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TURBOJET AND TURBOFAN CYCLE CONSIDERATIONS AND ENGINE CONFIGURATIONS FOR APPLICATION IN LIGHTWEIGHT AIRCRAFT

by Richard J. Roelke and Warner L. Stewart

Lewis Research Center
Cleveland, Ohio

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ABSTRACT

Engine studies were made to define cycle characteristics of small turbojet and turbofan engines for a selected set of lightweight aircraft flight requirements. From these cycle studies, representative engine conditions were selected and used as a basis for the examination of associated engine configurations. Emphasis was placed on the types of turbomachinery involved and their associated diameters.

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TURBOJET AND TURBOFAN CYCLE CONSIDERATIONS AND ENGINE CONFIGURATIONS FOR APPLICATION IN LIGHTWEIGHT AIRCRAFT

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Lewis Research Center

SUMMARY

The results of a study of the characteristics of small turbojet and turbofan engines for application in lightweight aircraft are presented in this report. Emphasis was placed on the types of turbomachinery involved and their associated diameters.

Cycle calculations were made for engine pressure ratios from 4 to 8 and turbine inlet temperatures of 1560° R (866.7 K) to 1960° R (1088.9 K). In addition, fan bypass ratios up to 5 and fan pressure ratios up to about 2.7 were considered. Engine compressor and fan diameters are indicated for a selected cruise thrust of 325 pounds (1445.7 N).

The study showed that the specific fuel consumption varied from a low of 0.9 pound fuel per pound thrust per hour (0.092 kg/(N)(hr)) for a high-bypass-ratio turbofan to a high of about 1.3 pounds fuel per pound thrust per hour (0.133 kg/(N)(hr)) for a low-pressure-ratio turbojet. For all engine cycles considered, the engine size was reduced significantly with an increase in turbine inlet temperature. The cycle penalties incurred when the fan pressure ratio was limited to 1.3 were small, which indicated that single-stage fans could be considered.

The gas-generator diameters of the turbofan engines were smaller than those of the turbojet. The fan diameter, however, dominated. The examination of low-bypass-ratio, single-shaft configurations indicated that turbomachinery of minimum size and tip speed could be achieved through the use of separate fan and compressor flows. A multiple-shafting arrangement was required for the high-bypass-ratio engines. Such a requirement, using a twin spool or gears, would increase the engine complexity, which must be weighed against the performance improvement thus obtained.

INTRODUCTION

As part of the gas-turbine technology program at the NASA Lewis Research Center, studies are being made to examine turbojet and turbofan engines for light-weight aircraft. These engines offer such potential advantages as compactness, light weight, and greater simplicity as compared with reciprocating or turboprop engines. In addition, improved aircraft performance in terms of cruise speed and rate of climb could be realized.

One of these studies was an analytical investigation of both these engine types for an example set of flight requirements. A parametric study was first made of the engine performance over a range of cycle conditions, including such variables as compressor pressure ratio and turbine inlet temperature. The range of these variables was restricted to reflect the desire for conservatism in the engine. In particular, the turbine inlet temperature range was restricted to 1960° R to avoid the requirement of exotic turbine materials or turbine cooling.

Based on these cycle studies, representative conditions were selected and used as a basis for the examination of associated engine configurations. The configuration study considered the types of turbomachinery involved and their associated diameters. Diameter was used as a preliminary gage to reflect the engine size.

The results of the study are presented in this report. Included are a description of the selected aircraft flight requirements, the engine-cycle characteristics as obtained for these requirements, and the results of the engine configuration study. In addition, since design simplicity is considered a desirable engine characteristic, a number of areas are indicated where improved engine performance must be weighed against the additional complexity involved.

ANALYTICAL METHOD AND ASSUMPTIONS

Whenever possible, the study calculations were made independent of a given aircraft configuration; however, certain data were required to perform the cycle calculations and to obtain an indication of the size of the engine parts. A cruise altitude of 25 000 feet (7620 m), and a Mach number of 0.65 were selected for the cycle calculations, and an engine thrust of 325 pounds (1445.7 N) was used to size the engine components.

The engine-cycle calculations made were limited to the cruise condition. It is recognized that for a given thrust at cruise the takeoff thrust would vary depending on the particular engine cycle and the matching of its components. For this study it was assumed that these variations could be accommodated by techniques such as adjusting the turbine inlet temperature at takeoff and allowing the takeoff distance to vary.

