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RESEARCH MEMORANDUM

for the

Air Research and Development Command, U. S. Air Force

COMPARATIVE STUDY OF TURBOFAN AND TURBOJET ENGINES

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and David S. Gabriel

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NATIONAL ADVISORY COMMITTEE
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INTRODUCTION

At the request of the Air Research and Development Command, U. S. Air Force, turbofan and turbojet engines were compared for all missions of current interest. This study is restricted to applications for flight Mach numbers up to 3.0 and for conventional hydrocarbon fuels. The report is presented in four parts:

(1) Cycle analysis of turbofan engines; descriptions of appropriate turbofan-engine designs for various missions and comparison of cycle performance of these engines with appropriate turbojet engines

(2) Comparison of several commercial engines proposed to the Air Force; comparison of these engine with predicted cycle performance

(3) Component performance and development problems for turbofan and turbojet engines

(4) Summary and conclusions based partly on mission studies.

Although the precision required for a preliminary study such as this is somewhat controversial, in the time available and in keeping with the objective of searching for areas of clear or outstanding advantages of turbofan powerplants, a precision of the order of 10 percent was considered adequate. Accordingly, absolute optimums in engine design were not sought. Much of the information is considered proprietary.

SYMBOLS

F_n net thrust, lb

M Mach number

P total pressure

W weight, lb

w weight flow, lb/sec

δ ratio of pressure to NACA standard sea-level pressure of 2116 lb/sq ft

Subscripts:

a air

e engine

g gross

s secondary

t total

1 compressor inlet

2 compressor outlet

CYCLE ANALYSIS

The turbofan engine has a bewildering variety of configurations and cycle variables. The confusion of using a large number of these configurations is avoided by selecting only engine designs which, although not necessarily optimum for each application, are sufficiently near optimum that useful and general comparisons of turbofan and turbojet engine performance may be made.

Most of the results presented have been extracted from references 1 and 2 and from unpublished NACA calculations. These two references appear to be the most reliable and consistent of those currently available.

Engine Design Variables

Turbofans may have numerous configurations. With regard to the exhaust system, the fan and turbine discharges may be mixed or unmixed. The compressor may be single- or two-spool. The fan may be separate from or integral with the primary compressor, or somewhere in between. The turbine drive may be separated in various degrees. Fortunately, most of these permutations have only a secondary effect on performance. Choice of configuration details, such as single- or two-spool compressors, turbine arrangement, and so forth, has no direct effect on cycle performance, but depends more on the required mission. For example, a two-spool design would probably be best for a high-pressure-ratio engine intended for subsonic applications, and a one-spool design for a low-pressure-ratio compressor designed for an all-supersonic mission. In this sense, then, the configuration details are not important for this study, with the exception of the question of a mixed or unmixed exhaust.

Thermodynamic performance is affected considerably by the choice of exhaust systems. At very high flight speeds the unmixed jet may be slightly superior in thrust and as much as 5 to 10 percent superior in specific fuel consumption to the mixed-exhaust turbofan if weight and cooling losses for the tailpipe are neglected. It appears, however (ref. 1), that the losses in performance and the weight penalty for cooling the inner duct separating the coannular exhaust system in the unmixed-exhaust engine will probably more than compensate for the cycle advantages of this engine configuration. In addition, substantial problems exist in development of efficient coburners and light, durable, variable-area exhaust systems for the unmixed engines. For these reasons, only the mixed jet will be considered in the following discussion. It may be concluded that the other details of configuration or component arrangement have only secondary performance effects in turbofan engines.

Cycle Design Variables

The selection of cycle design variables is of prime importance. These variables for the turbofan engine are as follows:

(1) Over-all compressor pressure ratio

(2) Turbine-inlet temperature

(3) Afterburner-outlet temperature

(4) Fan pressure ratio

(5) Ratio of secondary to total weight flow (bypass ratio)

For the mixed-exhaust turbofan, it is usually assumed that the mixing of the turbine and fan discharge air occurs at constant static pressure. Since the exit Mach numbers of the fan and turbine are usually assumed to be about equal (for low mixing loss), the fan- and turbine-discharge total pressures must be about equal. Consequently, for any given design of the primary cycle the relation between bypass ratio and fan pressure ratio is specified. That is, the work supplied to the fan is fixed by the combination of primary cycle pressure ratio and fan-discharge pressure, so that only certain combinations of fan pressure ratio and airflow that satisfy this work output may be used. Therefore, the fan pressure ratio and bypass ratio are not independent variables.

In the following discussion the effect of each of the design variables at low and high speeds is briefly described and reasonable values are selected for various missions.

Subsonic Performance

The design-point cycle performance of turbofan and turbojet engines at sea-level static conditions and at Mach 0.9 in the tropopause is shown in figures 1 to 3. As evident in figure 1, the thrust is largely unaffected by over-all engine pressure ratio, but the specific fuel consumption of both engines improves as pressure ratio increases up to about 11 or 12. For purely subsonic applications, pressure ratios around 11 are therefore preferred. Effects of over-all pressure ratio at Mach 0.9 in the tropopause are similar to the sea-level static effects, as shown in figure 2.

Figure 3 illustrates the effect of turbine-inlet temperature at Mach 0.9 in the tropopause. At constant bypass ratio, increasing turbine-inlet temperature increases both thrust and specific fuel consumption for the turbofan and the turbojet at slightly different rates. Because the increase in thrust tends to be compensated by an increase in specific fuel consumption, raising the turbine-inlet temperature will have only a small effect on the performance of subsonic airplanes with turbofan powerplants. Although it is not shown in figure 3, a slight reduction in turbofan specific fuel consumption can be achieved at high temperatures if the bypass ratio is increased with turbine-inlet temperature.

Figure 4 shows the variation of thrust per pound of airflow and specific fuel consumption for a Mach number of 0.9 at the tropopause for engines with an over-all primary-compressor pressure ratio of 10 and a turbine-inlet temperature of 2210° R. Both specific fuel consumption and specific thrust decrease as bypass ratio increases. Compared with a turbojet (0 bypass ratio), the turbofan with a bypass ratio of 0.8 has a 36 percent lower specific fuel consumption and a 70 percent lower specific thrust. A compromise between specific fuel consumption and engine size or weight is thus required. In general, most studies have shown that the best turbofan for these subsonic applications is one with a bypass ratio in the region from 0.5 to 0.7. The design fan pressure ratio would be about 2 for this range of bypass ratio.

It appears, therefore, that for subsonic turbofans an appropriate engine design will have an over-all primary-compressor pressure ratio of around 10±2, a fan pressure ratio of about 2.0 (both at sea-level static conditions), a bypass ratio of about 0.6, and a moderately high turbine-inlet temperature of around 2200° to 2300° R. Such an engine will have about a 25- to 30-percent advantage in specific fuel consumption over a turbojet engine of the same over-all compressor pressure ratio, but will be about twice as large in frontal area (although not necessarily in weight).

Specific Weight at Subsonic Conditions

The determination of weight is, of course, one of the most difficult of all parts of the engine comparison problem. The engine weights that were used to construct the curve shown in figure 5 were taken from references 1 and 2 and some of the engine proposals of the next division of this report. The turbofan engine weight is primarily a function of bypass ratio for a given compressor pressure ratio. At sea-level static conditions at a bypass ratio of 0.6, the specific weight of a turbofan is about 20 percent greater than for a turbojet with the same over-all compressor pressure ratio. This trend is accentuated at 0.9 Mach number in the tropopause, where the difference in engine specific weight is nearly 40 percent. The turbofan weighs considerably less per unit of frontal area, but at the subsonic flight conditions shown the low thrust per unit of frontal area evidently more than outweighs this advantage.

In addition to the effect of bypass ratio, compressor pressure ratio and absolute engine size also have first-order effects on engine weight. In general, however, both turbojets and turbofans are similarly affected by these variables, so that changes in design do not have a large effect on the relative weights of the two types.

It is evident from the cycle studies that the turbofan is not markedly superior to the turbojet at subsonic flight conditions. Differences in weight and specific fuel consumption partially offset each other, so that an engine designed to take advantage of the potential reduction in specific fuel consumption offered by the turbofan will be heavier than a corresponding turbojet. Mission studies show fairly conclusively that for long-range subsonic applications the turbofan may have about a 10-percent range advantage over the turbojet.

Supersonic Performance

In order to demonstrate the insensitivity of afterburning-turbofan cycle performance at supersonic speeds to over-all compressor pressure ratio and bypass ratio, figure 6 is presented. Included are afterburning turbofans having compressor pressure ratios from 6 to 12 and bypass ratios from 0.5 to 0.75 (with appropriate fan pressure ratios). Even for this wide range of design conditions, the turbofan performance (thrust per pound of air and specific fuel consumption) varies only about 10 percent for flight Mach numbers above 2.0.

Contrary to nonafterburning situations, the turbofan has lower thrust per pound of air and higher specific fuel consumption than the turbojet for all designs at all flight speeds below Mach 3.0. At Mach 3.0 the turbojet and turbofan have about the same thermodynamic performance.

The air-handling capacity of the turbofan is compared with that of turbojets with various over-all pressure ratios in figure 7. Compared with the turbojet with the same over-all compressor pressure ratio (6 in this case) the turbofan has a higher air-handling capacity, but compared with a low-pressure ratio turbojet the two engines have about the same air-handling capacity.

It may be concluded that within the limits of precision of this study the low-pressure-ratio turbojet and the turbofan engines have approximately the same performance, with full afterburning for flight speeds in the range from about 1.8 to 3.0. The turbojet has a slightly lower specific fuel consumption over most of the range. Compared with the high-pressure-ratio turbojet, the turbofan has higher thrust at flight speeds above Mach 2.0 because of its higher air-handling capacity. It is evident that for an all-supersonic mission there is little to choose between the low-pressure-ratio turbojet and the turbofan, if the engine size is set by the high-speed condition. If the engine size is set by the transonic condition, the high-pressure-ratio turbojet may be best because of its slightly higher transonic thrust. The relative merit of these engine designs for a two-speed mission is discussed in a subsequent section.

