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PRELIMINARY STUDY OF AN INTEGRAL FAN LIFT/CRUISE ENGINE FOR A 100-PASSENGER VTOL TRANSPORT

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ABSTRACT

A simplified mission analysis was performed to determine an optimum engine cycle for a 100-passenger VTOL transport with a range of 500 statute miles. The aircraft had a total of eight integral fan lift engines, three of which serve as cruise engines. Fan pressure ratio was varied from 1.2 to 1.3, overall pressure ratio from 7 to 13, and turbine inlet temperature from 2460° to 2860° R. Bypass ratio was selected to meet a 500-foot altitude flyover noise goal of 95 PNdB Airplane gross weight and direct operating cost (DOC) were calculated. The lowest DOC of 1.82 cents per seat-mile was achieved with a fan pressure ratio of 1.3, overall pressure ratio of 12, and turbine inlet temperature of 2860° R. Initially, acoustic treatment weight was accounted for by penalizing all engines equally in terms of percent bare engine weight. A sensitivity study on engine weight later showed that this penalty, if increased $2\frac{1}{2}$ times on the 1.3 fan-pressure-ratio engines, would eliminate the DOC and gross weight advantage they held over the 1.2 and 1 25 fan-pressure-ratio engines.

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PRELIMINARY STUDY OF AN INTEGRAL FAN LIFT/CRUISE ENGINE FOR A 100-PASSENGER VTOL TRANSPORT

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SUMMARY

A parametric study was made of an integral fan lift engine that provided cruise as well as lift thrust for a 100-passenger VTOL transport. The aircraft had eight engines, three of which operated during cruise. A 500 statute mile range was selected with a cruise Mach number of 0.75 at an altitude of 20 000 feet. Fan pressure ratio was varied from 1.2 to 1.3, overall pressure ratio from 7 to 13, and turbine inlet temperature from 2460° to 2860° R. Design point for the engines was sea level static, on a 90° F day. A noise goal of 95 PNdB at 500 foot altitude and 80 percent thrust determined the bypass ratio for each cycle.

A straight-line altitude versus Mach number flight path and a Breguet cruise were used in the mission analysis. Lift/drag ratios assumed a symmetrical drag polar and included variations in engine pod drag. The study assumed equal weight penalties for acoustic treatment of each fan pressure ratio. A sensitivity study was included to show the effect of an increasing weight penalty with higher fan pressure ratio.

Sized for takeoff, all but one of the cycles produced adequate cruise thrust with three engines. All required a reduction in duct nozzle exhaust area with increasing altitude; the greatest reduction was about 30 percent for the 1.2 fan-pressure-ratio engines at cruise. For the range of variables examined, the lowest gross weight of 102 200 pounds and DOC of 1.82 cents/seat-mile were achieved with an overall pressure ratio of 12, turbine inlet temperature of 2860° R, and fan pressure ratio of 1.3. The sensitivity study showed that the improvement with higher fan pressure ratio would be reversed, if the actual weight penalty of acoustic treatment

for the 1.25 and 1.3 fan pressure ratio engines were about $1\frac{1}{2}$ and $2\frac{1}{2}$ times greater, respectively, than for the 1.2 fan pressure ratio.

INTRODUCTION

Vertical takeoff and landing aircraft offer an improvement in short haul air transportation by relieving congestion at present airports, reducing airtime delays, serving communities currently without airports, and allowing city-center to city-center travel. A number of VTOL aircraft configurations and propulsion systems (e.g., rotors, tilt-wing propellers, and lift fans) have been studied both here and abroad (e.g., refs. 1, 2, and 3). None of the concepts has emerged outstandingly superior, and interest, therefore, continues in many of them.

In this present report, a low-pressure-ratio, lift-fan propulsion system is studied. Reference 4 reviews the requirements and problem areas of such systems. Some of the features that make this type of system desirable for civilian VTOL are: (1) good potential for meeting reduced noise limitations, (2) provision for safe management of powerplant or thruster failure, (3) capability of cruise speed approaching that of conventional jet transports, (4) use of available gas turbine technology, and (5) elimination of mechanical transmissions. Two general types of lift-fan systems currently under study are the remote-drive lift fan and the integral-drive lift fan. The integral system consists of high bypass, low-fan-pressure-ratio turbofans whose thrust is directed downward either by engine positioning or thrust vectoring. The remote type consists of a number of lift fans powered by a working fluid ducted from separately located powerplants. A remote system which ducts the exhaust from a turbojet engine to drive a tip-mounted turbine on the lift fan has been under investigation for a number of years by General Electric (ref. 5) and was used in the GE XV-5A (ref. 6). Another remote system under consideration uses a low bypass, high-pressure-ratio turbofan (air generator) to supply compressed air to an auxiliary burner just upstream of the tip-turbine lift fans. One air generator/lift fan VTOL configuration studied at the Lewis Research Center has been reported in reference 7.

The objective of the present study is to optimize the parameters of an integral fan lift/cruise cycle for a 100-passenger VTOL transport meeting a flyover noise goal of 95 PNdB at 500 feet altitude. Fan pressure ratio, overall compressor pressure ratio, and turbine inlet temperature were varied in the study, in order to minimize gross weight for a fixed range and payload. Direct operating costs were calculated from the gross weights.

The aircraft configuration had a total of eight engines, three of which were used for cruise. A 500-statute mile range was selected with a cruise Mach number of 0.75 at an altitude of 20 000 feet. Lift/drag ratios included a drag variation with engine pod size. Fan pressure ratios of 1.2, 1.25, and 1.3, overall compressor pressure ratios of 7, 10, and 13, and turbine inlet temperatures of 2460°, 2660°, and 2860° R were examined. The engines were sized for a maximum thrust/gross weight of 1.375 at sea level static, on a 90° F day. Bypass ratio for each cycle was determined so as to meet the specified noise goal. Initially, all the engine cycles were penalized with acoustic treatment weight equal to 20 percent of bare engine weight. A sensitivity study was later included to examine the effects on gross weight and DOC of increased weight penalties.

SYMBOLS

AR	wing aspect ratio
A_W	wetted area, ft^2
b	wing span, ft
BPR	bypass ratio
C_D	drag coefficient
C_{D_i}	induced drag coefficient

C_{D_o}	minimum drag coefficient
C_f	friction coefficient
C_L	lift coefficient
D	drag, lb
DOC	direct operating cost, cents/seat-statute mile
e	airplane efficiency factor
F_N	total net thrust, lb
FPR	fan pressure ratio
L	lift, lb
OPR	overall compressor pressure ratio
P	total pressure, lb/ft^2
SFC	specific fuel consumption, hr^{-1}
SPL	sound pressure level, dB
T_4	turbine inlet temperature, ^{o}R
V_R	relative velocity, ft/sec
W_F	total fuel weight, lb
W_G	gross weight, lb
W_L	payload, lb
W_P	installed propulsion system weight, lb
W_S	structure weight, lb

METHOD OF ANALYSIS

Range and payload were held fixed, and takeoff gross weight was calculated for each cycle. Since the fuel required for the mission is directly proportional to gross weight, the fuel fraction W_F/W_G for each cycle is constant. An arbitrary gross weight of 80 000 pounds was used to first calculate the fuel fractions and then an iteration on gross weight was performed, scaling the airframe and engines to meet the required payload.

Mission

A profile of the mission selected for the study is sketched in figure 1. It consists of (1) vertical takeoff and conversion to horizontal flight within an altitude of about 1000 feet, (2) climb, (3) Breguet cruise at Mach 0.75 and initial altitude of 20 000 feet, (4) descent to 1000 feet, and (5) conversion to vertical flight and landing. Total range is 500 statute miles with a payload of 100 passengers, or 20 000 pounds. Reserve fuel for an extended 550 statute mile range with a 20-minute hold at 5000 feet was included. This mission, to be flown on a 90° F day, is similar to one discussed in reference 8.

VTOL Transport

The airplane configuration used for the study is shown in figure 2. It consists of eight lift engines; two in each wing-tip pod, one in the aircraft nose, and three in the tail. The tail engines also serve as cruise engines.

An eight-engine configuration was chosen to best meet thrust/weight requirements under normal and engine-out conditions. During normal operation, an actual thrust/weight of 1.1 was assumed. The engines were sized to produce this thrust/weight while operating at 80 percent of their design thrust. The thrust margin is for control purposes. Control of the aircraft requires modulation of individual engine thrust while maintaining constant total thrust. Maximum available thrust/weight under normal operation, then, is 1.375; for engine-out operation, this requirement was lowered to 1.18. These thrust/weight requirements were estimated from reference 9 and are the same as were used in reference 7. Figure 3 indicates the amount of excess thrust that must be available during normal takeoff in order to meet these two F_N/W_G criteria. A configuration with less than seven engines requires each to be considerably oversized as regards normal operation - obviously, a nonoptimum situation. A very large number of engines, though, is undesirable from the operator's point of view as regards maintenance, probability of an engine malfunction, etc. Eight engines were chosen based on these considerations.