The engine-cycle calculation procedure was the same as that of reference 1, with one modification. The fan was assumed to be driven by a second turbine having an efficiency equal to the compressor-drive turbine instead of by one turbine driving both, as reported in the reference. The following engine variables were held constant for all the cycle calculations:

Inlet-diffuser pressure loss	0
Fan efficiency	0.85
Fan-duct pressure loss upstream of exhaust nozzle	0
Fan-duct-nozzle velocity coefficient	0.96
Compressor efficiency	0.80
Combustor efficiency	0.96
Combustor pressure loss	0.05
Turbine total efficiency	0.85
Jet-nozzle velocity coefficient	0.96

The cycle pressure ratios covered ranged from 4 to 8 and the turbine inlet temperatures from 1560° R (866.7 K) to 1960° R (1088.9 K). The cycle pressure ratio is defined as

TABLE I. - ASSUMED DESIGN PARAMETERS

Compressor	
Centrifugal:	
Slip factor	0.85
Specific speed	105
Axial-centrifugal (axial stage):	
Blade-tip speed, ft/sec (m/sec)	1150 (350.5)
Pressure ratio	1.5
Axial critical velocity ratio	0.6
Inlet radius ratio	0.45
Axial:	
Blade-tip speed, ft/sec (m/sec)	1150 (350.5)
Axial critical velocity ratio	0.6
Inlet radius ratio	0.45
Fan	
Blade-tip speed, ft/sec (m/sec)	1350 (411.5)
Axial critical velocity ratio	0.6
Inlet radius ratio	0.35
Turbine	
Blade-tip speed, ft/sec (m/sec)	1200 (365.8)
Axial critical velocity ratio at exit	0.7

the pressure rise from the compressor or fan inlet to the compressor discharge. Fan bypass ratios up to 5 were investigated, and fan pressure ratios up to the optimum for each bypass ratio were covered.

Determination of the general size and design characteristics of the fan, compressor, and turbine required that design parameters, such as blade speed, axial gas velocities, specific speed, and inlet radius ratio, be established. The values of these parameters are given in table I. The values listed in table I were chosen to represent typical engine design practices. In a detailed engine design, changes in some of these parameters could result in better engine configurations. Such changes were made in some of the representative engines that are discussed in the succeeding section.

RESULTS AND DISCUSSION

The analytical results for the turbojet and turbofan engines are presented separately. For each engine, the cycle characteristics are discussed and the compressor diameters are indicated. In addition, fan pressure ratios and sizes are given along with a number of turbofan engine configurations. In this study, emphasis was placed on component size and design simplicity.

Turbojet

The turbojet cycle characteristics are shown in figure 1. The specific fuel consumption ranges from a low of 1.08 pounds fuel per pound thrust per hour (0.110 kg/(N)(hr)) to a high of 1.36 pounds fuel per pound thrust per hour (0.139 kg/(N)(hr)) for the conditions studied. The lower fuel consumption occurs at the higher pressure ratios; however, the largest gains in reduced fuel consumption are obtained at the lower pressure ratios. Also, the optimum turbine inlet temperature, with respect to low fuel consumption, increases with increasing pressure ratio. At the lower pressure ratios, the optimum temperature is somewhat less than 1560° R (866.7 K), whereas for pressure ratios of 7 to 8, the optimum temperature is about 1660° R (922.1 K).

As the turbine inlet temperature is increased from 1560° R (866.7 K), the size of the engine is reduced, as indicated by the specific thrust. For example, at an engine pressure ratio of 6, the specific thrust is increased from 33.1 to 50.8 pounds thrust per pound air per second (324.6 to 498.2 N/(kg/sec)) for engine temperatures of 1560° R (866.7 K) and 1960° R (1088.9 K), respectively. This increase represents a 35-percent