A reasonable turbofan design choice for a Mach 3.0 mission would have a primary compressor pressure ratio of 6 ± 2 and a fan pressure ratio of about 1.5 to 2.0 (at sea-level static), a bypass ratio of 0.5 to 0.7, and a turbine-inlet temperature of around 2200° R. Higher turbine-inlet temperatures do not appreciably alter the relative comparisons.

Supersonic-Engine Weight Considerations

For engines sized by the high-speed thrust requirements (e.g., Mach 3.0 interceptors) the turbofan engines will be about 30 percent lighter than turbojet engines of the same over-all compressor pressure ratio, as shown in figure 8. However, this weight advantage disappears if the turbofan is compared with the low-pressure-ratio turbojet. For engines sized at the transonic flight condition, turbofans and high-pressure-ratio turbojets have about the same weight.

Cruise-Dash Mission

The comparison of turbojet and turbofan engines for a cruise-dash mission is a very complex problem. Hence, only a cursory presentation is possible in this report. As shown in figures 5 and 8, the turbofan is heavier for a given amount of thrust at a nonafterburning cruise condition, but is lighter at high Mach number dash condition (afterburning). Hence, the relative advantages of the two engines depend to a large extent on whether the cruise or dash condition is critical for engine size.

The engine size necessary for cruise can be determined from the thrust required at the tropopause (cruise below the tropopause results in higher specific fuel consumption, while cruise at higher altitudes results in a larger engine). The engine size required for the supersonic dash is primarily a function of dash altitude. The engine sizes required for a typical cruise-dash mission are shown in figure 9. For a turbofan, the cruise condition dictates the engine weight up to an altitude of almost 75,000 feet, while for a turbojet the same is true up to only about 55,000 feet. However, since the turbojet-engine weight for cruise is smaller than that of the turbofan, the turbojet-engine weight is less than the turbofan weight below about 65,000 feet.

The specific fuel consumption is also important for range calculations. For the two engines selected, the turbofan has about a 25-percent advantage in specific fuel consumption at cruise, and about a 5-percent disadvantage for dash. The combined effects of weight and fuel consumption on airplane range for a two-speed mission are shown in figure 10. For the engine weights shown in figure 9 (no excess thrust for dash) the turbojet and turbofan ranges are not much different up to 65,000 feet, where the engine weights are about the same for both engines. Above this altitude the lighter weight of the turbofan results in an increasing range advantage over the turbojet. Also shown in figure 10 is the effect of an excess thrust requirement, such as would be needed for maneuverability. Such requirement increases the turbofan range advantage at high altitudes. At low altitudes the range ratio tends to approach the all-subsonic results (about 10-percent advantage for the turbofan).

The results shown in figure 10 are general, in that a high-altitude or high-combat g-load will always favor the turbofan for a cruise-dash mission. Of course, the particular numerical results depend on the airplane and engine configurations. For example, a lower-pressure-ratio turbojet would have less weight penalty at high altitudes, but the cruise fuel consumption would suffer.

Summary

Considering only engines which are reasonably near optimum in design the turbofan appears to have clearly superior (greater than about 10 percent) performance to the turbojet only in cruise-dash missions requiring very high altitudes, high combat g-loads, or both. Although the turbofan is better for the cruise-dash missions, it is not appreciably inferior to the turbojet for many other missions and would probably be an adequate powerplant for all but special cases.

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COMPARISON OF ENGINE PROPOSALS WITH ANALYSIS

Data have been obtained from eight engine proposals to compare with analytical results. Included in the proposals are five turbojets and three turbofans. For comparison, four analytical engines were used - two turbojets and two turbofans. These engines, with a few pertinent figures, are as follows:

Type	Engine	Manufacturer	Specific weight with afterburning at takeoff, W_e/F_n	Over-all compressor pressure ratio, P_2/P_1
		Engine proposals		
Turbojet	TJ31F7	Wright	0.185	7.5
	278A	GE	.171	8.0
	279E	GE	.158	8.0
	640-C2	Allison	-----	---
	670-C2	Allison	.194	6.9
Turbofan	WTF 10	Wright	0.199	---
	WTF 12	Wright (60% Bypass)	.235	12
	700-PD5	Allison (80% Bypass)	.202	6
		Analytical studies		
Turbojet	NACA Study engine	--------------------	0.28	2.3
	Prop. Res. Corp.	--------------------	.192	6
Turbofan	Prop. Res. Corp.	(50% Bypass)	0.185	6
		(67% Bypass)	.232	6

The selection of engines was restricted to design Mach numbers of about 3 to obtain a fair comparison. Two engines were designed for slightly lower Mach numbers (TJ31F7, M = 2.75; 700-PD5, M = 2.8); hence, they probably have slightly lighter weights than if they had been designed for a Mach number of 3. One engine (the NACA study engine) was designed for a considerably higher Mach number; hence, it is probably heavier than required for a Mach number of 3. The three Propulsion Research Corporation engines are from reference 1.

Specific Fuel Consumption

The specific fuel consumption of the engines is plotted in the upper parts of figures 11 to 14 as a function of flight Mach number for both military and maximum thrust conditions. The two areas of interest are at a Mach number of 1 with military thrust (approximate cruise condition) and a Mach number of 3 with maximum thrust. The specific fuel consumption of turbofans is about 25 to 30 percent less than that of turbojets at the Mach 1 military condition, while a slight difference exists in the opposite direction at the Mach 3 maximum condition. The specific fuel consumption shown in the engine proposals agrees closely with that of the analytical studies.

The best military specific fuel consumption at a Mach number of 1 was obtained with the WTF 12 turbofan. As will be shown in the weight discussion, this low fuel consumption was obtained at the expense of engine weight.

The best specific fuel consumption at maximum thrust for a Mach number of 3 was obtained with the 640-C2 turbojet, which had the highest turbine-inlet temperature of any engine proposal. This high turbine-inlet temperature was also the cause of the high military specific fuel consumption at a Mach number of 1 for this engine.

Thrust-to-Weight Ratios

The thrust-to-weight ratios of the various engines are plotted against Mach number in the lower parts of figures 11 to 14. At a Mach number of 1, the military thrust-to-weight ratios of the turbofans (both proposals and analytical studies) fell at or below those of the turbojets.

The WTF 12 has the lowest maximum thrust-to-weight ratio over the complete Mach number range shown, for two reasons: First, the WTF 12 has an over-all compressor pressure ratio of 12, which is more suited to an all-subsonic mission (note the WTF 12 has the best military specific fuel consumption at a Mach number of 1). Second, the WTF 12 was derated at high Mach numbers so that the thrust characteristics would match the WS-110A mission requirements. The vertical line at a Mach number of 3 shows the available increase in thrust for afterburner temperatures higher than the WTF 12 proposals.

Ignoring the WTF 12 and making allowances for weights as mentioned previously, it is evident that all of the engine proposals have about the same maximum thrust-to-weight ratio variation in the range from Mach 1 to 1.5. The same trend is also evident in the analytical studies.

The two engines with the best thrust-to-weight ratios at Mach 3 are both designed for all-supersonic missions, the 700-PD5 turbofan and the NACA study turbojet. (The 700-PD5 has good specific fuel consumption at Mach 1, but the nonafterburning thrust is so low that only a very unusual mission could take advantage of it.) Both of these engines were designed for engine pressure ratios of about 1.5 at takeoff. Although the thrust-to-weight ratios of these engines at takeoff (see table) are good, it is evident that a moderate-pressure-ratio turbojet (P_2/P_1 of 6 to 8) is somewhat better.

It is interesting to note that the NACA study engine exhibits most of the characteristics usually attributed to a turbofan: light weight, high augmentation ratio, and large thrust variation with Mach number. Although the airflows are not shown, the NACA study engine also has high corrected airflow at a Mach number of 3. However, this engine does not have good subsonic specific fuel consumption at military thrust. The 700-PD5 and the NACA study engine probably would have the same maximum thrust-to-weight ratios if they had been designed for the same Mach number.

The analytical studies indicate a large weight advantage for the turbofan over the conventional turbojet at a Mach number of 3. The turbofan proposals (except for the 700-PD5) do not show this weight advantage. Apparently the WTF 10, like the WTF 12, has a higher-than-optimum compressor pressure ratio for light weight.

Summary

The low potential specific fuel consumption of turbofans at about a Mach number of 1 (military thrust) is realized in the engine proposals investigated. At a Mach number of 3 (with maximum thrust) there is little difference in specific fuel consumption among all the turbofans and turbojets.

As shown in both the engine proposals and the analytical studies, the low military specific fuel consumption of turbofans near a Mach number of 1 is achieved at a low thrust-to-weight ratio.

The maximum thrust-to-weight ratio is insensitive to engine type in a Mach number range from 1 to 1.5. At a Mach number of 3, the maximum thrust-to-weight ratios of the WTF 10 and WTF 12 were not as high as those of analytical studies, apparently because of a compromise in the direction of higher compressor pressure ratio.

The best thrust-to-weight ratio near a Mach number of 3 was obtained with the 700-PD5 turbofan. The analytical studies indicated, however, that a low-compressor-pressure-ratio turbojet (the NACA study engine) would be about as attractive as the 700-PD5 at this Mach number.

Although the turbojets with moderate (6 to 8) compressor pressure ratios were not outstanding at a Mach number of 3, their thrust-to-weight ratios were excellent at takeoff.

COMPONENT PROBLEMS FOR TURBOFAN ENGINE

In order to examine the component problems for ducted-fan engines designed for flight Mach numbers up to 3.0, the proposed Wright WTF 12 turbofan engine has been compared with the proposed GE-279A and the Wright TJ31F7 turbojet engines. These engines were selected for comparison because these proposals gave the most complete information on components and their ranges of operation. Engine data were available for the following operating conditions:

Mach number	Altitude, ft	Thrust
Static	Sea-level	--------------------
0.9	36,000	Maximum (nonafterburning) (Minimum sfc)
2.0	45,000	Maximum Military
2.5	60,000	Maximum 75 Percent
3.0	60,000	Maximum 75 Percent

No data at Mach 3.0 were available for the TJ31F7 engine, but data for Mach 2.75 at 60,000 feet were available for this proposed engine. These flight conditions represent requirements for the WS-110A mission.