Estimates of wetted areas were made from a rough layout of the 80 000 pound airplane based on the similar-sized aircraft of reference 8. Airplane drag without wing-tip pods was calculated at the cruise condition from the wetted areas. A parabolic drag polar

$$C_D = C_{D_o} + \left(\frac{C_{D_i}}{C_L^2}\right)C_L^2$$

was assumed. C_{D_o} was calculated from the relationship

$$C_{D_o} = \frac{1}{4\left(\frac{L}{D}\right)_{max}^2\left(\frac{C_{D_i}}{C_L^2}\right)}$$

where

$$\left(\frac{L}{D}\right)_{max} = \frac{0.725\sqrt{\frac{\pi}{4}}\,b}{\sqrt{C_f A_W}}$$

and

$$\left(\frac{C_{D_i}}{C_L^2}\right) = \frac{1}{e\pi AR}$$

The equations for C_D, C_{D_o}, and (C_{D_i}/C_L^2) can be found in most textbooks on subsonic aerodynamics such as reference 10. The equation for $(L/D)_{max}$ is an empirical relationship based on fighter-bomber-transport configurations compiled by Langley Research Center. Wing loading was taken to be 110 lb/ft^2 and wing aspect ratio was 5.8. A flat-plate-friction coefficient was used where Reynolds number was based on two-thirds of fuselage length. Engine pod drag was estimated from a total drag coefficient calculated from empirical expressions for streamlined bodies of revolution found in reference 11. Cruise L/D was then obtained for each engine cycle. Lift/drag variation during climb was calculated for one cycle (FPR = 1.25, OPR = 10, T_4 = 2660° R) by a simple iteration on flight path angle. The straight-line Mach number-altitude relationship shown in figure 4(a) was used for the flight path. This slightly exceeds FAR 91.70 which restricts speed of all aircraft at altitudes below 10 000 feet to 250 knots or less. The resulting variation with Mach number is shown in figure 4(b). For the other cycles, this curve was simply scaled in proportion to the cruise L/D values. Descent L/D variation was obtained by linear interpolation between the cruise value and the value at start of climb.

Takeoff and landing fuel was calculated by assuming 1 minute of operation at takeoff thrust (F_N/W_G = 1 1). The method of reference 12 was used to calculate climb and descent fuel. Cruise fuel was calculated from the standard Breguet equation.

Takeoff Gross Weight Iteration

With fuel fractions known for each cycle, the simple relation

$$W_G = W_L + W_S + W_P + W_F$$

was used to calculate gross weights that met the 20 000-pound payload requirement. This equation can be rewritten as

$$W_G = \frac{W_L}{1 - \frac{W_S}{W_G} - \frac{W_P}{W_G} - \frac{W_F}{W_G}} \tag{1}$$

where

W_L	20 000 lb
W_F/W_G	fuel fraction
W_S	$f_1(W_G, W_P)$
W_P	$f_2(W_G$, cycle parameters)

The f_1 and f_2 are functions which scale the structure and installed propulsion system weights, respectively, with gross weight. The structures scaling was determined from body, wing, tail, landing gear, and flight control weight trends of reference 13. Fixed equipment weight for a 100-passenger aircraft was estimated from data in reference 3. The engine weight scaling used the bare engine weight to cycle parameter correlation of reference 14. Installation weight effects, including cruise nacelles, nozzles, inlet and outlet doors, engine mounts, lift pod cowling and inlets, were based on weight/unit area (or/lb of thrust) figures from reference 8. Bare engine dimensions, needed for both the installation and structures weight calculations were obtained from the correlation of reference 14.

With the scaling functions for structure and installed propultion weight determined, equation (1) was solved iteratively for W_G.

The proper variation of weight penalty with PNdB of suppression for these engines not being known exactly, all the engines were penalized equally, even though more suppression was required for the FPR = 1.3 engines as will be pointed out in the discussion on noise. Based on some preliminary and unpublished design data from a study of integral lift engines by General Electric, it was decided to use bare engine weight as calculated from reference 14 and to assume that it already included a 20-percent weight penalty for acoustic suppression. The study was first done under this assumption and then was repeated for one overall pressure ratio and turbine inlet temperature with weight penalties of 32, 44, and 56 percent for all fan pressure ratios. The $\Delta P/P$ drop in the duct and core nozzle were kept constant throughout the study.

Engines

An integral fan lift/cruise engine is shown in figure 5. It is a turbofan engine characterized by a low fan pressure ratio, a high bypass ratio, and relatively short length. It is referred to as an integral fan engine simply to differentiate it from the remote fan concept wherein a working fluid is ducted to individual tip-turbine lift fans. The splitter rings in the duct and in the turbine exhaust, as well as the duct walls, are lined with acoustic suppression material. Cruise performance calculations indicated that a variable area duct nozzle would be required, as shown in the figure.

Fan pressure ratios of 1.2, 1.25, and 1.3; overall pressure ratios of 7, 10, and 13; and turbine inlet temperatures of 2460°, 2660°, and 2860° R were examined. Pressure rise across the hub portion of the fan was assumed to be less than across the bypass section and was set at a constant 1.05 for all fan pressure ratios. Design point was at sea level static conditions on a 90° F day. All engine performance was calculated using the computer code GENENG II described in reference 15. This code uses actual component performance maps which it scales to input design point values of pressure ratio, mass flow, and efficiency. Design values of adiabatic efficiency, pressure losses, and velocity coefficients were chosen as follows:

Fan efficiency (bypass)	0.87
Fan efficiency (core)	0.84
Compressor efficiency	0.85
Combustor efficiency	0.9875
HP turbine efficiency	0.87
LP turbine efficiency	0.83
Inlet recovery	0.99
Combustor pressure loss, $\Delta P/P_{in}$	0.07
Core nozzle pressure loss, $\Delta P/P_{in}$	0.03
Duct pressure loss, $\Delta P/P_{in}$	0.02
Core nozzle velocity coefficient	0.99
Duct nozzle velocity coefficient	0.99

A cooling bleed schedule with turbine inlet temperature, representative of convection cooling, was incorporated into the cycle performance calculations. The schedule is shown in figure 6(a). Figure 6(b) shows the total cooling bleed split between high pressure and low pressure turbines.

The values used for the component efficiencies and cooling bleed are representative of a design approach which emphasizes weight savings, low cost, and maintainability.

Design point ($F_N/W_G = 1.375$), takeoff and noise-rating point ($F_N/W_G = 1.1$), start of climb, and cruise performance were calculated for all cycles. Climb and descent performance was calculated for one cycle (FPR = 1.25, OPR = 10, $T_4 = 2660^\circ$ R). From this, an interpolation scheme was devised for the other cycles to determine SFC, thrust, etc. along the climb and descent paths. Given the L/D ratio, SFC, and thrust along the flight path, the fuel fractions for each segment of the mission, for each cycle, and for an 80 000 pound gross weight were calculated.

Noise and Bypass Ratio

Bypass ratio for each cycle was selected to meet a 500-foot altitude flyover noise goal of 95 PNdB. Three sources of noise were considered - core jet, duct jet, and fan machinery. The noise rating condition was based on average engine thrust during normal takeoff or landing. For the 80 000-pound gross weight, the F_N/W_G requirement of 1.1 specified an overall net thrust of 88 000 pounds.

The fan machinery noise versus fan pressure ratio is shown in figure 7 for a thrust level of 88 000 pounds. It was derived from estimates of single-stage fan machinery noise in reference 16, by adjusting for thrust level and distance. Figure 7(a) shows anticipated improvements with 15 and 18 PNdB of reduction, as well as the unsuppressed machinery noise.

Jet noise was calculated by the method described in reference 17. At jet velocities below 1000 feet per second, there is uncertainty as to how sound pressure level varies. In this study, the SPL equation of reference 17 and the relationship $f(V_R)$ between SPL and relative jet velocity (fig. 1 of ref. 17) were modified to obtain agreement with recent jet noise data published in reference 16. Combined jet and machinery noise levels were obtained by adding logarithmically octave SPL's after assuming a typical machinery noise spectrum with peak frequency in the sixth octave.

The duct jet noise, being primarily a function of jet velocity, did not vary noticeably for constant fan pressure ratio and for the range of bypass ratios being considered. The assumed schedule of fan machinery noise, too, does not account for any variation with bypass ratio. These two components of noise, then, were specified by the 88 000-pound thrust level and the fan pressure ratio. The difference between their sum and the noise goal represented how much the core jet could contribute to the total and still meet the goal. By increasing bypass ratio, the core jet noise could always be made low enough to meet the noise goal, as long as machinery and duct jet noise did not already exceed it.

From figure 7(b), which shows machinery and duct jet noise singly and combined, it can be deduced that design fan pressure ratios of 1.2 and 1.25 with 15 PNdB of machinery noise suppression could meet the goal with some bypass ratio. It can also be seen that a fan pressure ratio of 1.3 with 15 PNdB of machinery noise suppression would not meet the goal with any bypass ratio. For the 1.3 fan pressure ratio, 18 PNdB of machinery noise suppression was assumed.

It should be pointed out that a tradeoff between the amount of suppression and BPR would result in an optimum value of each. This was not done in the study for two reasons. First, the possible variation in amount of machinery noise suppression was so constrained by the maximum considered achievable (18 to 20 PNdB) and the minimum required for just machinery plus duct jet noise to meet the goal, that the addition of an extra variable into the study, BPR, did not seem warranted. Secondly, the proper variation of weight penalty with suppression was not known, and so was initially ignored. It is recognized also that $\Delta P/P$ penalties resulting from acoustic suppression vary significantly, and although a sizable duct pressure loss was included, it was held constant throughout the study.

Direct Operating Cost

DOC was calculated for each cycle using the relations presented in reference 18. Flight crew costs were based on a flight crew of 2. Fuel and oil were assumed to cost 1.5 and 92.6 cents per pound, respectively. Insurance cost was calculated at a 3-percent rate of the initial airplane cost. Maintenance cost assumed an hourly labor rate of $5.00. A yearly utilization rate of 3000 hours was determined from reference 19 for the approximate block time of 1 hour. Initial cost of the aircraft was based on an airframe cost of \$71 per pound and an engine cost of \$100 per pound of weight. Included in the total aircraft cost was \$350 000 for avionics, the figure used in reference 18. The materials and labor maintenance cost equations of reference 18 were modified to account for the different amount of usage that the lift/cruise and pure lift engines would receive.