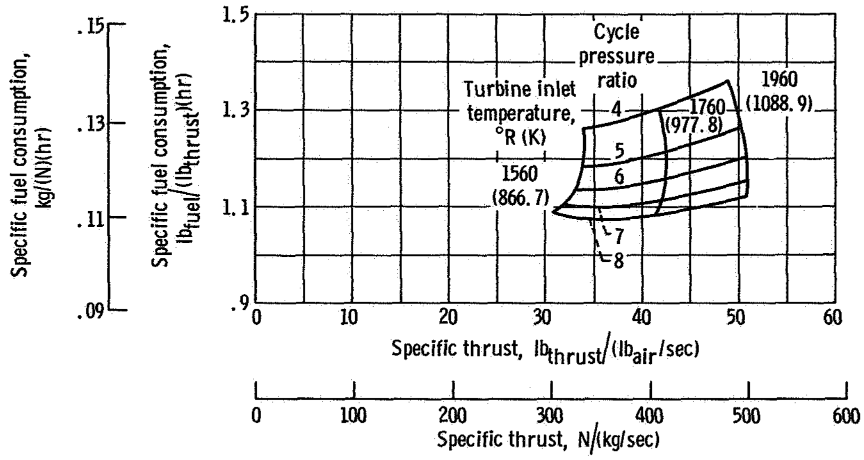
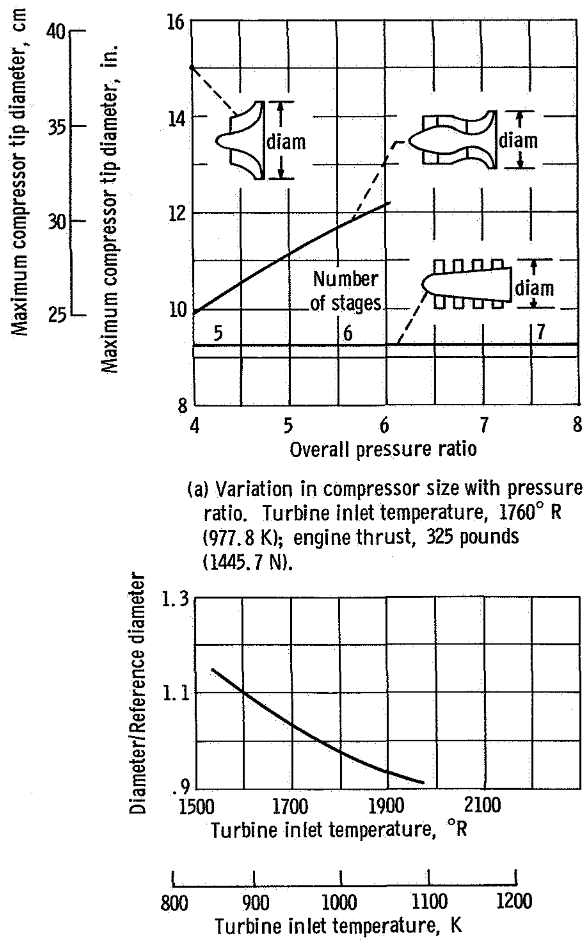


Figure 1. - Turbojet cycle characteristics at altitude of 25 000 feet (7620 m) and Mach number of 0.65.

reduction in engine airflow and consequently, in engine frontal area. With higher temperatures, however, improved high-temperature materials and/or additional cooling of the hot sections of the engine are necessary. Therefore, the turbine inlet temperature selected depends on balancing the reduced engine size with the added complexity required by higher temperatures.

The engine pressure ratio primarily influences the engine configuration and only slightly the engine airflow; that is, at the lower pressure ratios, the compressor may be a centrifugal, axial-centrifugal, or axial design. At the higher pressure ratios, only the axial compressor is considered applicable. The engine pressure ratio also affects the number of turbine stages. Utilizing present turbine technology and design practices, a single-stage turbine can drive a compressor having a pressure ratio up to 4.5 to 5. At engine pressure ratios above these values, a two-stage turbine is required.

The variation in compressor diameter, which is indicative of engine diameter, is shown in figure 2(a) as a function of compressor pressure ratio. The compressors are sized for an engine having a cruise thrust of 325 pounds (1445.7 N) and a turbine inlet temperature of 1760° R (977.8 K). Three compressor types are illustrated: a single-stage centrifugal, a combined axial-centrifugal, and an axial compressor. The centrifugal compressor provides a simple, rugged design but is limited to a pressure ratio of about 4. Also, the compressor-rotor diameter is much larger (62 percent larger in comparison with the axial compressor) than the other compressor types at the same pressure ratio. Employing back-to-back centrifugal compressors reduces the diameter but presents the problem of ducting air to the rear compressor. Adding an axial stage ahead of the centrifugal stage significantly reduces the diameter and permits higher



(a) Variation in compressor size with pressure ratio. Turbine inlet temperature, 1760° R (977.8 K); engine thrust, 325 pounds (1445.7 N).

(b) Diameter change with temperature at pressure ratio of 6.

Figure 2. - Variation in turbojet compressor tip diameter with engine pressure ratio and turbine inlet temperature.

pressure ratios. At a pressure ratio of 4, the combined axial-centrifugal compressor has a tip diameter of 9.9 inches (25.15 cm) as compared with that of 15 inches (38.10 cm) for the single-stage centrifugal compressor. The smallest compressor diameter is obtained with an axial design. For example, the axial compressor is 7 to 31 percent smaller than the combined axial-centrifugal compressor at pressure ratios of 4 and 6, respectively. Also, with the axial compressor, the pressure ratio is not limited. In figure 2(a) the approximate number of stages for a given pressure ratio is indicated; at the higher pressure ratios, 7 to 8 stages are required along with a multistage turbine.