From the meager data available, a complete evaluation of component problems (stress, temperature, and aerodynamic limits) for the turbofan engine is not possible. However, an attempt has been made, based on the available data, to compare the relative component problems of turbofan and turbojet engines designed to fulfill the same mission requirements. The following components are compared: inlets, compressors, primary burners, turbines, afterburners, and exhaust nozzles.

Inlets

For an airplane with good subsonic as well as good supersonic performance, the air inlet must be adjustable. Therefore, the inlet problem may be measured by the range of compressor equivalent-flow requirements. For the three reference proposed engines, the maximum equivalent flow was for maximum thrust at 0.9 flight Mach number at 36,000 feet altitude. Minimum equivalent flow occurred at the Mach 3.0 conditions at 60,000 feet. The ratio of maximum to minimum flow for the turbofan was 1.98, compared with 1.93 for the turbojet.

Thus, for the range of flight conditions considered, the amount of inlet adjustment required for the two engine types is about the same. This indicates that the inlet problems for the turbofan engine are comparable to those for the turbojet.

Compressors

For all high Mach number engines that utilize compressors, the compressor must operate over an appreciable range of equivalent speeds and flows. For the two-spool turbofan engine the compressor problem is apparently no more severe than for the turbojet engine. For the single-spool turbofan, the wheel speed of the compressor will be low because of limiting speed of the larger-diameter fan. In this case, a large number of compressor stages will be required, and the compressor will present more severe weight problems. At very high flight Mach numbers, the fan approaches pressure ratios of 1, and choking limits may restrict the airflow capacity of the engine. The fan, however, is a low-pressure-ratio unit and inherently has a much broader range of useful operation than a compressor of much higher pressure ratio has.

The turbofan also presents problems in regard to matching of the fan with the bypass duct and compressor. The flow divider that separates bypass air from compressor air may require some development to avoid adverse effects on compressor or duct performance as bypass ratio varies. Inspection and maintenance of the compressor component also present some problems for the turbofan engine.

Primary Combustors

The primary combustor for the turbofan engine is almost the same as that for the turbojet engine. Therefore, combustor problems are comparable for the two types of engines. Accessibility renders maintenance and inspection somewhat more complicated for the turbofan combustor.

Turbines

For the single-spool turbofan engine the turbine is smaller in diameter than the fan and must produce high specific work. This requires either a large number of low-work stages or a smaller number of high-work low-wheel-speed stages. The high-work low-speed turbines have low values of blade- to jet-speed ratio, and this type of turbine is inherently less efficient than those commonly used on turbojet engines. For a two-spool turbofan the problems are very similar to turbojet problems.

It appears that the turbine problem is more severe for the single-spool turbofan than for the turbojet engine. Either turbines with many stages or turbines with low blade- to jet-speed ratios are required. Turbines with many stages present weight and stage-matching problems, whereas the turbines with low blade- to jet-speed ratios have inherently low efficiencies. The two-spool turbofan presents fewer turbine problems but may entail a weight penalty. The turbofan also presents turbine maintenance and inspection problems.

Afterburners

For all high flight Mach number engines, afterburner pressure is high and good combustion efficiency should be achieved at high flight Mach numbers.

The turbofan afterburner itself presents no more difficult problems than that for the turbojet engine, except for the problem of mixing of bypass air and turbine-exit air. Streams of nonuniform temperature and velocity may produce adverse effects on afterburner pressure loss and combustion efficiency. Therefore, the turbofan may require special aids to promote mixing of the bypass and turbine air as well as a longer mixing length than required for the turbojet engine. The lower afterburner-inlet temperature may tend to reduce the combustion efficiency slightly, but afterburner cooling should be easier.

Exhaust Nozzles

Engines capable of flight at a Mach number of 3, in general, require some type of adjustable convergent-divergent exhaust nozzle. To evaluate the complexity of these adjustable nozzles, the ratio of minimum to maximum throat area and exit area can be compared. For the turbofan engine studied, the minimum throat area was 61.8 percent of the maximum value; whereas for the two turbojets this value was 63.6 and 65.0 percent. The minimum exit area for the turbofan nozzle was 30.8 percent of maximum, compared with 35 and 39.3 percent for the turbojets. These data indicate that the range of adjustment required for the turbofan is about the same as for the turbojets. Therefore, the exhaust-nozzle problems for the turbofan and turbojet engines are considered comparable.

Summary

On the basis of the information available for three engine designs, there appear to be no extremely difficult component development problems in the Mach 3.0 turbofan-engine design. The inlet, exhaust-nozzle, combustor, and afterburner developments required are comparable to those for advanced turbojets. Compressor and turbine development problems may be more severe.

RESULTS AND DISCUSSION

The relative merits of the turbojet and turbofan depend to a large extent on the particular mission under consideration. Flight speed is probably the most important mission variable for a turbojet-turbofan comparison. This discussion is therefore subdivided into three sections: (1) all-subsonic missions, (2) subsonic cruise with supersonic dash, and (3) all-supersonic missions.

All-Subsonic Mission

The comparison of turbojet and turbofan designs for all-subsonic missions reduces to a weight advantage for the turbojet and a specific-fuel-consumption advantage for the turbofan. The compressor pressure ratio of both the turbojet and the turbofan should be high (10 to 12) for an all-subsonic mission.

Four all-subsonic missions (ref. 1) are as follows:

	$\frac{\text{Turbofan range}}{\text{Turbojet range}}$
VTOL fighter-bomber	0.75
Fighter-bomber	.96
Transport	1.13
Tanker	1.13

The VTOL fighter-bomber places the greatest premium on low engine weight; hence, the range is considerably less with a turbofan. The fighter-bomber, with a 1.5-g combat requirement, has about the same range with either type of engine. Only the tanker and transport, with their large emphasis on specific fuel consumption, show an advantage for the turbofan. It should be noted, however, that operation of the tankers or transports with higher payloads over shorter ranges (same gross weight) would reduce the advantage.

The turbofan has an advantage over the turbojet only for very long-range missions in all-subsonic flight. This advantage is small (approximately 10 percent) and would tend to become smaller if the airplanes were to operate largely over ranges much less than maximum. The engine proposals previously presented were designed for operation near a Mach number of 3; therefore, no conclusions are drawn regarding their suitability for all-subsonic missions.

Subsonic Cruise with Supersonic Dash

For a mission that requires operation at both subsonic and supersonic speeds, the turbofan has two advantages: (1) low subsonic specific fuel consumption at military (nonafterburning) thrust, and (2) high maximum (afterburning) thrust-to-weight ratio at a supersonic Mach number. The disadvantage of the turbofan for a two-speed mission is the same as for an all-subsonic mission; that is, low thrust-to-weight ratio with military thrust.

The two-speed missions from reference 1 are as follows:

	Turbofan range / Turbojet range
VTOL local interceptor	0.80
VTOL area interceptor	.84
Local interceptor	1.48
Area interceptor	1.84
Long-range interceptor	1.45
Subsonic-supersonic bomber	1.05

As in the all-subsonic missions, VTOL severely penalizes the turbofan. The subsonic-supersonic bomber shows a 5-percent advantage for the turbofan, but a slight change in airplane configuration or altitude could change this result considerably. The local, area, and long-range interceptors all show marked advantages for the turbofan. The interceptors other than VTOL have a 1.5-g combat requirement; therefore, both the good subsonic specific fuel consumption and high supersonic thrust-to-weight ratio of a turbofan are advantageous for such missions. Although turbojets with compressor pressure ratios below 6 were not considered in reference 1, a low-pressure-ratio turbojet would also be promising for a non-VTOL interceptor mission. The amount of subsonic flight, of course, would determine the relative advantages of the low-pressure-ratio turbojet and the turbofan.

The value of a turbofan for a two-speed mission depends on whether or not the low thrust-to-weight ratio at subsonic military thrust is sufficient for cruise. If the dash condition is at a high enough altitude, or if a high combat g-load is required, the engines will be large enough that subsonic military thrust will not be a problem. On the other hand, if takeoff or cruise conditions size the engines, then the increased engine weight over a turbojet installation can easily eliminate any significant advantage for a turbofan.

As for a suitable engine for a two-speed mission, the 700-PD5 is the only turbofan that has any of the high Mach number weight advantages that a turbofan should have. But the bypass ratio of the 700-PD5 is so high that the subsonic military thrust is probably inadequate for cruise. The WFF 12 has a higher thrust-to-weight ratio at cruise, but falls considerably below the values obtained in the analytical studies at a Mach number of 3. Thus, probably none of the turbofan proposals examined is a suitable engine for a high Mach number two-speed mission.

All-Supersonic Mission

No all-supersonic missions were included in reference 1, but the unimportant role of specific fuel consumption at other than design conditions permits considerable simplification in the approach. At design conditions (a Mach number of 3 is assumed) the performance of all engine types approaches that of a ramjet. (Some improvements over ramjet specific fuel consumption can be obtained by careful design or high turbine-inlet temperature, but these differences are smaller than the accuracy of this investigation.) Thus, the suitability of engines for all-supersonic missions reduces to the thrust-to-weight comparison over a range of Mach numbers.

As mentioned previously, the variation of turbofan thrust-to-weight ratio with Mach number is duplicated closely by a low-compressor-pressure-ratio turbojet; thus, there is no important characteristic of a turbofan for all-supersonic missions that cannot be duplicated with some form of turbojet.

If takeoff tends to be critical, then a conventional turbojet with a compressor pressure ratio of 6 to 8 would probably be the proper choice. If the transonic drag rise is large and a high thrust is required at this condition, the choice of engine type is indeterminate. Probably other smaller factors would need examination. When design-point thrust tends to be critical, then a turbojet with a low compressor pressure ratio (3 to 4) would probably be the proper choice. It would be a better choice than a turbofan, because the simpler geometry should result in fewer development problems.

Several conventional turbojets were included in the engine proposals. However, no low-pressure-ratio turbojets were included. Thus, although there appears to be no clear advantage for a turbofan (over a low-pressure-ratio turbojet) for an all-supersonic mission, the 700-PD5 turbofan is a suitable alternative to a low-compressor-pressure-ratio turbojet, which would be the likely choice for a mission where design-point thrust is critical.

