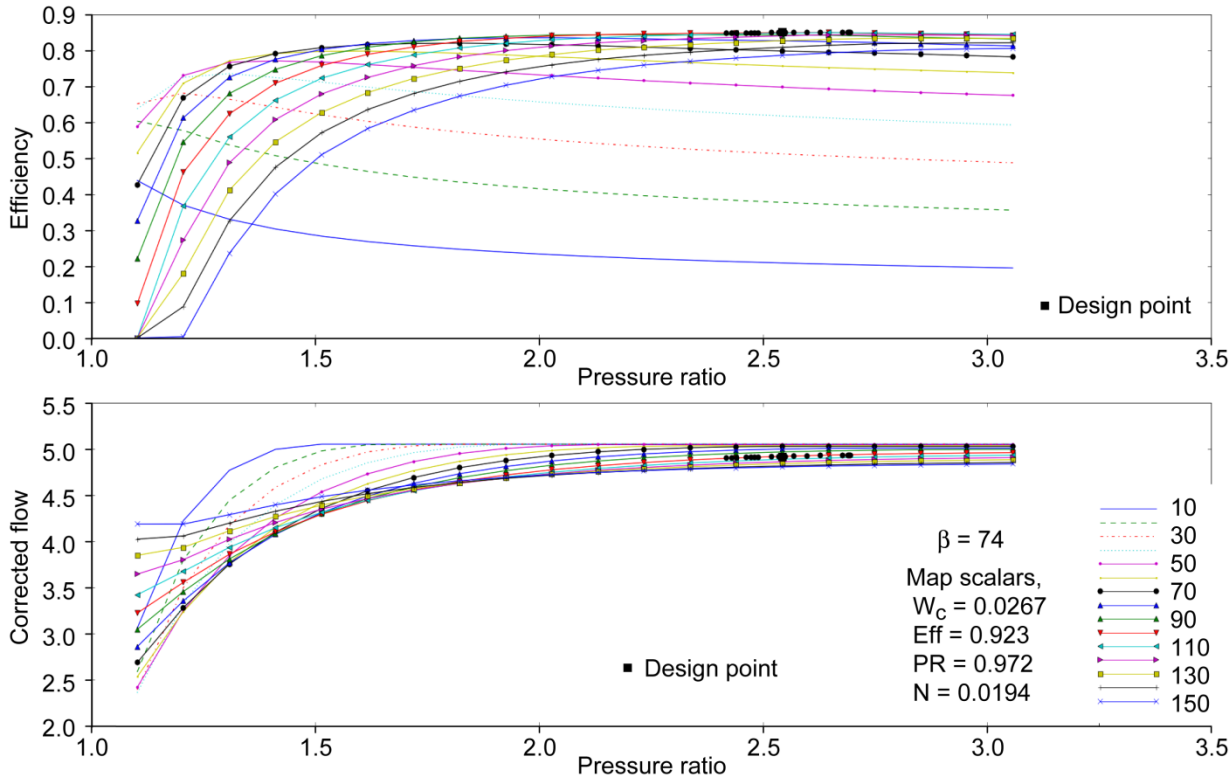

Figure 10.—Two-spool core turboshaft engine low-pressure turbine performance maps and operating points.