RESULTS AND DISCUSSION

The bypass ratios for each cycle, picked such that at 80 percent design thrust (F_N=W_G = 1.1), the total noise of eight engines just equalled 95 PNdB at 500-foot altitude, are shown in figure 8. The noise calculations which determined these bypass ratios, however, were with engines sized for an 80 000-pound aircraft. The final gross weights, and consequently engine size, were 30 to 40 percent higher, but since a doubling of gross weight would raise the noise level by only 3 PNdB, this increase in gross weight did not seem to justify the additional iteration.

The higher fan pressure ratios, which extract more energy from the core stream by virtue of pressure ratio, do not require as high a bypass airflow to achieve low core exit velocity. Increasing turbine inlet temperature adds energy to the core stream and requires a higher bypass ratio.

In calculating cruise performance, a plot of engine thrust versus SFC was desired for each cycle by varying the turbine inlet temperature. The SFC corresponding to the thrust required by cruise L/D would then be used in the cruise fuel calculation. None of the engines, however, would operate satisfactorily at cruise unless the duct nozzle areas were reduced from their design point values. The reason for this requirement can best be illustrated by the fan performance map in figure 9. With increasing altitude and Mach number, the fan operating point tends toward increasing airflow, slightly lower pressure ratio, and most importantly, poorer efficiency. The result being that the operating point of all fan pressure ratios, at the cruise condition, would be beyond the extent of the fan map (A-B).

To simply extend the map and continue in that direction would mean rapidly decreasing efficiency and a point would be reached where the core would simply not have sufficient energy to drive the fan. Moving the fan design point toward the surge line increases the altitude and Mach number at which the operating point falls off the map to about half the cruise values, but does not eliminate the problem. By decreasing duct nozzle area, though, airflow decreases, pressure ratio increases, and most importantly, the operating point returns to a good efficiency region of the map (B-C). Cruise performance was therefore run for each cycle at varying duct nozzle area reduction. The best SFC and highest thrust for a given turbine inlet temperature were obtained with duct nozzle area reductions of about 25, 20, and 15 percent for the 1.2, 1.25, and 1.3 fan pressure ratios, respectively. A constant weight penalty was included in the propulsion system weight to account for the variable area nozzles.

The final gross weights calculated for each cycle are shown in figure 10. All engine cycles provided adequate thrust for cruise except the FPR = 1.2, OPR = 10, $T_4 = 2460^{\circ}$ R case, implying that it would have to be sized for the cruise condition. The greatest variation in gross weight is due to fan pressure ratio. The lowest gross weight of 102 200 pounds was achieved with the highest fan pressure ratio of 1.3. This was over 8000 pounds less than the best 1.2 fan pressure ratio cycle. This can be explained by the fact that higher fan pressure ratio means a higher thrust/unit airflow and, consequently, lower airflow, smaller diameter, and lighter engine for the same thrust. In a gross-weight mission analysis, though, thrust is not held fixed; a lighter engine implies lower gross weight and even less thrust required. The lowest gross weight was also achieved at the highest turbine inlet temperature considered, 2860° R. There is a trend to be seen, though which implies ever-diminishing returns with higher and higher temperatures. Crossplotting the data against T_4 shows that 2860° R is very nearly optimum, and higher T_4 would not yield any further reduction in gross weight. For each value of turbine inlet temperature, an optimum in overall pressure ratio was reached. The lowest gross weight occurred at an overall pressure ratio of about 12.

Figure 11 shows airplane component weight trends with fan pressure ratio for an overall pressure ratio of 10 and a turbine inlet temperature of 2660° R. The curves labeled "ENGINES" are uninstalled, but acoustically-treated engine weights; inlets, nozzles, pods etc., are included under "AIRFRAME" weight.

The trends in DOC with cycle parameters shown in figure 12 are essentially the same as the gross weight trends. The overall optimum is at the same values of cycle parameters - FPR = 1.3, OPR = 12, and $T_4 = 2860^{\circ}$ R. The average values of DOC, based on 1972 dollars, are roughly twice those of present-day commercial aircraft. The two largest portions of the DOC were maintenance and depreciation, both of which were unfavorably affected by the large number of engines.

All the results presented thus far were obtained under the assumption of an equal acoustic treatment weight penalty, that is, 20 percent of bare engine weight, for all fan pressure ratios. Figure 13 shows the results of varying this assumption, on gross weight and DOC. The gross weight iteration was repeated with 32, 44, and 56 percent weight penalties for an overall pressure ratio of 10, turbine inlet temperature of 2660° R, and for each fan pressure ratio. It can be seen from the figure that a weight penalty increase of about 30 percent (in terms of percent bare engine weight) on the 1.3 FPR would eliminate any gross weight and DOC advantage it otherwise would have over the 1.2 FPR engine. Similarly, an increase of about 10 percent on the 1.25 FPR would eliminate any gross weight and DOC advantage that it had over the 1.2 FPR. The weight penalties involved, then, in acoustically treating these engines is going to be a determining factor in the selection of a fan pressure ratio for a VTOL design.

As an example, consider the suppression versus fan pressure ratio schedules shown in figure 14. Curve A corresponds to what was used in the study. Curve B differs only at FPR less than 1.25. It can be deduced from figure 7(b) that at these FPR's the noise goal could have been met with less than 15 PNdB of suppression, but with higher BPR's. Curve B corresponds to a minimum of machinery noise suppression and a maximum of core jet noise reduction by BPR. Curve C is 2 PNdB higher than curve B and is included for the sake of comparison. It represents the case of more machinery noise suppression and less core jet noise reduction by means of BPR. It should be noted that the BPR's associated with each cycle here are really correct at each FPR only for schedule A. Now also consider the suppression weight schedule shown in figure 15. The shape of the curve was based on data taken from reference 20. The level was adjusted to agree with as yet unpublished data from a design study of integral fan lift engines by General Electric. This weight schedule, scaled for engine diameter and length, can now be related to the curves of figure 13. This was done for the three suppression schedules A, B, and C. The results are shown in figure 16 where they are superimposed on a replot of figure 13. Comparing these sample suppression curves with the originally assumed constant 20 percent penalty curve, it can be seen that the highest FPR has suffered the most. In fact, for schedule C, the gross weight and DOC values are nearly the same for the 1.3 and 1.2 FPR cases. An optimum is reached at a FPR of about 1.26.

CONCLUDING REMARKS

A study was made of turbofan-powered VTOL aircraft, carrying 100 passengers fro 500 statute miles at a cruise Mach number of 0.75. Eight integral fan lift engines were assumed, 3 of which were also used for cruise. The engines were constrained to meet a noise goal of 95 PNdB at 500 feet.

Within the range of cycle parameters studied, the lowest gross weight of 102 200 pounds and best DOC of 1.82 cents/seat-mile were obtained with a fan pressure ratio of 1.3, overall pressure ratio of 12, and turbine inlet temperature of 2860° R. This, however, is presuming that 18 PNdB of machinery noise suppression is achievable for a 1.3 FPR engine at a weight penalty equal to that (in terms of percent bare engine weight) for a 1.2 and 1.25 FPR with 15 PNdB of suppression.

An increase in the weight penalty from 20 to about 50 percent on the 1.3 FPR eliminates the gross weight and DOC advantage it had over the 1.2 FPR. An increase to about 30 percent does the same to the 1.25 FPR.

The sensitivity study, combined with some sample suppression and suppression weight schedules, indicates that the optimum fan pressure ratio for an integral fan lift engine meeting a noise goal is highly dependent on the real weight penalties that will be brought about by acoustic treatment.

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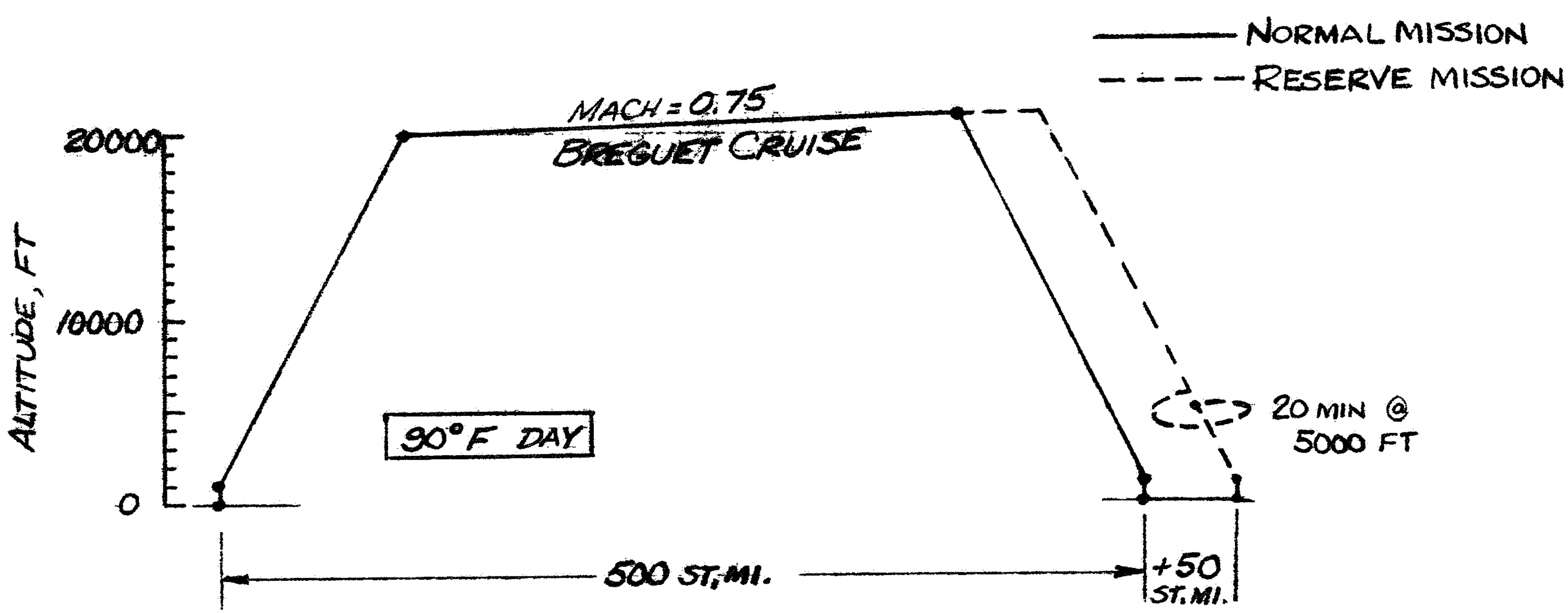


FIGURE 1. — MISSION PROFILE.