CONCLUSIONS

The turbofan is outstanding for cruise-dash missions with a very high dash altitude and short range, a high combat g-load (requiring excess thrust), or both. The turbofan has a marginal advantage (10 percent) for long-range subsonic missions and cruise-dash missions at more moderate altitudes or combat g-loads. The turbofan has no advantage for an all-supersonic mission and is at a substantial disadvantage for VTOL missions.

The question of suitability of engine proposals for all-subsonic missions was not considered, since the engines selected were for design Mach numbers near 3. As for cruise-dash missions, none of the turbofan proposals appeared to have the combination of fair military thrust-to-weight ratio at subsonic cruise with high maximum thrust-to-weight ratio at supersonic dash that was indicated by analytical studies. For all-supersonic missions a turbojet can be designed to approximately match the performance of any turbofan. Because the turbojet probably has slightly less development problems, it is probably a better choice for all-supersonic missions. However, no low-pressure-ratio turbojets (such as would be required to match turbofan performance) were included in the available engine proposals.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, July 18, 1957

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1. Sohn, Robert L., ed.: Performance of a Series of Ducted-Fan Engines over a Wide Range of Operating Conditions and Applications. WADC Tech. Rep. 55-201, Wright Air Dev. Center, Air Res. and Dev. Command, USAF, Wright-Patterson Air Force Base, June, 1955. (Contract AF 33(616)-2435.)

2. Wright Aeronautical Division, Curtiss-Wright Corporation: Turbofan Engine Study, pts. I through IV. WADC Tech. Rep. 55-288, Wright Air Dev. Center, Air Res. and Dev. Command, USAF, Wright-Patterson Air Force Base, 1954-1955. (Contract AF 33(616)-2471.)

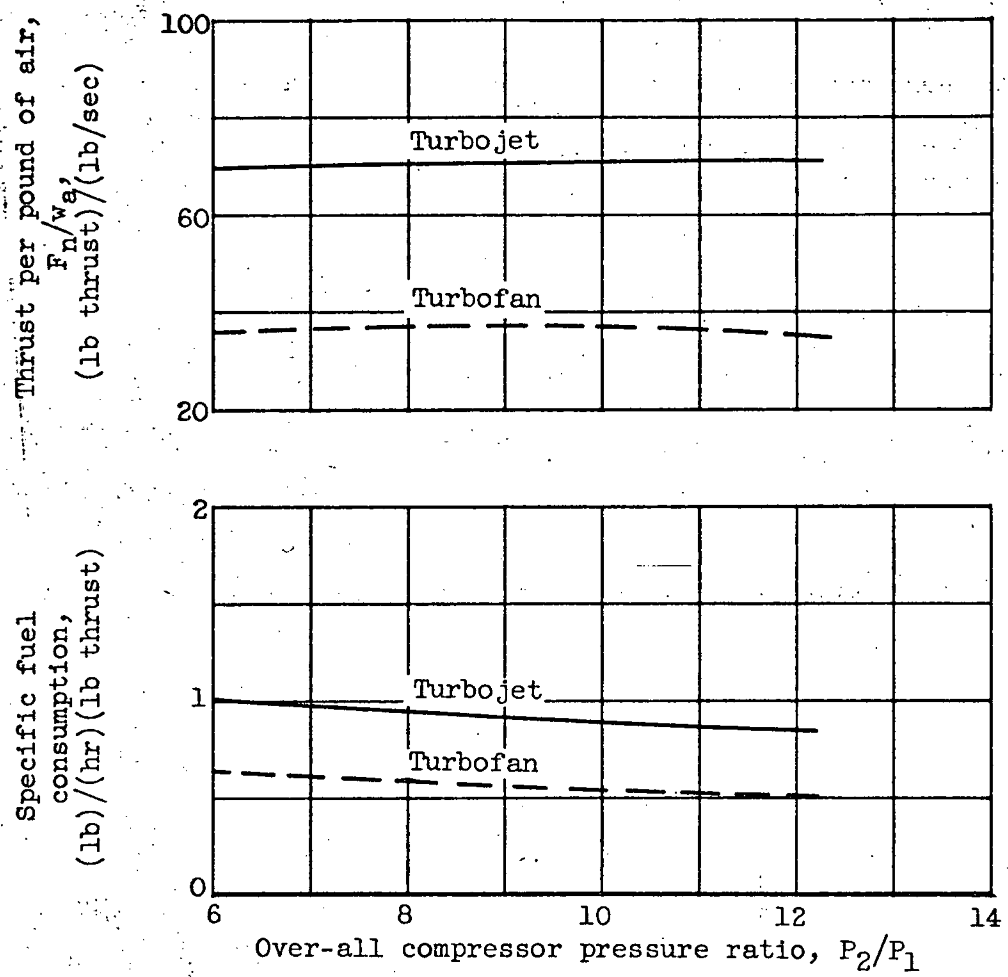

Figure 1. - Effect of over-all pressure ratio at sea-level static conditions (nonafterburning). Turbine-inlet temperature, 2210° R; fan pressure ratio, 1.73 to 1.80; bypass ratio, 0.67.

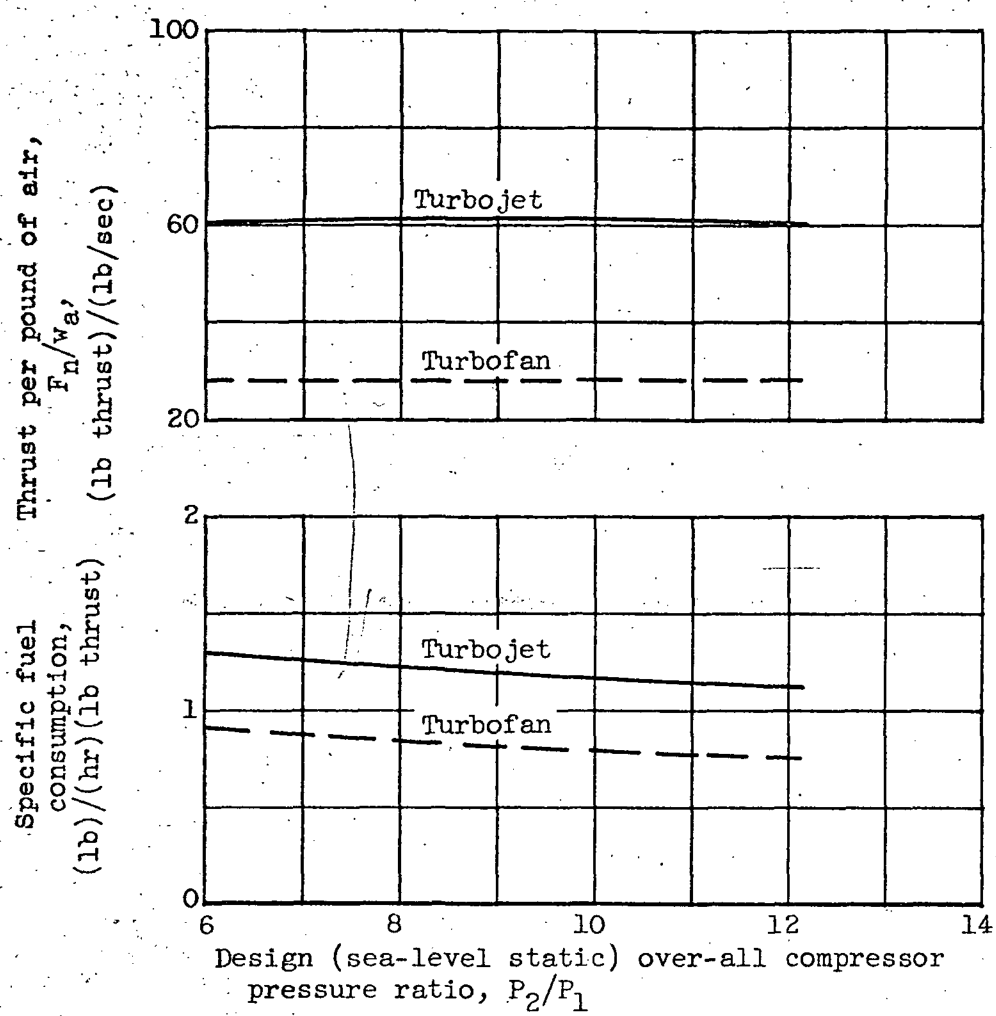

Figure 2. - Effect of over-all pressure ratio at tropopause with flight Mach number of 0.9 (non-afterburning). Turbine-inlet temperature, 2210° R; bypass ratio, 0.67.