The effect of varying the turbine temperature on the component diameters of figure 2(a) is shown in figure 2(b). At a temperature of 1560° R (866.7 K), the diameter is

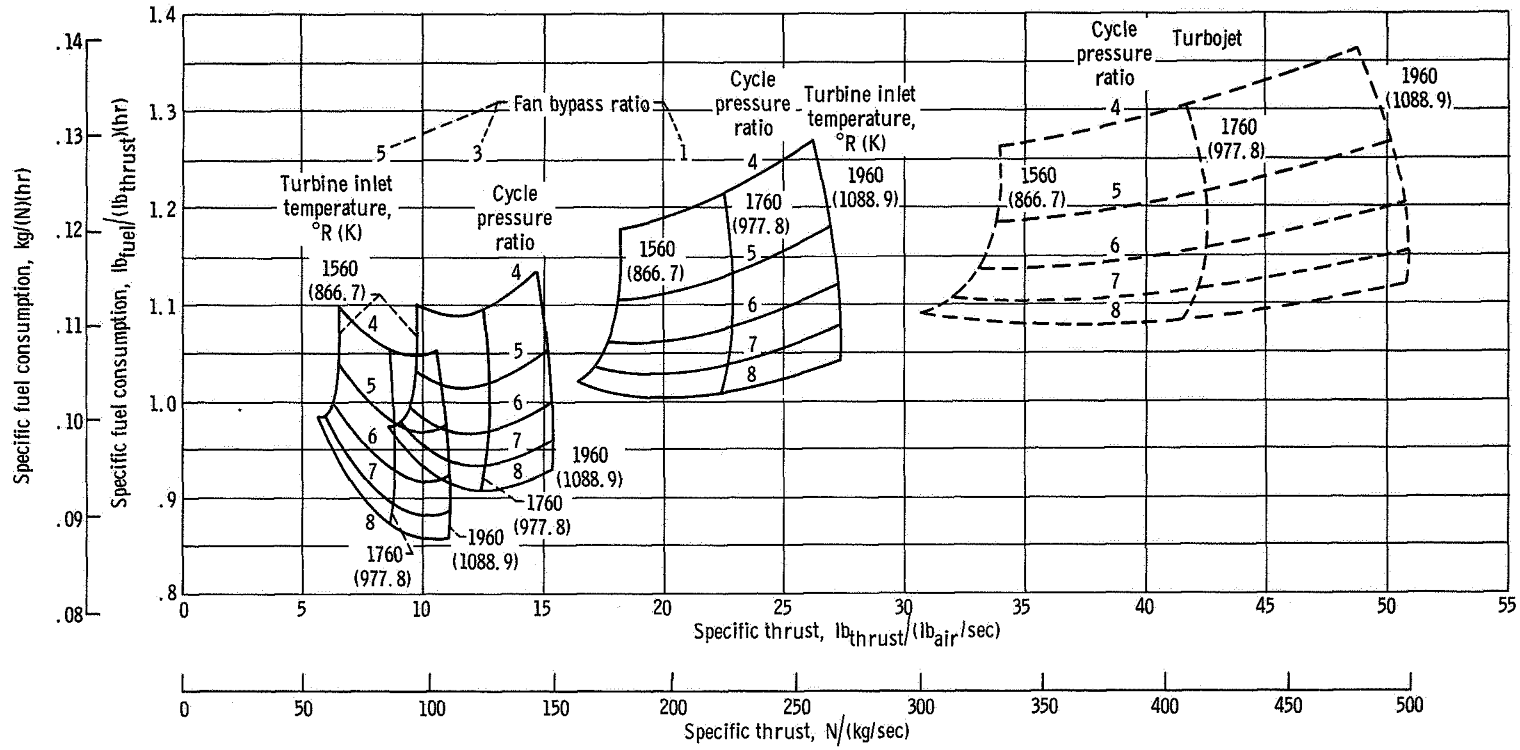


Figure 3. - Turbofan-engine-cycle characteristics at altitude of 25 000 feet (7620 m) and Mach number of 0.65. Fan pressure ratio is at thermodynamic optimum or at 1.3, whichever is smaller.

increased about 12 percent but is decreased about 9 percent at 1960° R (1088.9 K), which again shows the strong influence that temperature has on engine size.