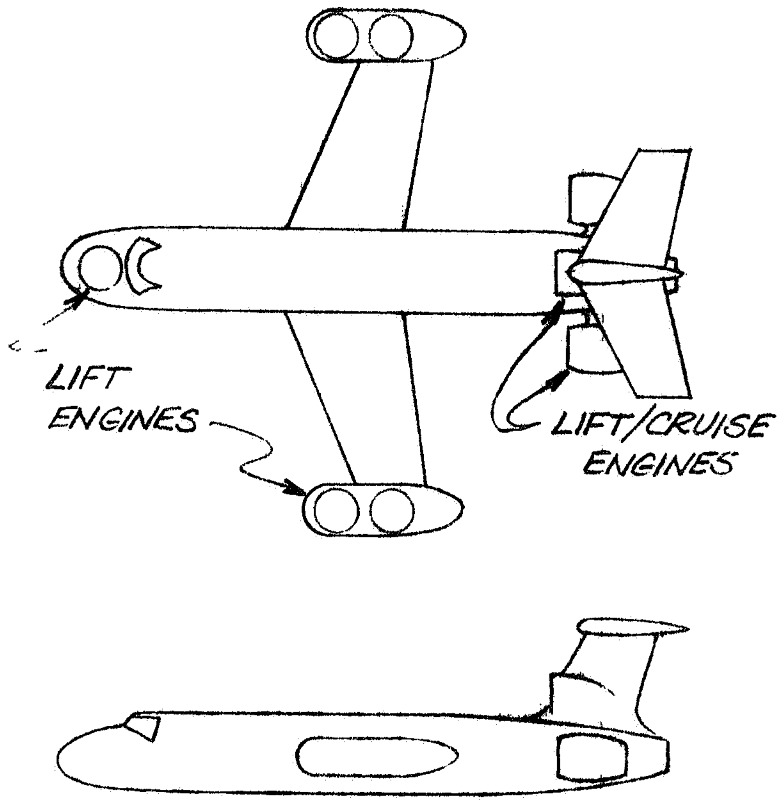


FIGURE 2.— CONCEPTUAL SKETCH OF 100-PASSENGER VTOL TRANSPORT WITH EIGHT INTEGRAL LIFT ENGINES

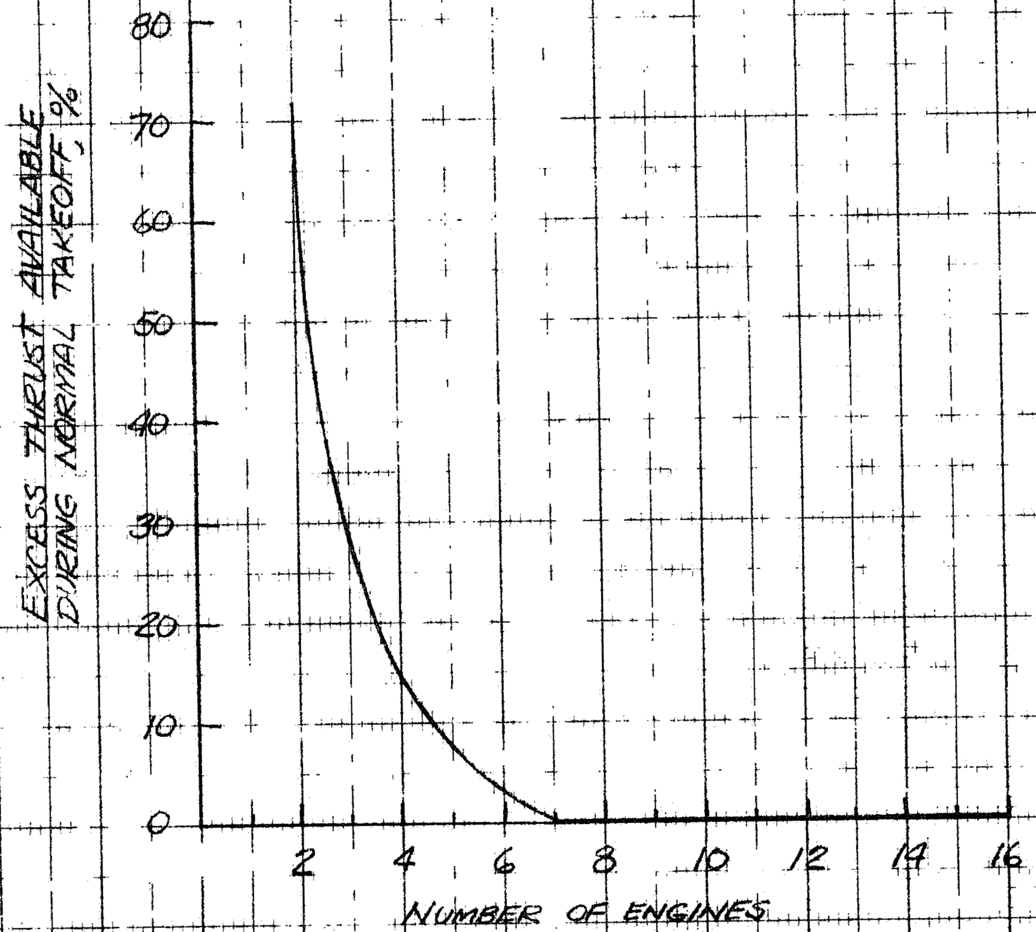
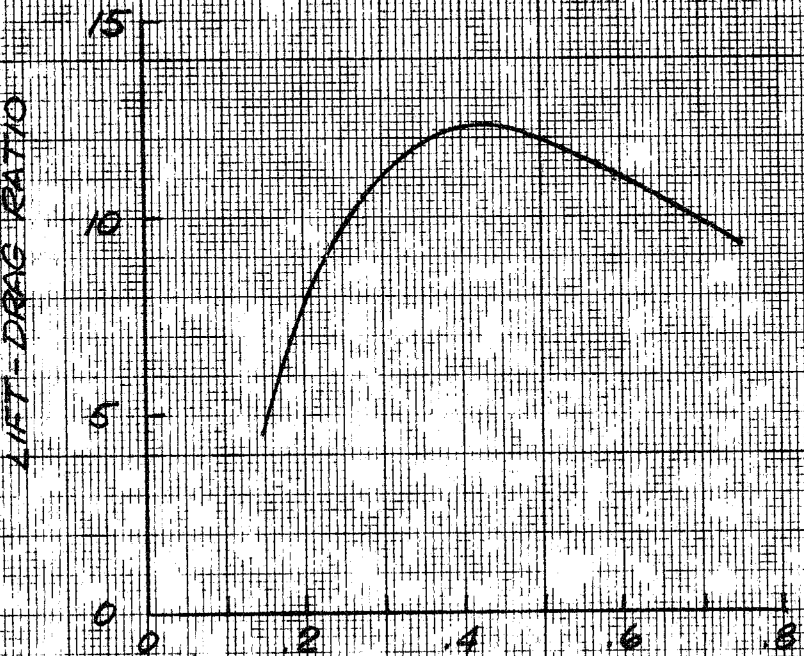
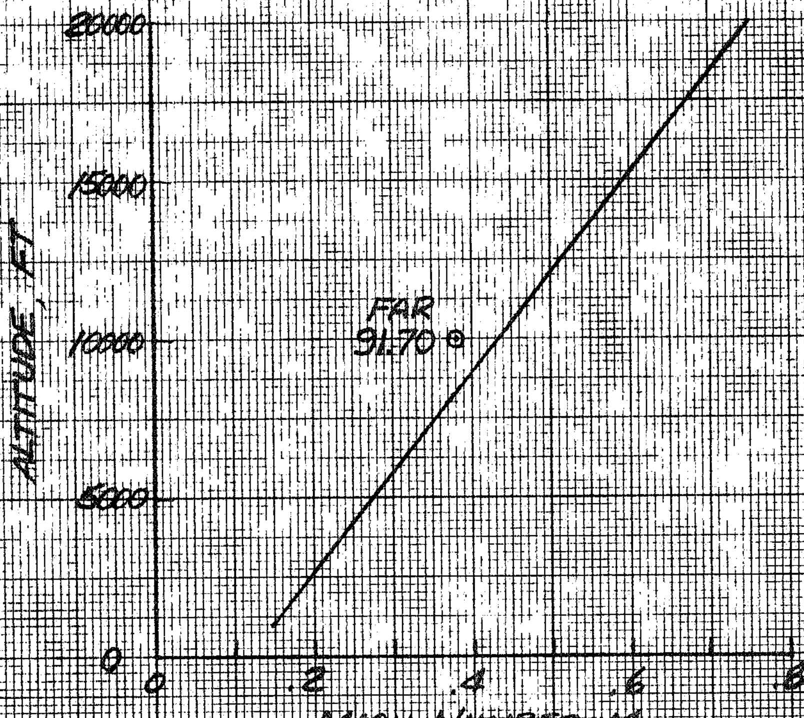


FIGURE 3. — EXCESS THRUST AVAILABLE PER ENGINE DURING NORMAL TAKEOFF VS. NUMBER OF ENGINES. NORMAL THRUST/WEIGHT REQUIRED IS 1.375. ENGINE-OUT THRUST/WEIGHT REQUIREMENT IS 1.18125.