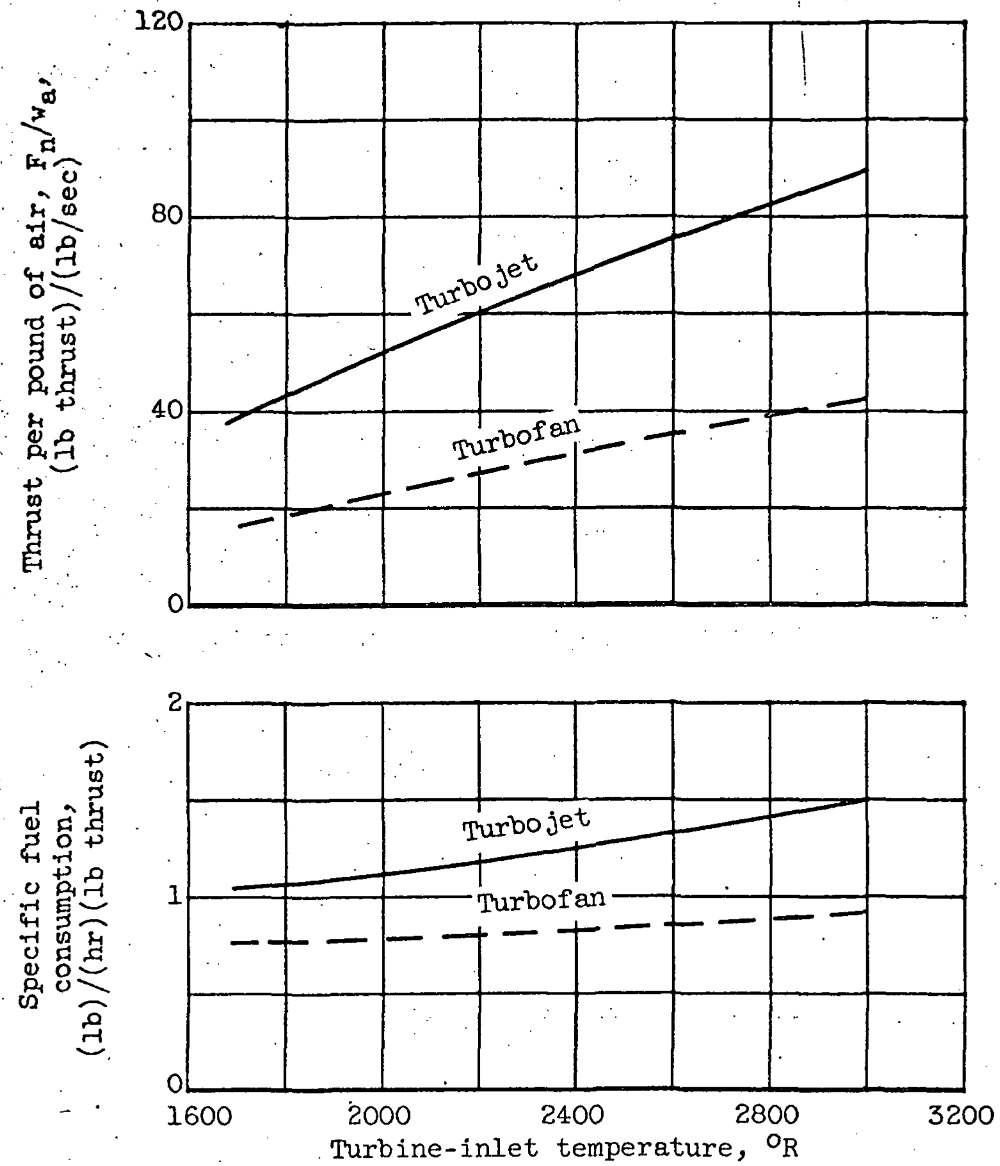

Figure 3. - Effect of turbine-inlet temperature at tropopause with flight Mach number of 0.9 (non-afterburning). Over-all compressor pressure ratio, 10; bypass ratio, 0.67.

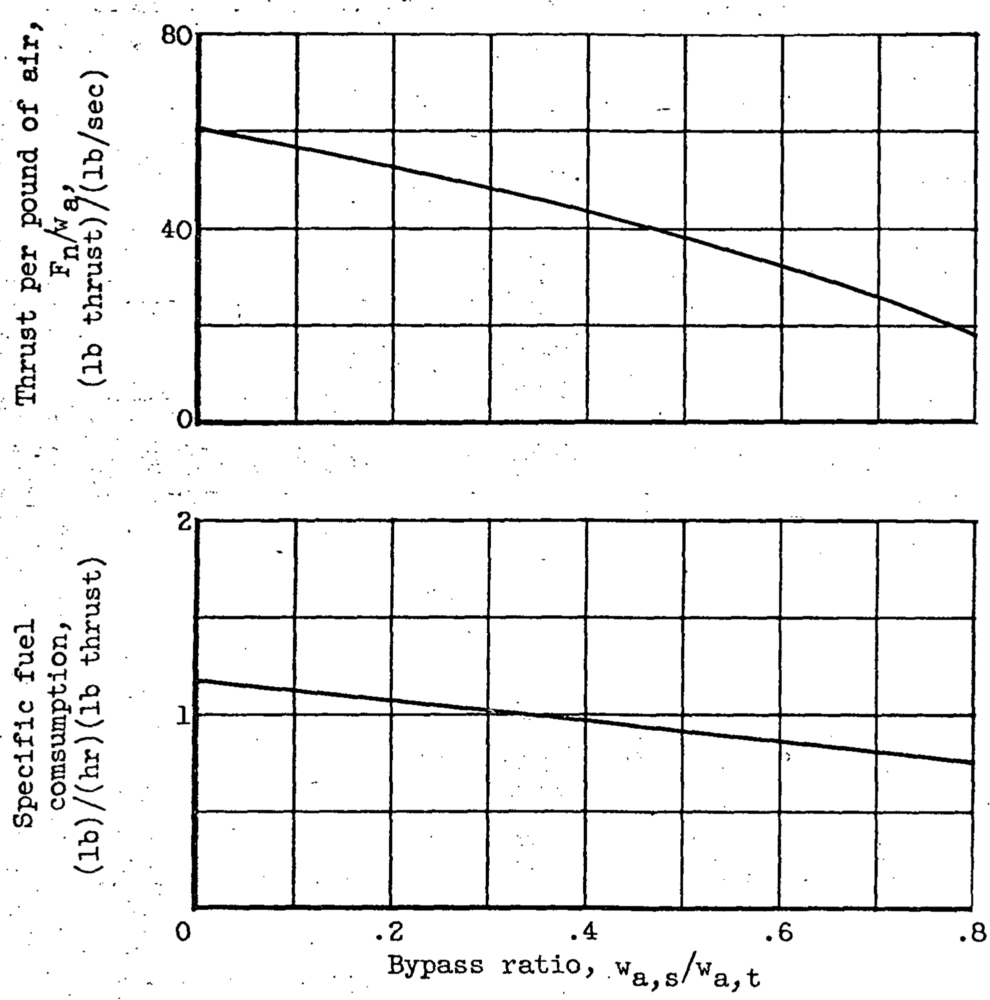

Figure 4. - Effect of bypass ratio at tropopause with flight Mach number of 0.9 (nonafterburning). Turbine-inlet temperature, 2210° R; over-all compressor pressure ratio, 10.

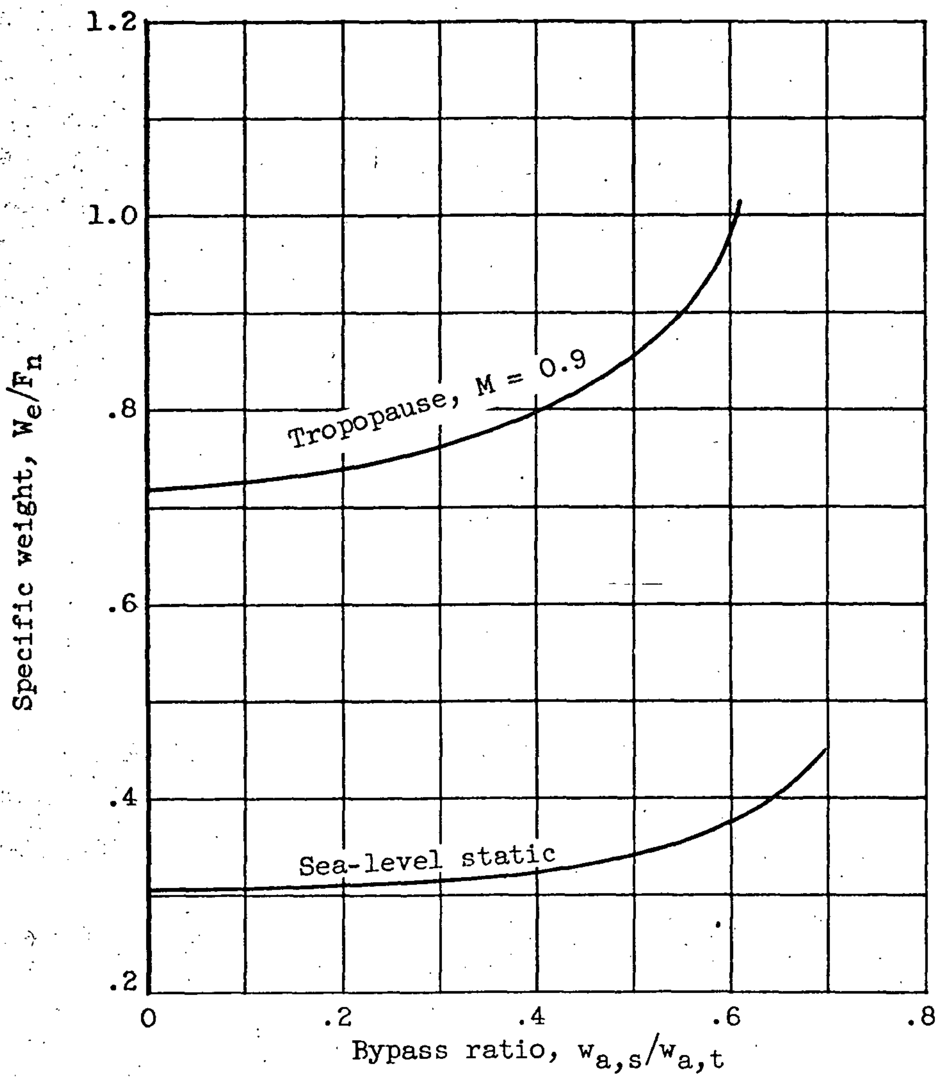

Figure 5. - Engine specific weight at subsonic flight conditions (nonafterburning). Turbine-inlet temperature, 2210° R; over-all compressor pressure ratio, 6.