Turbofan

The turbofan-cycle characteristics are shown in figure 3 with the turbojet data included for comparison. The fan curves are typical for that type engine; that is, an increase in the bypass ratio reduces the fuel consumption and increases the total engine flow. The specific fuel consumption ranges from a high of 1.27 pounds fuel per pound thrust per hour (0.130 kg/(N)(hr)) at a bypass ratio of 1 to a low of 0.86 pound fuel per pound thrust per hour (0.088 kg/(N)(hr)) at a bypass ratio of 5. At the intermediate bypass ratio of 3, the specific fuel consumption is between 0.9 and about 1.1 pounds fuel per pound thrust per hour (0.092 and 0.112 kg/(N)(hr)). The effects of turbine inlet temperature and engine pressure ratio are similar to those of the turbojet.

TABLE II. - VARIATION OF OPTIMUM FAN PRESSURE RATIO WITH TURBINE TEMPERATURE AND BYPASS RATIO

Turbine inlet temperature		Bypass ratio, Fan-duct air flow/Gas-generator air flow		
°R	K	1	3	5
		Fan pressure ratio		
1560	866.7	1.6	1.3	1.2
1760	977.8	2.0	1.4	1.3
1960	1088.9	2.3	1.6	1.4

Fan pressure ratio. - Each combination of turbine temperature and bypass ratio has an optimum fan pressure ratio. In general, the optimum fan pressure ratio increases with decreasing bypass ratio and with increasing temperature. The approximate level of this pressure ratio is given in table II. The higher fan pressure ratios would obviously require more than one fan stage and/or high blade speeds. Also, as the fan power requirement is increased, either by a higher pressure ratio or a higher bypass ratio, a multistage fan turbine is necessary. For these reasons, the fan pressure ratio was

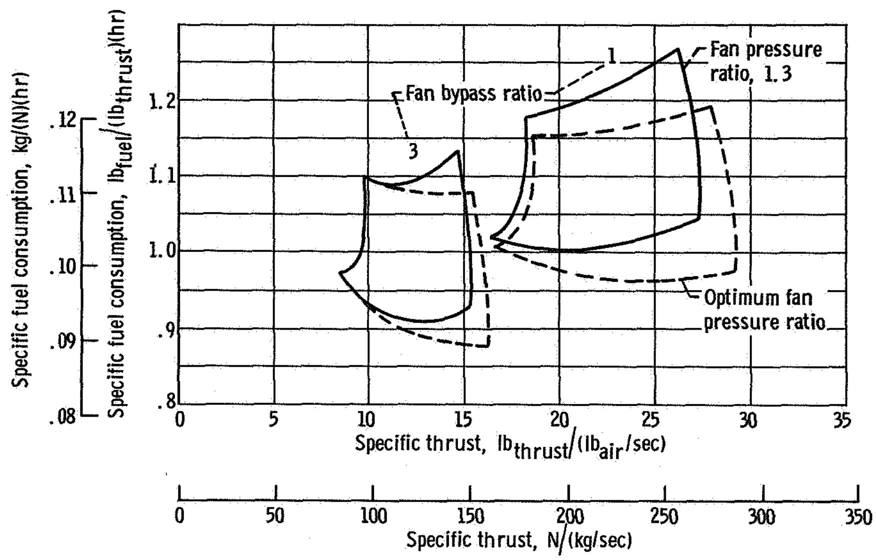


Figure 4. - Comparison of optimum turbofan-engine-cycle characteristics with turbofan having fan pressure ratio of 1.3. Altitude, 25 000 feet (7620 m); Mach number, 0.65.

limited to a maximum of 1.3. The engine performance penalty incurred by doing this is small (fig. 4); therefore, this is the preferred configuration for the fan.

Component size. - The variation in airflow (a measure of component size) with bypass ratio for a selected set of engine parameters is shown in figure 5. As the bypass ratio is increased, the gas-generator flow decreases only slightly, an indication that that