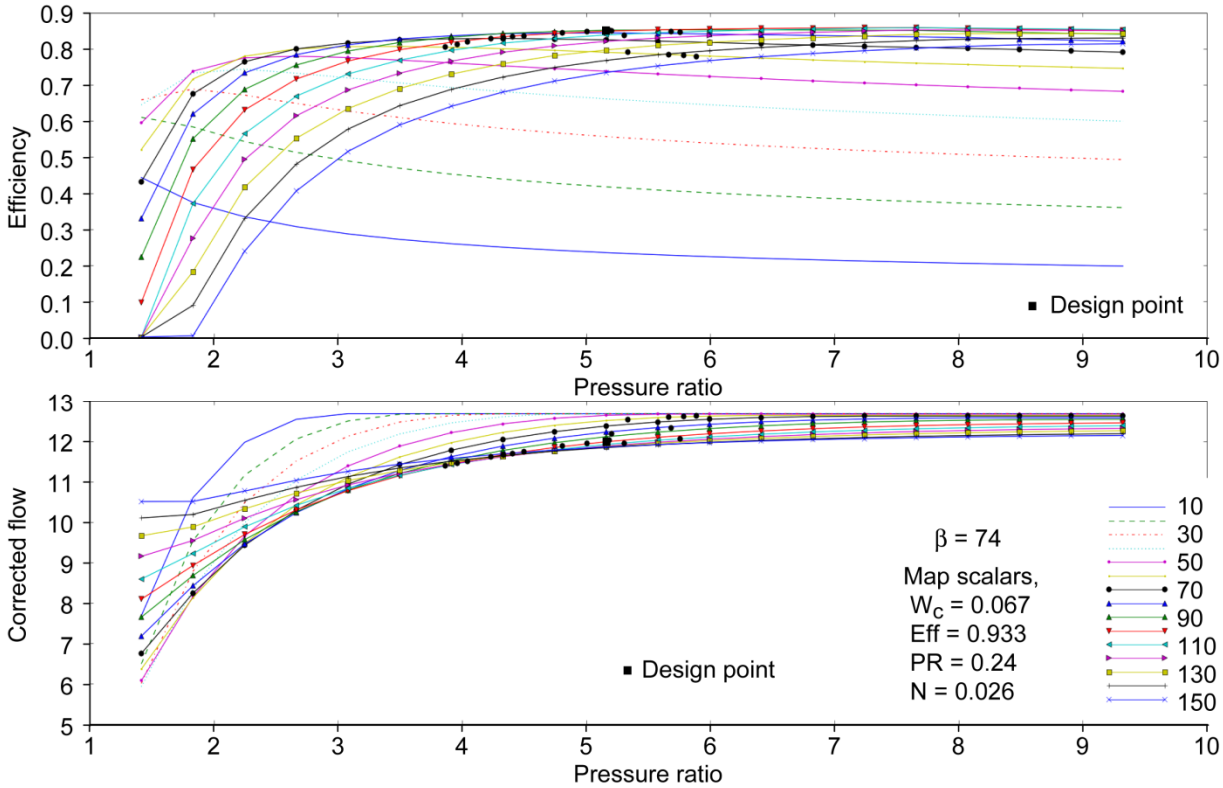

Figure 11.—Two-spool core turboshaft engine power turbine performance maps and operating points.

Engine gas flow path dimensions and weights were generated assuming an all axial compression system; output from the WATE analysis is shown in Figure 12. It resulted in an engine plus accessories total weight of 891 lb (about 11 percent lighter than assumed in the LCTR2 study). Separating compression into two separate spools allowed some reduction in compressor and turbine weight for the core engine, although the core total shaft weight increased (the sum of the two, lighter core shafts required weighed 80 lb more than the one core shaft of the one-spool concept). It also enabled a slightly larger compression exit blade height (0.65 in., almost 14 percent higher). Very small, latter stage blade heights are still an issue for this concept. This two-spool core concept also adds the complexity of a third shaft, which might be an issue considering the limitations on the third shaft diameter size available imposed by the other two shafts and torque. The third shaft might not be an issue for an aft turbine power assembly, if such an option can be reasonably integrated with the drive system.

To get away from the small axial stages at compression system exit, another engine model was investigated in which the axial high pressure compressor from the previous WATE iteration was replaced with a centrifugal stage. Although this centrifugal stage is operating at higher temperatures than centrifugal stages in present engines, it is worthwhile to understand centrifugal stages with increased temperature capability might enable good solutions to the LCTR2 engine requirements. As shown in Figure 13, it is very similar to the two-spool core, all axial, except for the high pressure compressor and combustor arrangement. Its weight at 1051 lb is a bit more than the other two engine concept weights, exceeding the LCTR2 weight requirement by 5 percent, and alleviates the small axial compressor blade height issue. The additional weight versus the two-spool, all axial engine is from the centrifugal compressor and combustor. Conservative, heavy materials were assumed, based on the high temperature environment for these components. This additional weight could be mitigated through better materials or careful engine flow path design, but those efforts are deferred to follow-on studies. The engine layout also has the same potential issue with its third shaft as previously discussed.

Weights		Dimensions	
Bare engine weight	841.1	Engine length	64.6
Accessories weight	50.0	Engine pod C. G.	17.8
Engine weight	891.1	Engine max diameter	24.0
Inlet and nacelle weight	96.8	Nacelle max diameter	28.8
Total engine pod weight	987.9	Total engine pod length	70.8