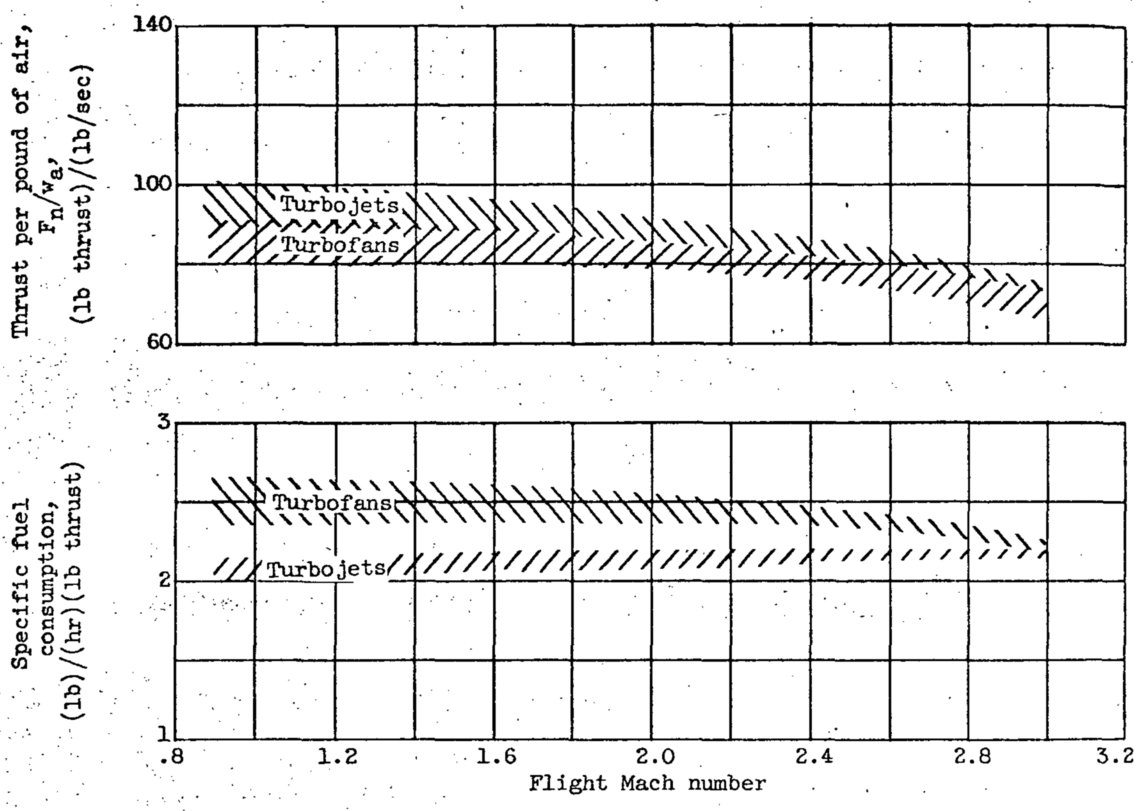

Figure 6. - Supersonic performance of afterburning engines. Afterburner temperature, 3500° R; turbine-inlet temperature, 2210° R; bypass ratio, 0.5 to 0.75; over-all compressor pressure ratio, 6 to 12 for turbofans, 3 to 6 for turbojets.

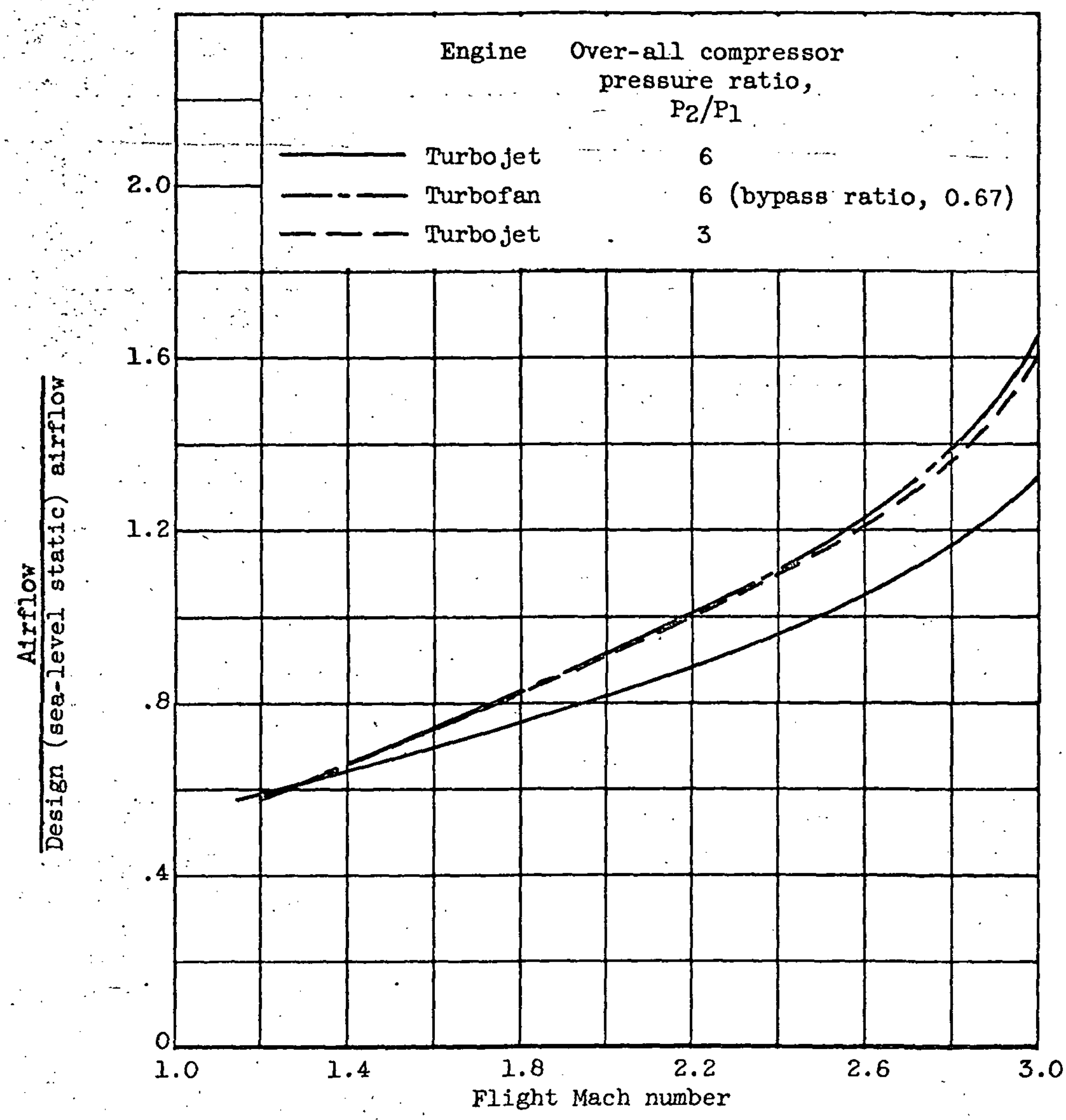

Figure 7. - Variation of airflow with Mach number at tropopause.

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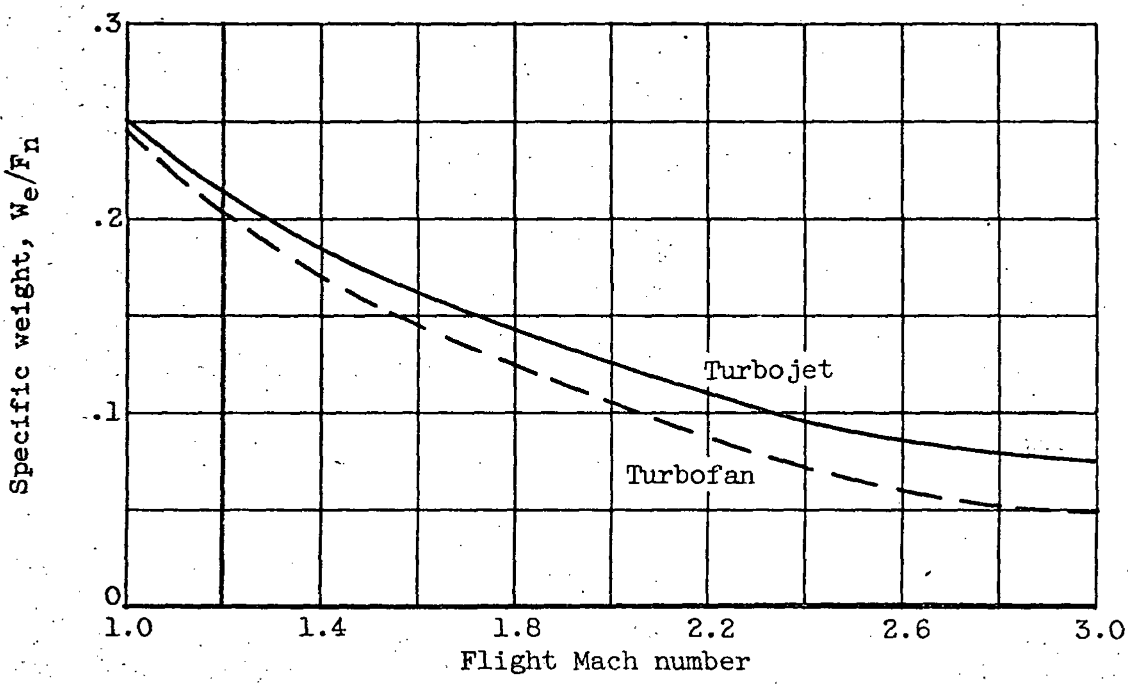
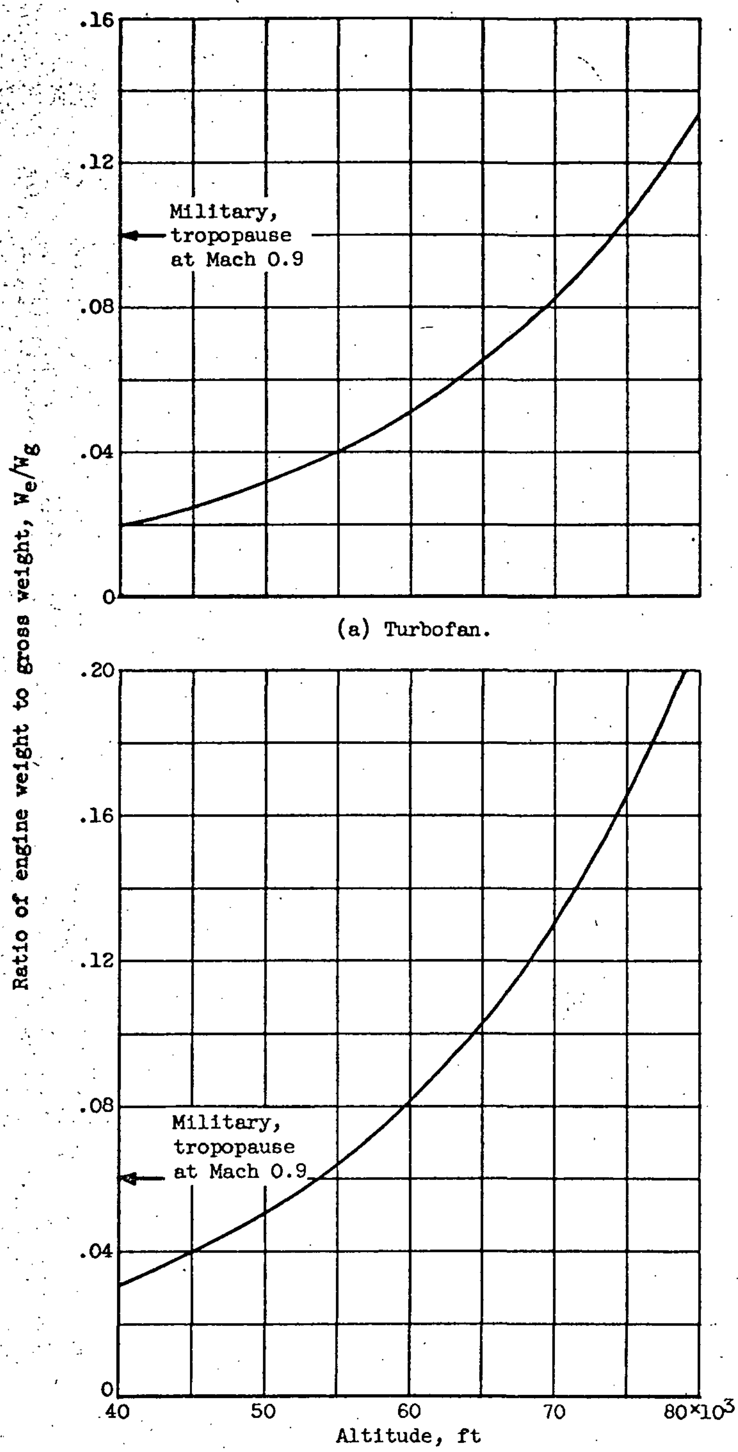