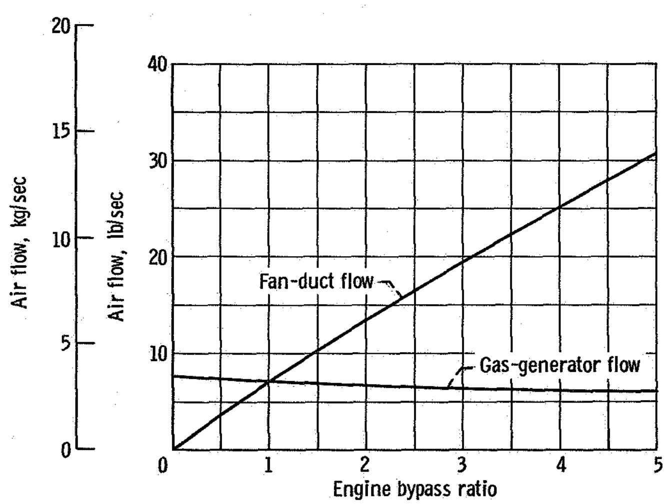


Figure 5. - Variation of engine air flow with bypass ratio. Cycle pressure ratio, 6; turbine inlet temperature, 1760° R (977.8 K); fan pressure ratio, 1.3; altitude, 25 000 feet (7620 m); Mach number, 0.65; thrust, 325 pounds (1445.7 N).

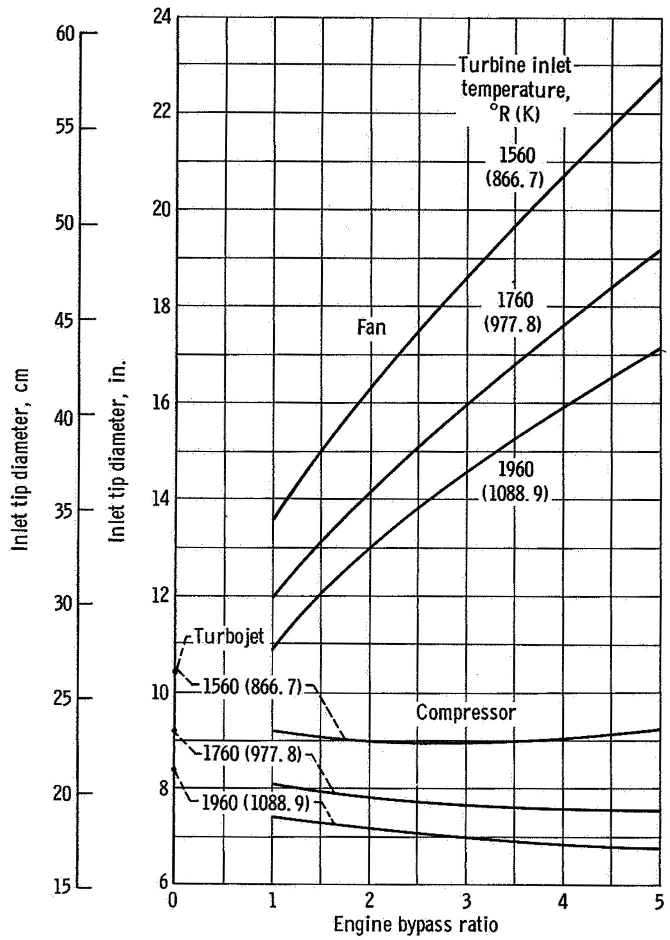
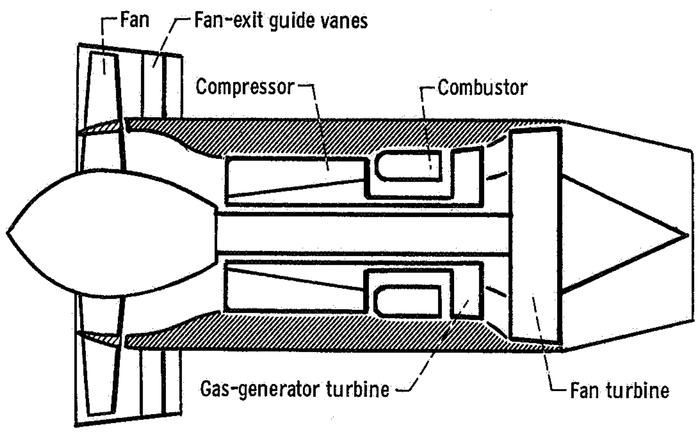


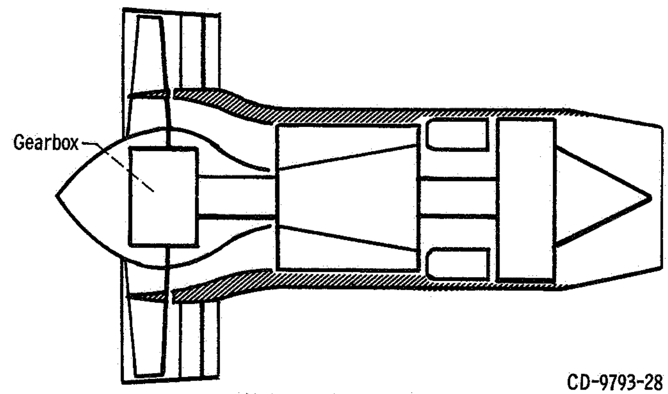
Figure 6. - Variation in axial-flow compressor and front fan size with bypass ratio and temperature for engines with cruise thrust of 325 pounds (1445.7 N) and pressure ratio of 6.

part of the engine is nearly fixed in size. The only varying parts of the engine are the fan and the fan turbine; therefore, one gas generator could be mated with a wide variety of fans.