$L/D = (((-12.21M + 156.7)M - 254.6)M + 134.2)M - 10.32$



b) (L/D) vs. MACH NO. (FOR FPR=1.25, OPR=10, T_4=2660°R)



a) ALTITUDE vs. MACH NO.

FIGURE 4. — FLIGHT PATH AND LIFT-DRAG RATIO.

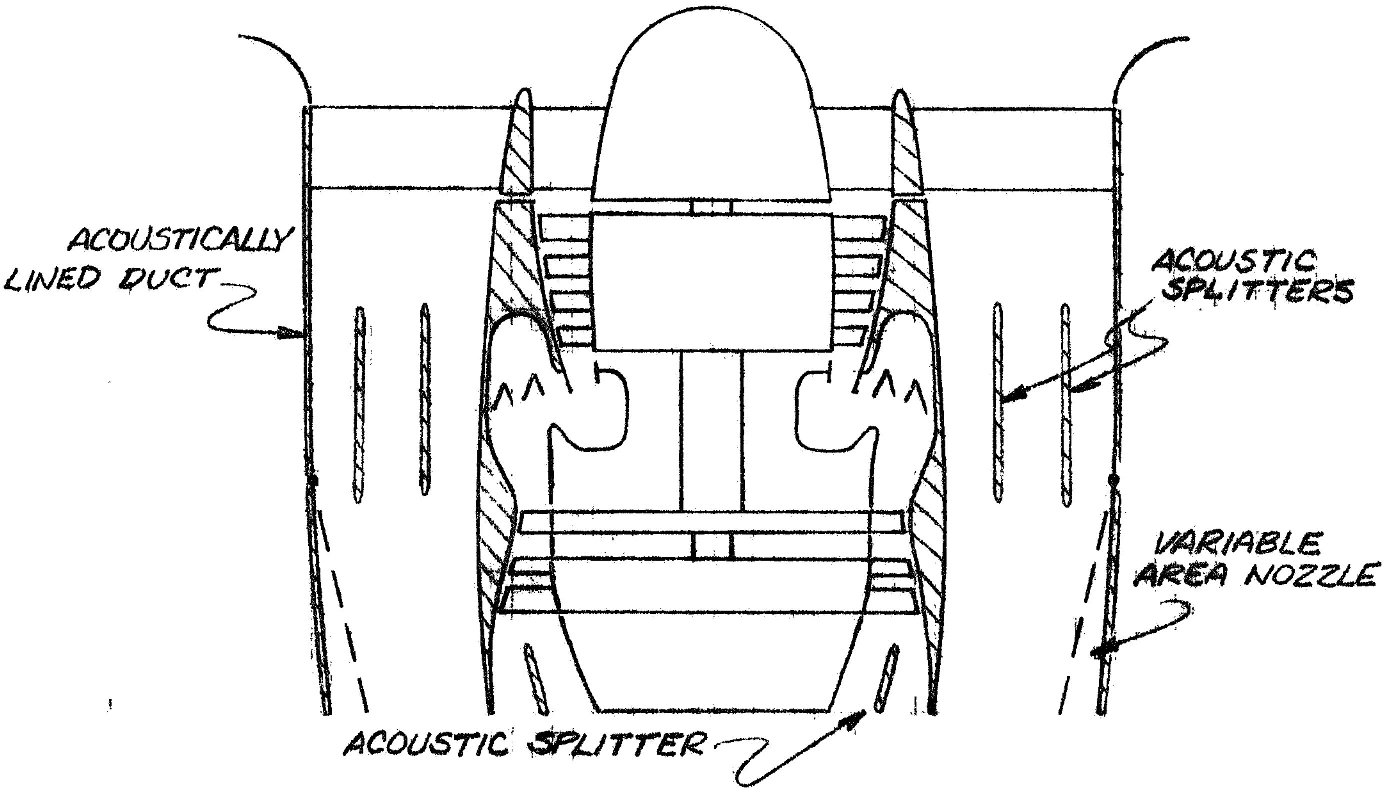
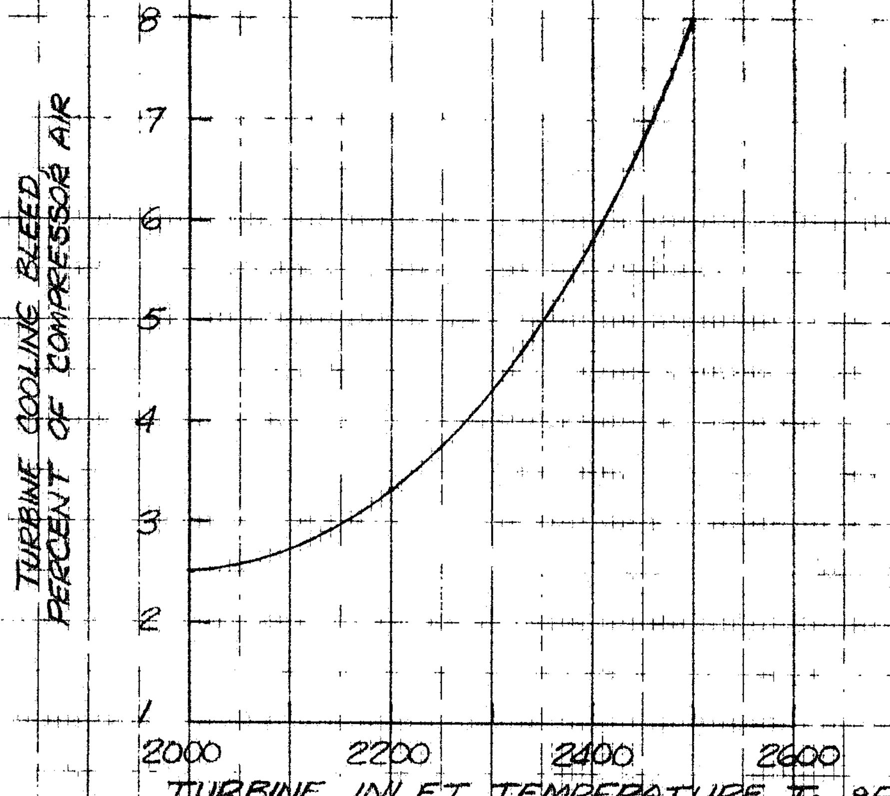
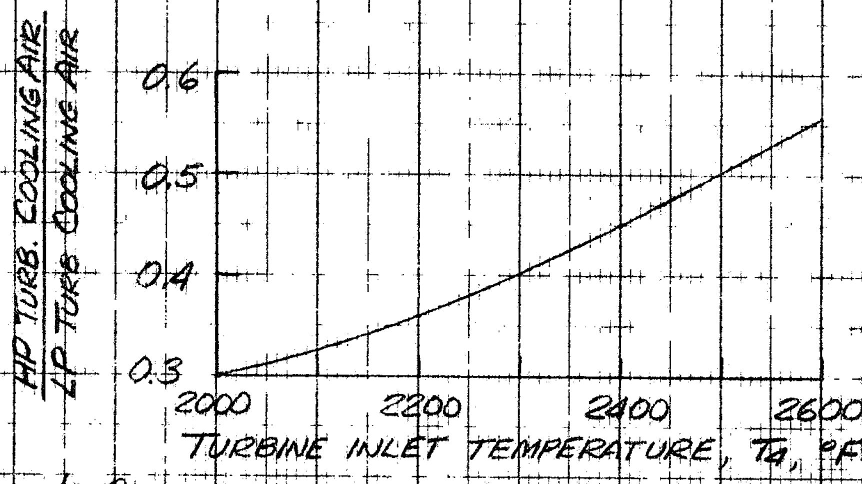