Figure 8. - Specific weight at supersonic flight conditions. Turbine-inlet temperature, 2210° R; afterburner temperature, 3500° R; over-all compressor pressure ratio, 6; bypass ratio, 0.6.



(b) Turbojet.

Figure 9. - Engine weight at flight Mach number of 3. Turbine-inlet temperature, 2210° R; over-all compressor pressure ratio, 6; bypass ratio, 0.67; afterburner temperature, 3500° R.

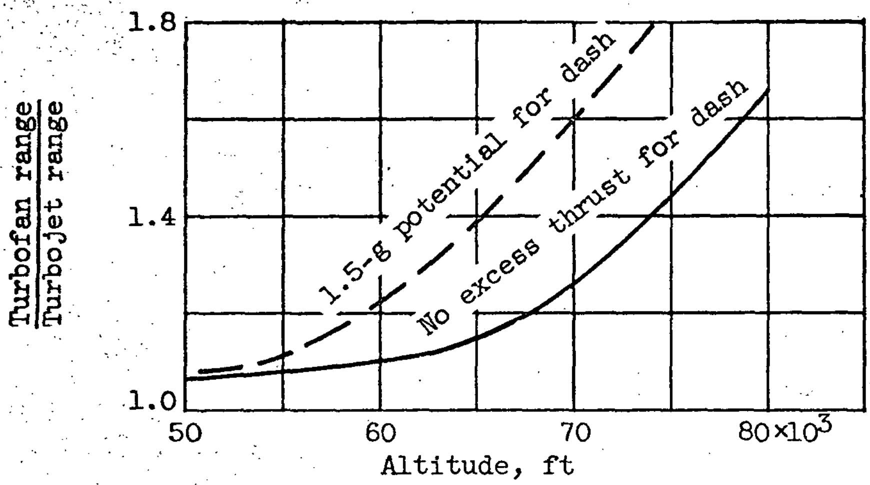


Figure 10. - Comparison of turbofan and turbojet engines for cruise-dash mission. Turbine-inlet temperature, 2210° R; overall compressor pressure ratio, 6; bypass ratio, 0.67; dash afterburner temperature, 3500° R; cruise at Mach 0.9 (2/3); dash at Mach 3 (1/3).

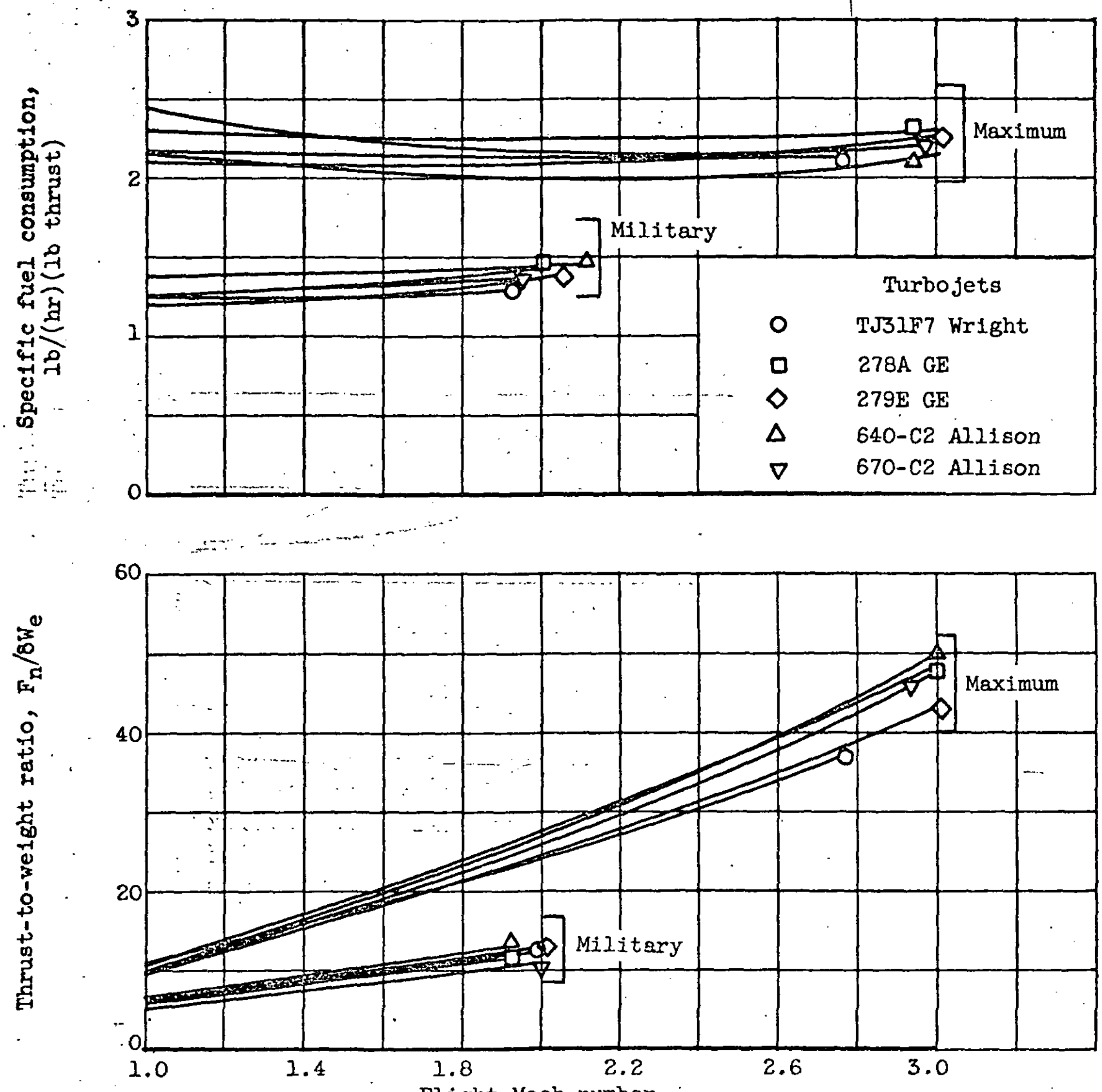


Figure 11. - Turbojet-engine proposals.

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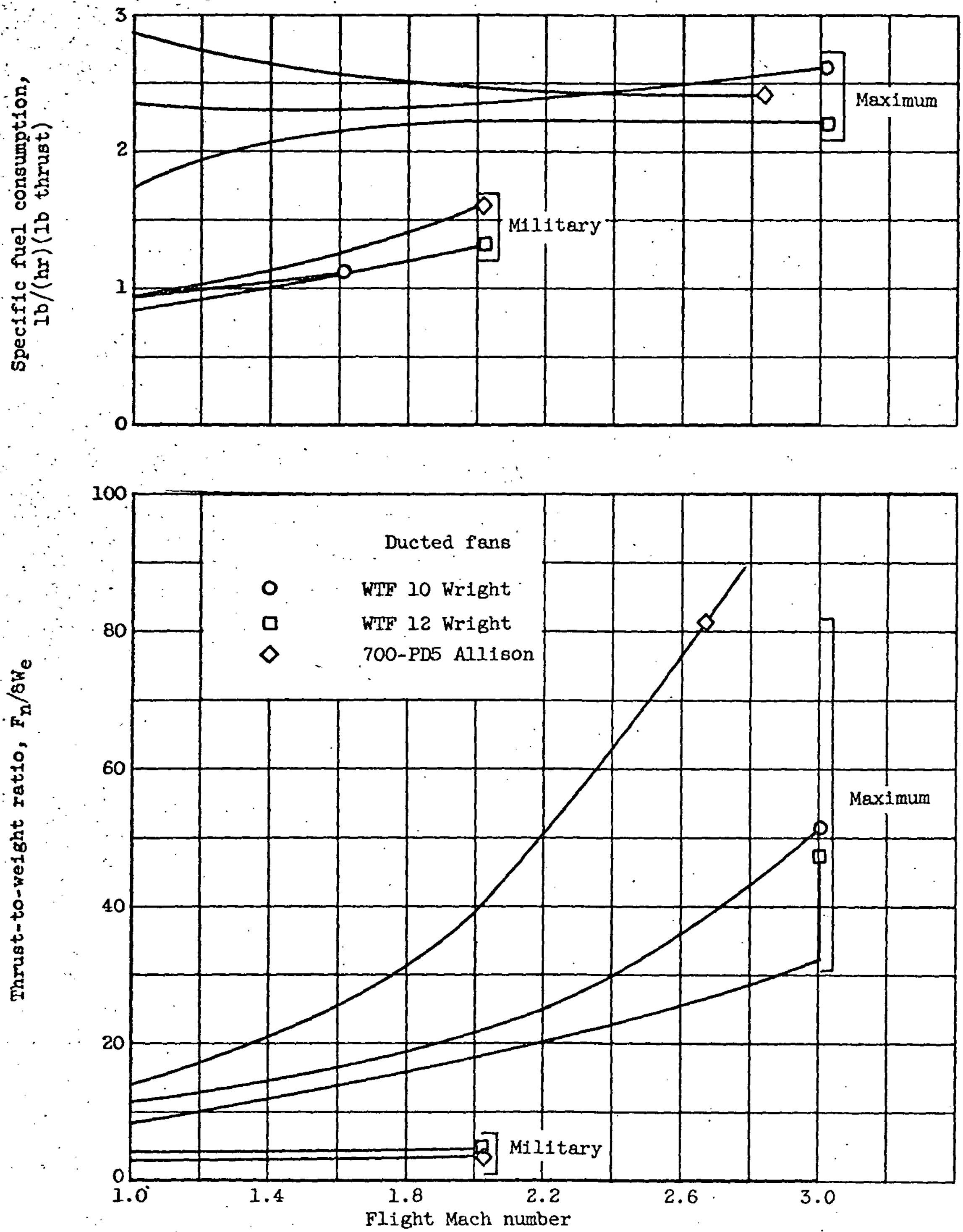