The sizes of the compressor and the fan for a spectrum of fan engines are shown in figure 6. The engines were again sized to the cruise condition of 325 pounds (1445.7 N) thrust. The specific thrust for each engine cycle agrees with that shown in figure 3 for an engine pressure ratio of 6, and the particular configuration assumed was a front-fan, dual-shaft engine similar to that shown in figure 7(a). The fan size increased rapidly with bypass ratio, as expected, but was significantly affected by the turbine temperature. For example, at a bypass ratio of 3, the fan diameter is decreased about 14 percent by an increase in the temperature from 1560° R (866.7 K) to 1760° R (977.8 K). An additional 7-percent reduction is realized if the temperature is raised to 1960° R (1088.9 K).



(a) Dual-shaft turbofan.



(b) Geared fan turbofan.

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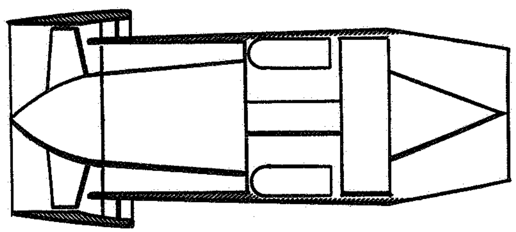
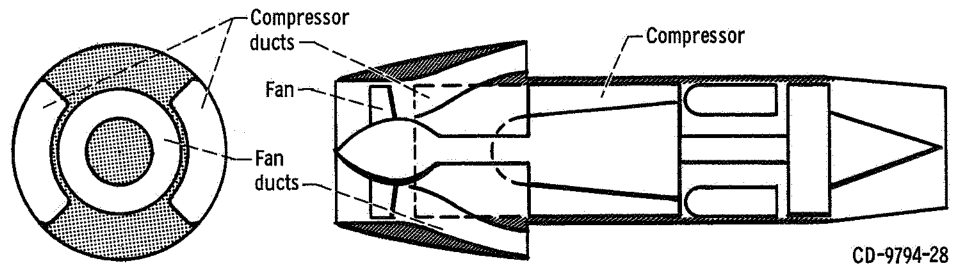
Engine configu-ration	Bypass ratio	Fan					Compressor					Turbine temper-ature		Gas-generator turbine diameter		Fan turbine diameter	
		Pres-sure ratio	Tip speed		Tip diameter		Pres-sure ratio	Tip speed		Tip diameter						in.	cm
			ft/sec	m/sec	in.	cm		ft/sec	m/sec	in.	cm	°R	K	in.	cm		
Dual shaft	5	1.25	1350	411.5	19.2	48.77	4.8	1150	350.5	7.6	19.30	1760	977.8	8.5	21.59	11	27.94
Geared fan[a]	3	1.3	1350	411.5	16.0	40.64	4.62	1100	335.3	8	20.32	1760	977.8	9.9	25.15	--	-----

[a]This engine has one turbine only.

Figure 7. - High-bypass turbofan engine configurations sized to cruise thrust of 325 pounds (1445.7 N).



(a) Single-shaft turbofan with total flow passing through fan.



(b) Single-shaft turbofan with separated fan and compressor flows.

Engine configuration	Bypass ratio	Fan					Compressor					Turbine			
		Pressure ratio	Tip speed		Tip diameter		Pressure ratio	Tip speed		Tip diameter		Inlet temperature		Tip diameter	
			ft/sec	m/sec	in.	cm		ft/sec	m/sec	in.	cm	°R	K	in.	cm
Single-shaft turbofan (total flow passes through fan)	1	1.3	1350	411.5	11.9	30.23	4.62	1070	326.1	9.4	23.88	1760	977.8	10.3	26.16
Single-shaft turbofan (fan and compressor flows separated)	1	1.3	1100	335.3	8.4	21.34	6	1150	350.5	8.8	22.35	1760	977.8	9.2	23.37

Figure 8. - Low-bypass turbofan engine configurations sized to cruise thrust of 325 pounds (1445.7 N).