FIGURE 5.— SCHEMATIC OF INTEGRAL FAN
LIFT/CRUISE ENGINE

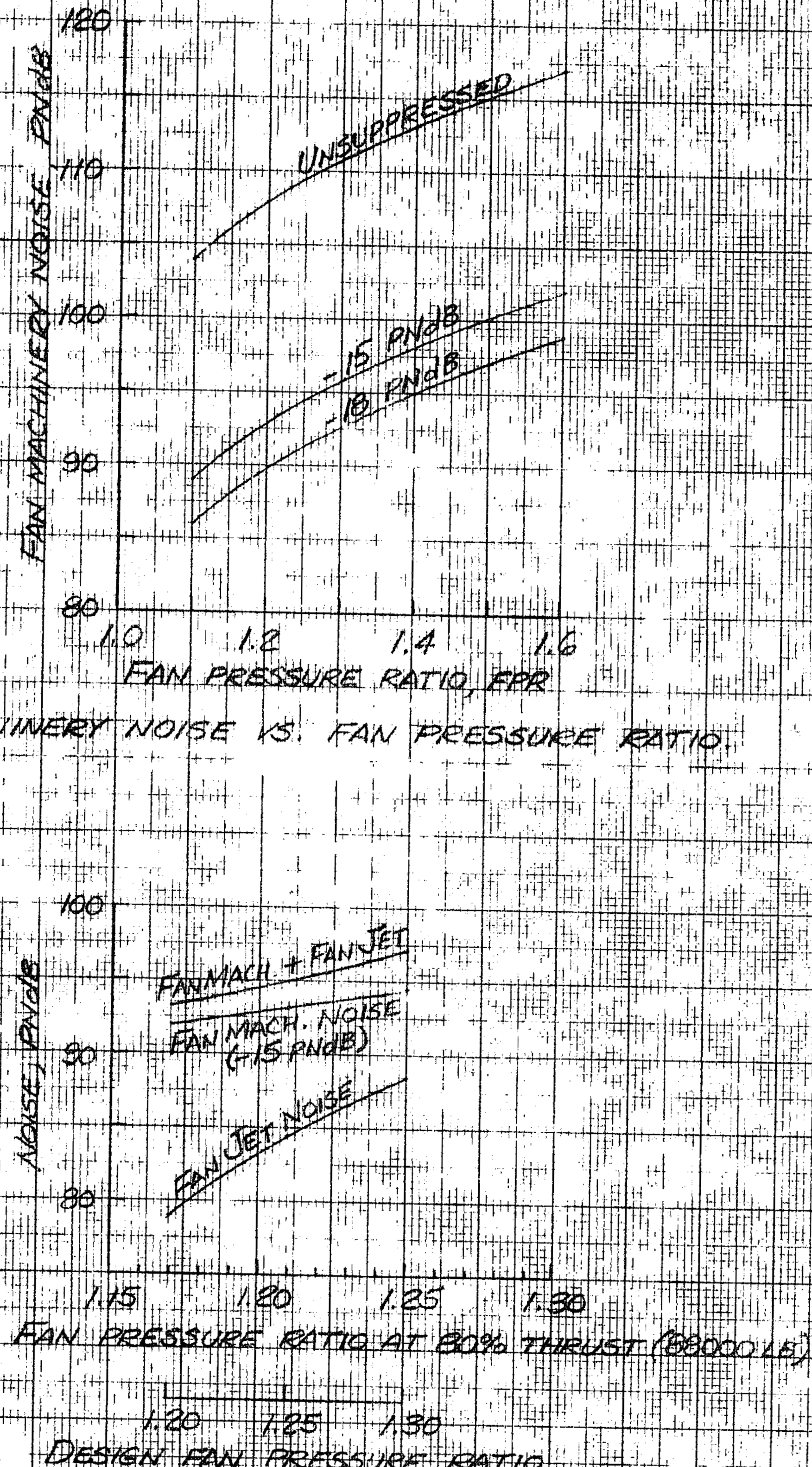


a) TURBINE COOLING BLEED VS INLET TEMPERATURE



b) COOLING BLEED SPLIT BETWEEN HIGH AND LOW PRESSURE TURBINES

FIGURE 6. — TURBINE COOLING BLEED.



a) FAN MACHINERY NOISE VS. FAN PRESSURE RATIO.

b) FAN JET AND MACHINERY NOISE VS. FAN PRESSURE RATIO.

FIGURE 7. — FAN JET AND MACHINERY NOISE. THRUST: 88000 LB, ALTITUDE = 500 FT.

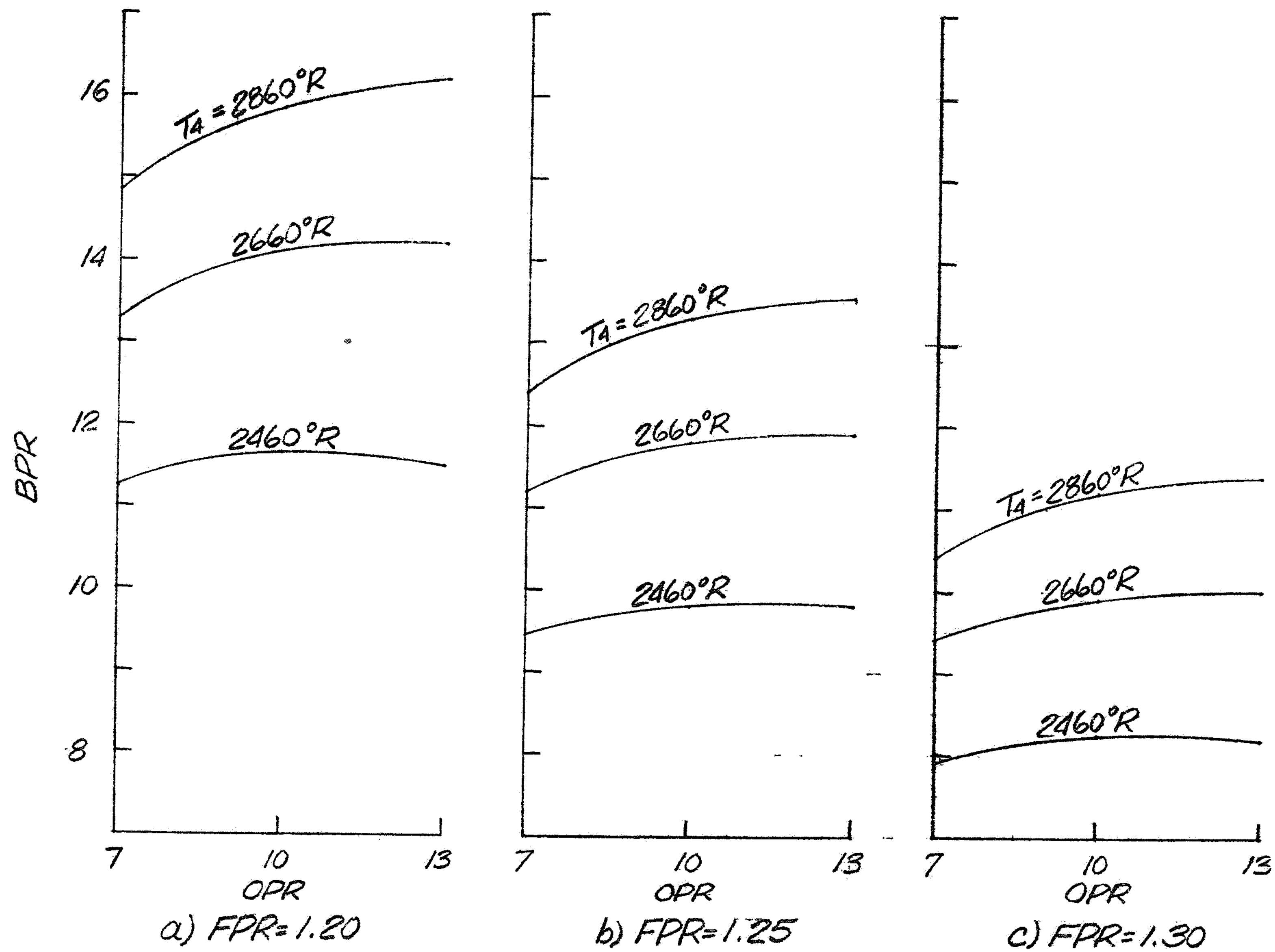


FIGURE 8. – BYPASS RATIO AS A FUNCTION OF FAN PRESSURE RATIO, OVERALL PRESSURE RATIO, AND TURBINE INLET TEMPERATURE, FOR MEETING A FLYOVER NOISE GOAL OF 95 PNdB AT 500 FT ALTITUDE.

CHKD. BY DATE JOB NO.

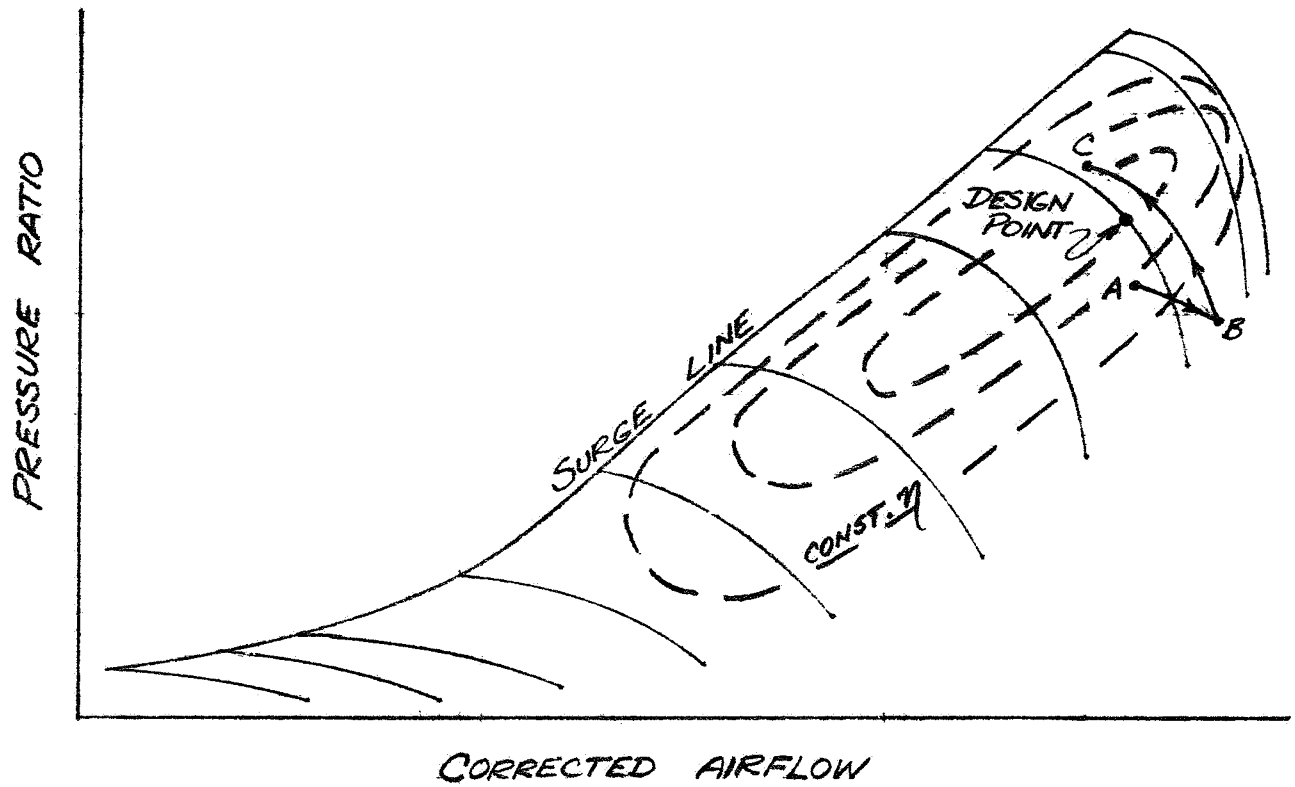


FIGURE 9.— TYPICAL FAN PERFORMANCE MAP SHOWING EFFECTS OF INCREASING ALTITUDE AND MACH NO. (A-B), AND DECREASING DUCT NOZZLE AREA (B-C).