Figure 12. - Turbofan-engine proposals.

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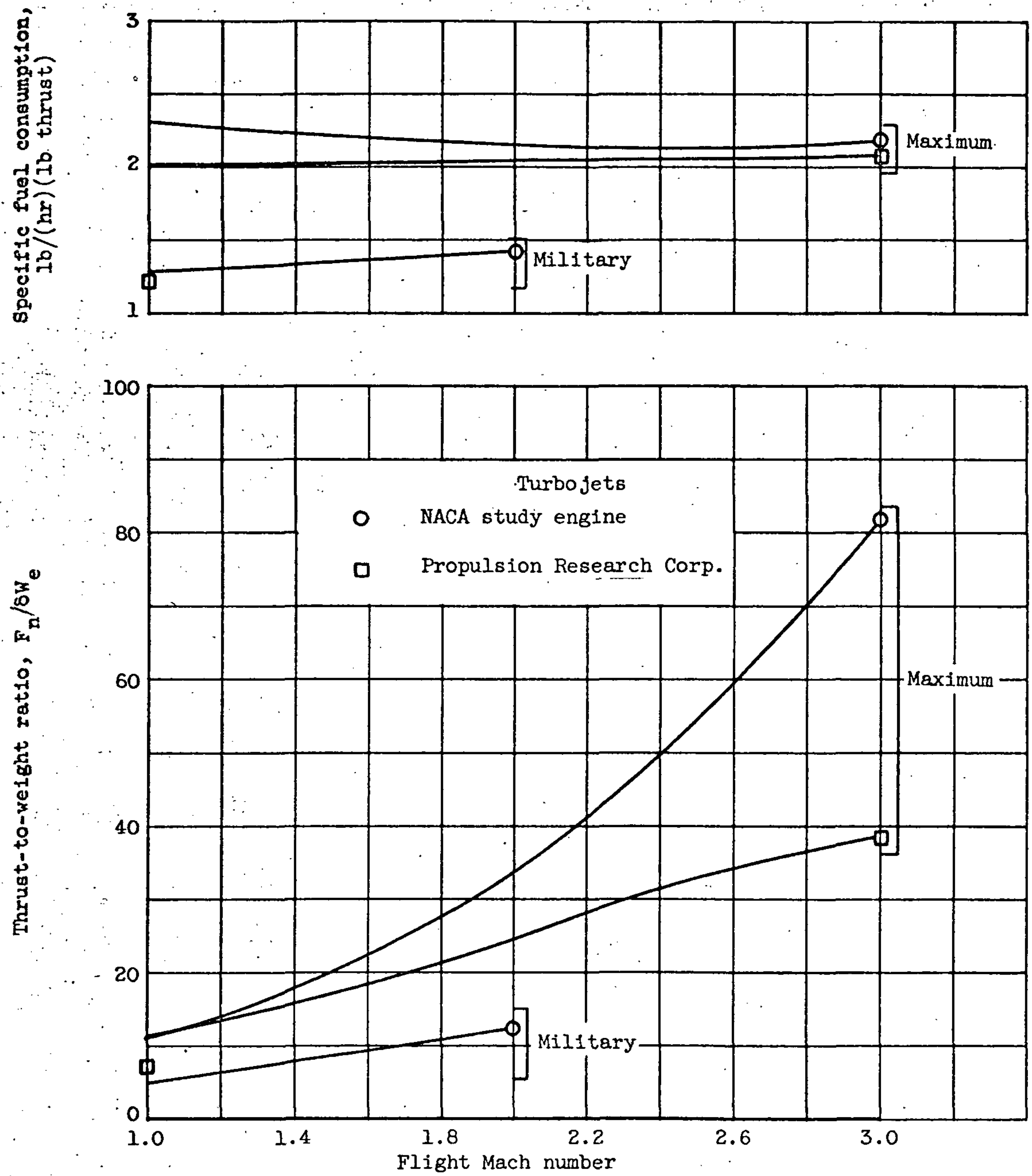

Figure 13. - Turbojet analytical studies.

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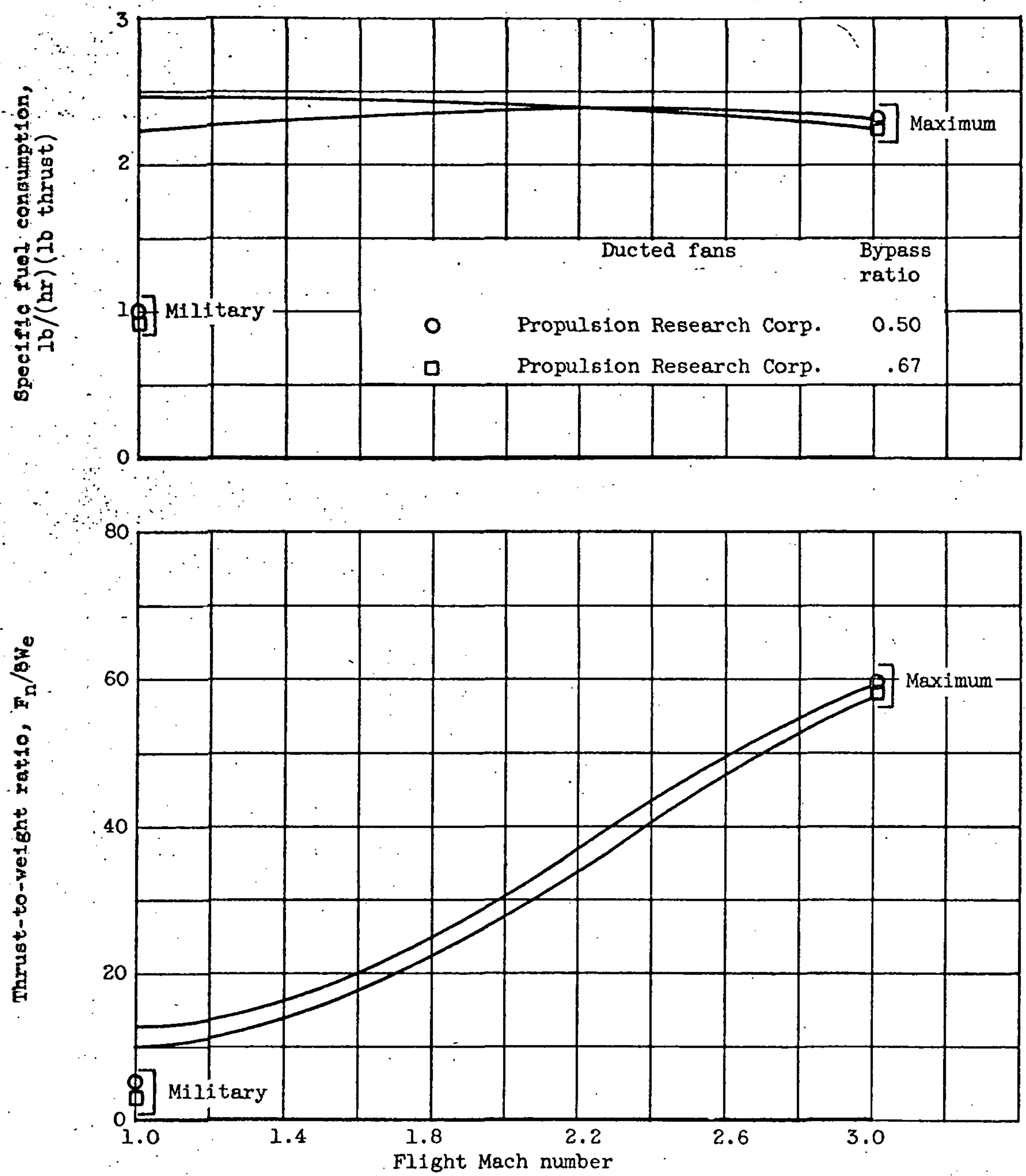

Figure 14. - Turbofan analytical studies.

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Unclassified when detached from rest of report

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COMPARATIVE STUDY OF TURBOFAN AND TURBOJET ENGINES

By Harold R. Kaufman, William A. Benser, and David S. Gabriel

ABSTRACT

Turbofan and turbojet engines are compared in all missions of current interest. The comparison includes considerations of cycle performance, several current engine proposals to the Air Force, and component performance and development problems. The study is restricted to applications for flight Mach numbers up to 3.0 and to the use of conventional hydrocarbon fuels.

INDEX HEADINGS

Engines, Turbojet	3.1.3
Engines, Ducted Propeller	3.1.5
Engine Types, Comparison	3.1.12

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