Only multistage axial-flow compressor sizes are indicated in the figure; however, a combined axial-centrifugal compressor could also be considered for the fan engine. The increase in diameter of this combined unit would be similar to that given for the turbojet compressor. The axial compressor size is affected not only by the airflow and engine temperature but also by the fan pressure ratio. The front fan supercharges the air entering the gas generator, but not always to the same degree. As stated in the preceding section, the fan pressure ratio was 1.3 or less; thus for a temperature of 1560° R (866.7 K), the optimum pressure ratio is about 1.2 at a bypass ratio of 5. Therefore, although the gas-generator weight flow is almost constant between bypass ratios of 3 to 5, the size of the compressor increases because of the lower air density. This increased size is also the reason why the three turbojet points lie above the fan engine curves: the compressor has no supercharging.

Example fan engines. - Fan engine variations, in addition to the dual-shaft arrangement, were investigated and are shown schematically in figures 7 and 8. These engine configurations indicate some simplifications to the basic fan engine. The dimensions given in the figure are for an engine having 325 pounds (1445.7 N) thrust at cruise.

The dual-shaft configuration (fig. 7(a)) is a conventional fan-engine arrangement for a high bypass ratio. The engine has concentric dual shafts with the front fan driven by a second turbine. An alternative to the double shafts is to gear the fan to the compressor shaft, as shown in figure 7(b) for an engine with a bypass ratio of 3. These engines have a comparatively low fuel consumption (fig. 3) and small, compact gas generators, but these advantages must be weighed against the added engine complexity indicated.

The configurations shown in figures 8(a) and (b) are for low-bypass-ratio engines having only a single shaft and no gearbox. In the engine shown in figure 8(a), the fan-tip speed is 1350 feet per second (411.5 m/sec), the same as that for the engine in figure 7(a), but the compressor- and turbine-blade speeds have been reduced even with an increase in diameter. This decrease in blade speed presents a more difficult design problem for the compressor and turbine, possibly necessitating additional stages in these components.

Some of these problems can be eliminated by separating the fan and compressor flows at the engine inlet, as shown in figure 8(b). This engine as shown has two sets of ducting. The fan flow enters the center ducting, is pressurized in the fan, and is accelerated in the fan nozzle. The gas-generator flow enters the compressor ducts, placed outside the fan ducts, and goes through a transition duct to the compressor inlet. Separating the flows of the fan and the gas generator permits the fan-tip speed to be decreased without posing a speed limitation on the compressor and turbine. The disadvantage of this engine is the additional ducting required.

Another possible engine arrangement that is not shown is the free-turbine aft fan. Because the total flow of the fan and gas generator passes through the fan-turbine rotor, this engine is limited to low and intermediate bypass ratios (up to about 3). A potential problem of this engine is sealing the fan turbine flow from the fan flow; however, with moderate engine temperatures and a small engine size, this sealing may not be a serious problem.

SUMMARY OF RESULTS

Turbojet and turbofan cycle characteristics and engine configurations were studied for application to lightweight aircraft. The engine configuration study considered the types of turbomachinery involved and their associated diameters. The following results were obtained:

1. The specific fuel consumption varied from about 1.1 to 1.3 pounds fuel per pound thrust per hour (0.112 to 0.132 kg/(N)(hr)) for the turbojet engine over the range of conditions studied. The corresponding range of specific fuel consumption for the turbofan engine at moderate to high bypass ratios was 0.9 to 1.1 pounds fuel per pound thrust per hour (0.092 to 0.112 kg/(N)(hr)).

2. For all engine cycles considered, the engine size was reduced substantially by an increase in the turbine inlet temperature. However, the final selection of temperature must be made not only with this consideration but also with that of the possible additional engine complexity required by the high temperature.

3. The cycle penalties incurred in limiting the fan pressure ratio to 1.3 for the turbofan engine were small, which indicated that single-stage fans could be considered over the range of conditions investigated.

4. At corresponding pressure ratios, the axial-flow compressor had the minimum rotor diameter, the combined axial-centrifugal was 10 to 30 percent larger, and the single-stage centrifugal was about 60 percent larger.

5. The gas-generator diameters were smaller for the fan engines than for the turbojet. The fan diameter, however, dominated.

6. The examination of low-bypass-ratio, single-shaft configurations indicated that turbomachinery of minimum size and tip speed could be achieved by the separation of fan and compressor flows, although additional ducting would be required.

7. A multiple-shafting arrangement was required for the higher bypass-ratio engines. Such an arrangement or a single shaft with gears increases the engine complexity. This factor must be weighed against the performance improvement thus obtained.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, April 23, 1968,
126-15-01-23-22.

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