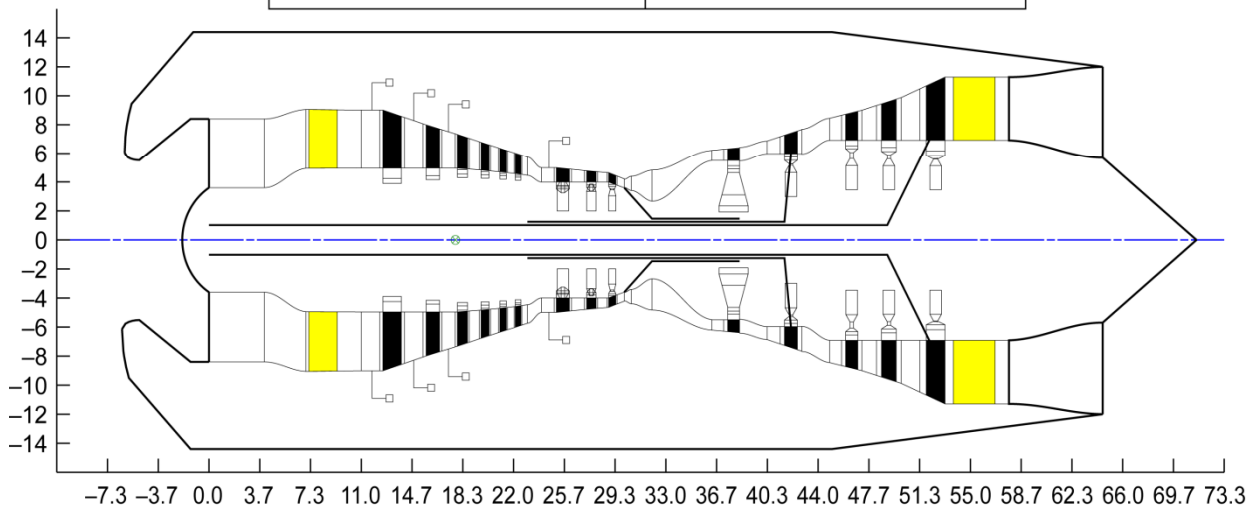

Figure 12.—All axial, two-spool core turboshaft engine WATE output.

Weights		Dimensions	
Bare engine weight	1000.5	Engine length	62.5
Accessories weight	50.0	Engine pod C. G.	19.1
Engine weight	1050.5	Engine max diameter	24.0
Inlet and nacelle weight	97.6	Nacelle max diameter	28.8
Total engine pod weight	1148.1	Total engine pod length	68.8

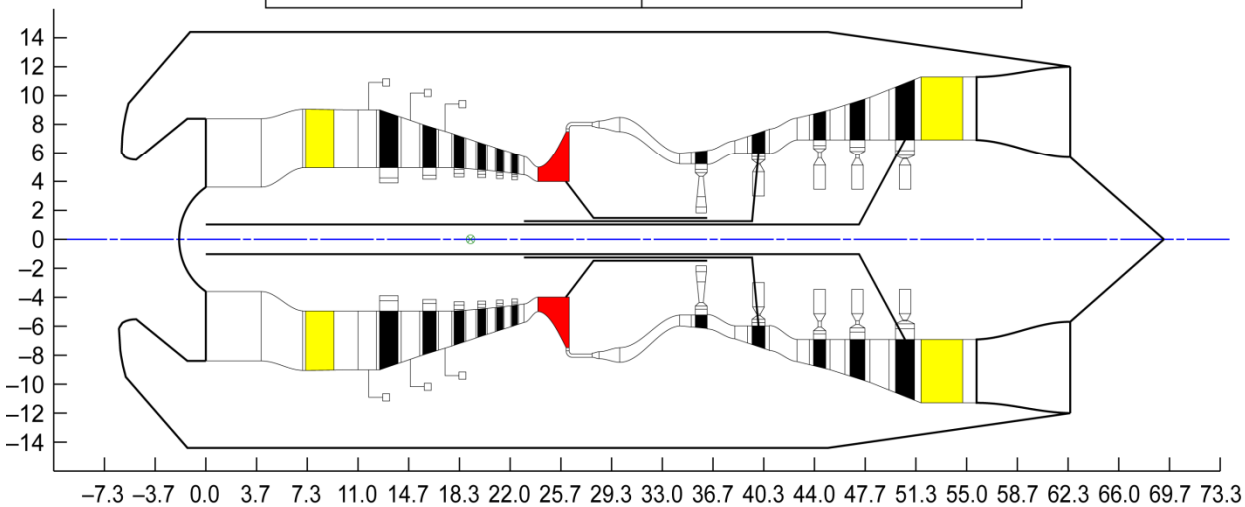

Figure 13.—Axicentrifugal high-pressure compressor, two-spool core turboshaft engine WATE output.

Conclusions

An engine thermodynamic, flow path and weight analysis has been performed to help define relevant areas of interest to achieve engine performance to meet LCTR2 performance goals and enable a new vehicle class and operations into its particular niche in the national airspace. Studies suggest that modern, large gas turbine engine temperatures and efficiencies are needed for the next generation of 7,500 hp class turboshaft engines. There has already been some recent work to look at different compression configurations for this specific engine. Further efforts are needed to guide research to develop and verify designs for axial and centrifugal compressors to operate at temperatures, efficiencies, and sizes beyond the present state of the art. Turbine technology will also be needed to maintain high efficiencies, reduced cooling requirements, and especially verify efficient operation over a wide power turbine speed range.

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13. SUPPLEMENTARY NOTES

14. ABSTRACT
In support of the Fundamental Aeronautics Program, Subsonic Rotary Wing Project, an engine system study has been undertaken to help define and understand some of the major gas turbine engine parameters required to meet performance and weight requirements as defined by earlier vehicle system studies. These previous vehicle studies will be reviewed to help define gas turbine performance goals. Assumptions and analysis methods used will be described. Performance and weight estimates for a few conceptual gas turbine engines meeting these requirements will be given and discussed. Estimated performance for these conceptual engines over a wide speed variation (down to 50 percent power turbine rpm at high torque) will be presented. Finally, areas needing further effort will be suggested and discussed.

15. SUBJECT TERMS
Turboshafts; Gas turbine engine; Performance prediction; Weight

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