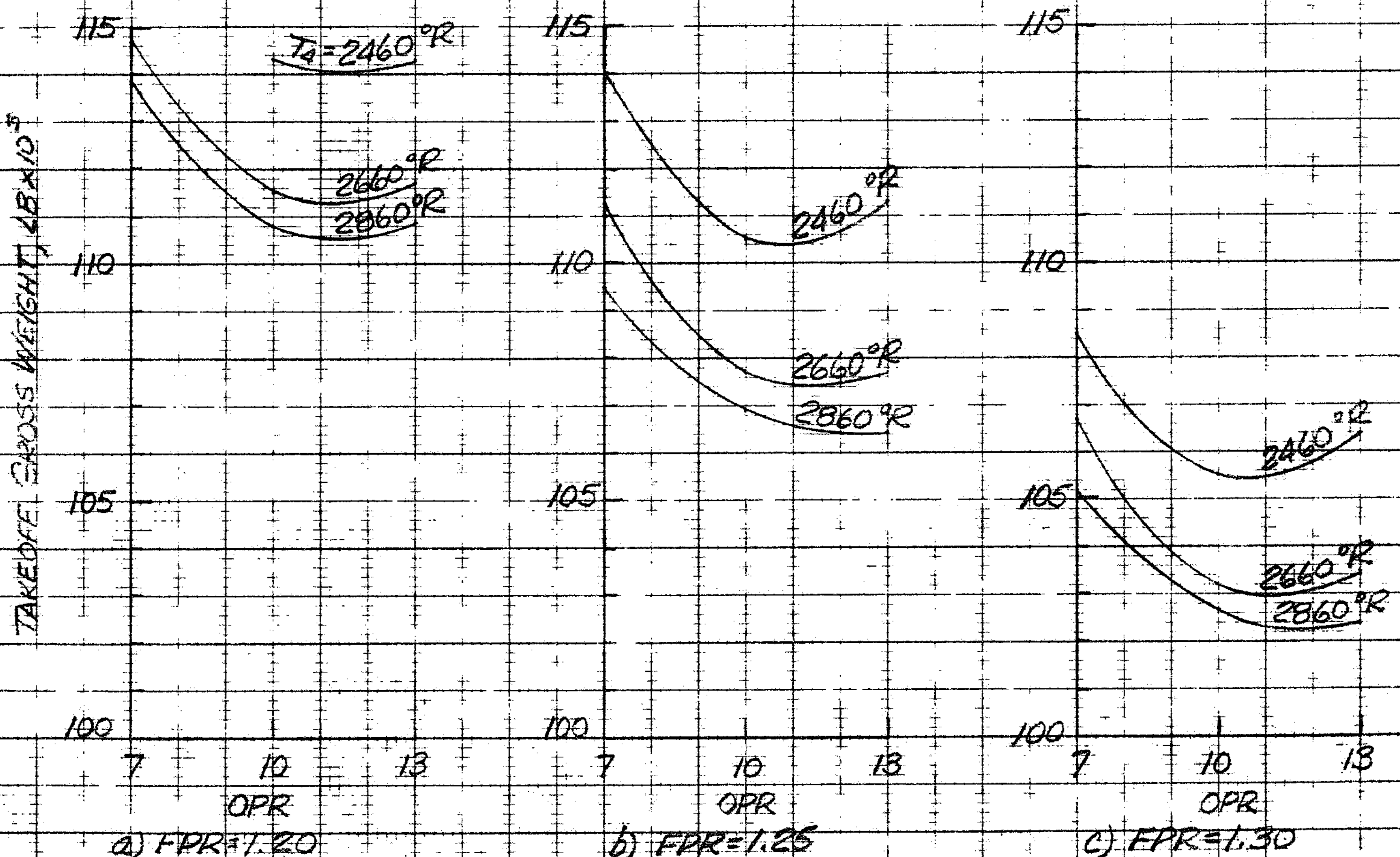
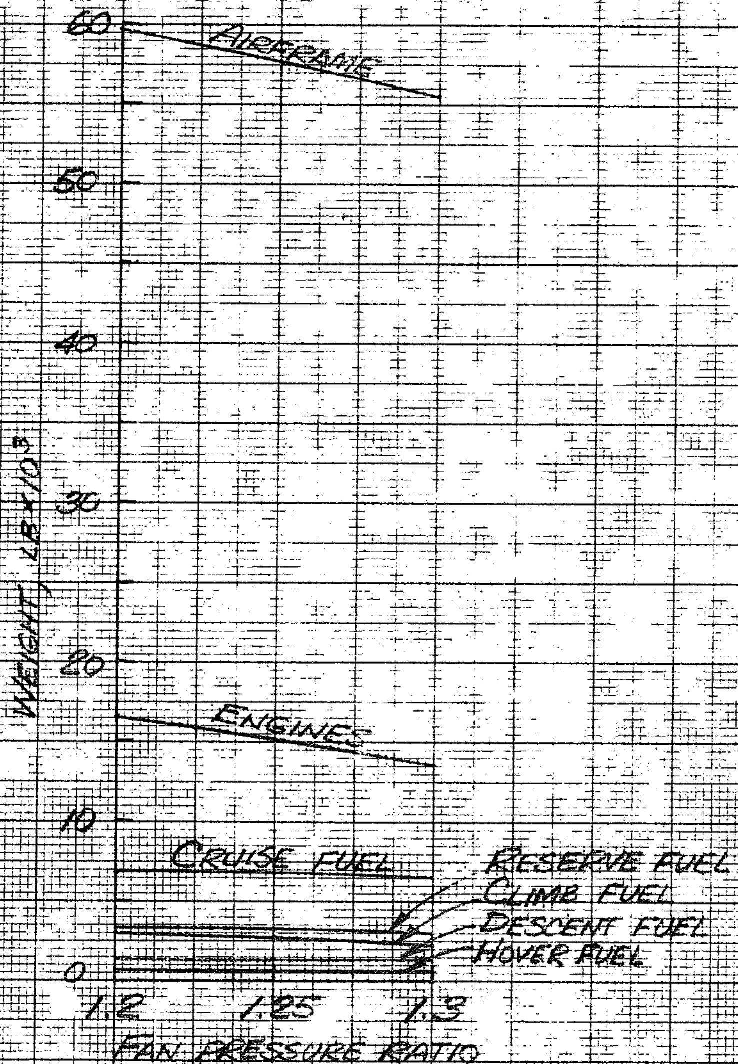
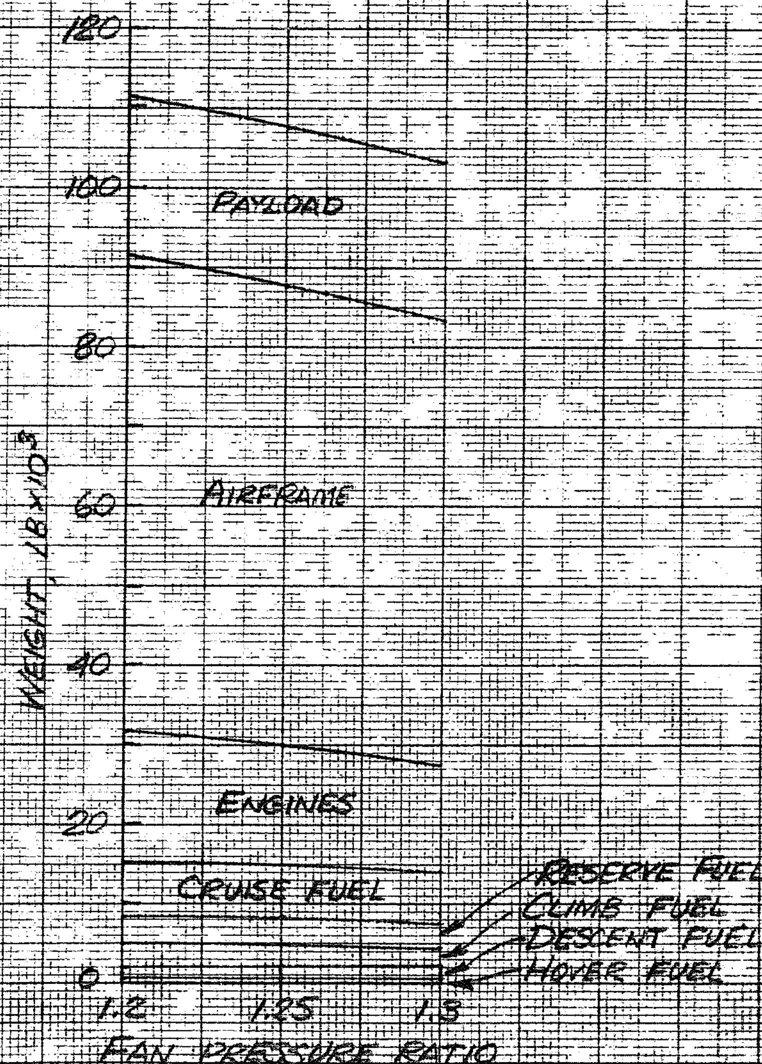


FIGURE 10. — TAKEOFF GROSS WEIGHT AS A FUNCTION OF FAN PRESSURE RATIO, OVERALL COMPRESSOR PRESSURE RATIO, AND TURBINE INLET TEMPERATURE.



a) INDIVIDUAL WEIGHT TRENDS



b) CUMULATIVE WEIGHT TRENDS

FIGURE 11. — WEIGHT TRENDS WITH FAN PRESSURE RATIO, OVERALL PRESSURE RATIO = 10, TURBINE INLET TEMPERATURE = 2260°R.

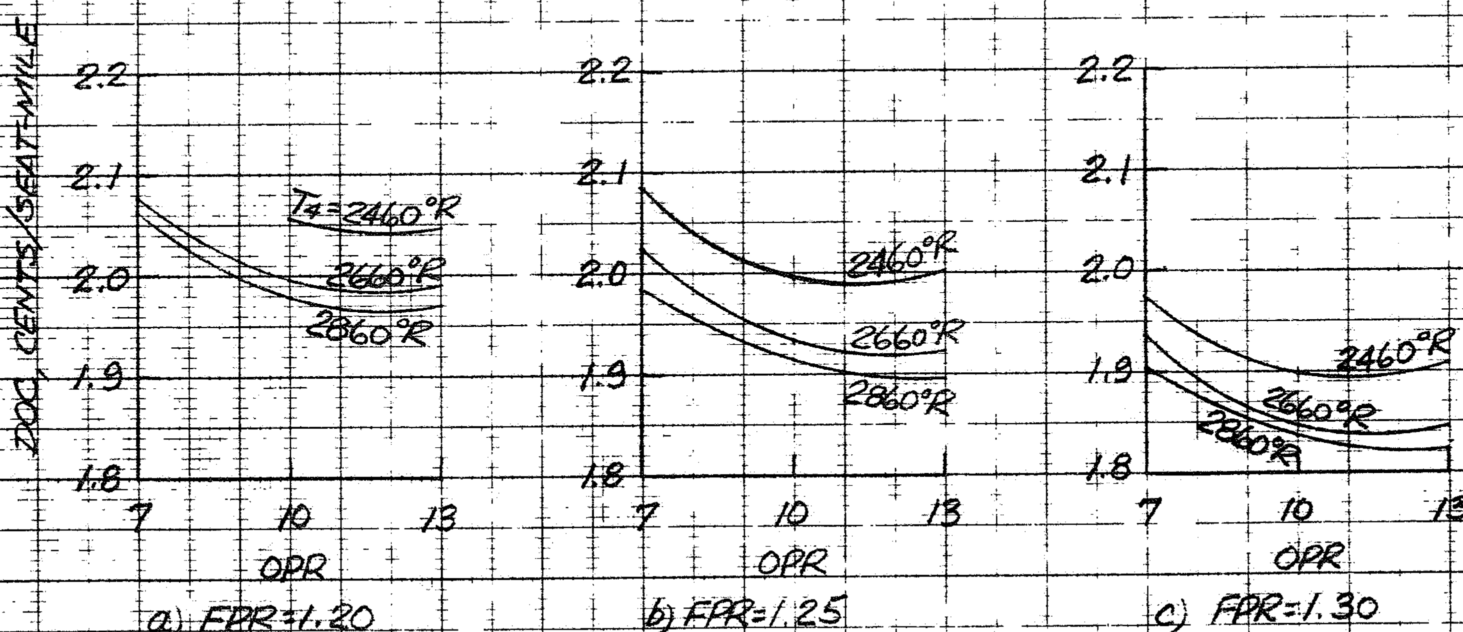


FIGURE 12. — DIRECT OPERATING COST AS A FUNCTION OF FAN PRESSURE RATIO, OVERALL COMPRESSOR PRESSURE RATIO, AND TURBINE INLET TEMPERATURE

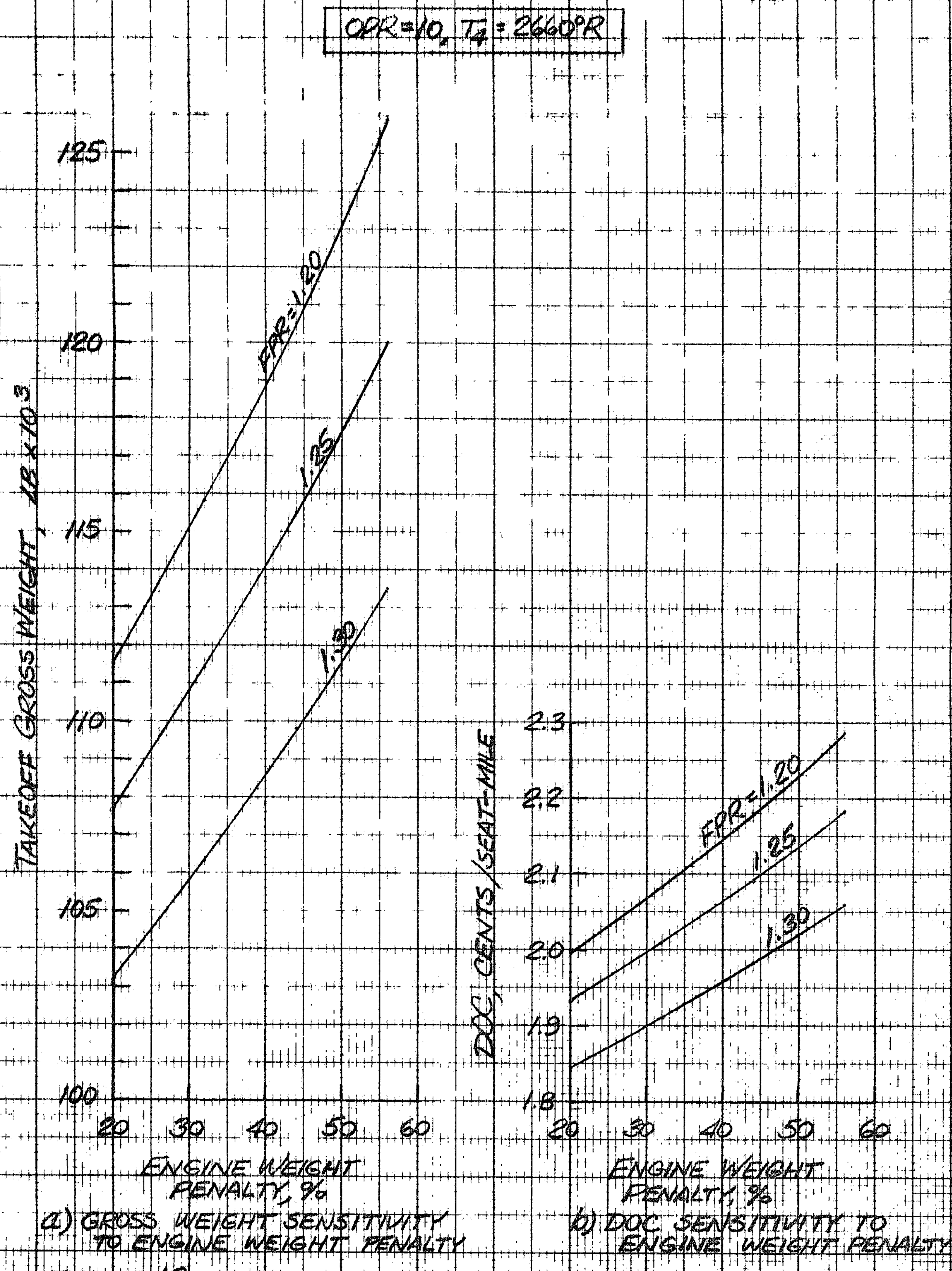


FIGURE 13. — GROSS WEIGHT AND DOC SENSITIVITY TO ENGINE WEIGHT PENALTY WITH FAN PRESSURE RATIO AS A PARAMETER (OPR=10, T_4=2660°R)

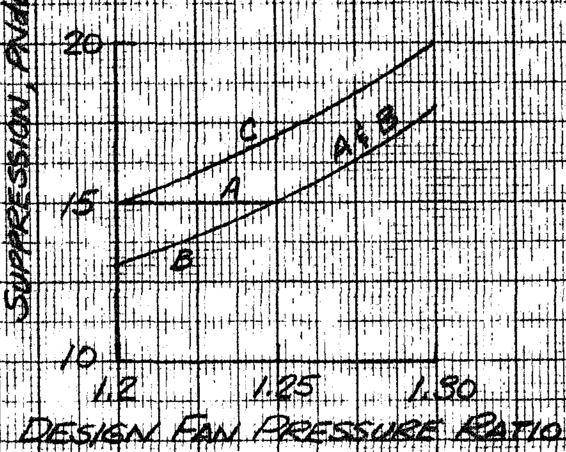


FIGURE 14. — THREE SCHEDULES OF MACHINERY NOISE SUPPRESSION WITH FAN PRESSURE RATIO.

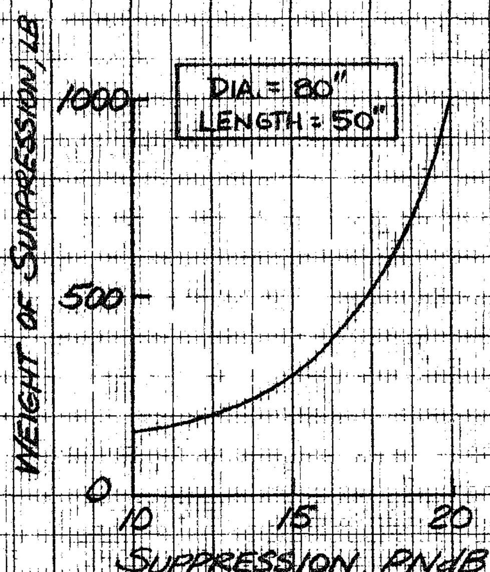


FIGURE 15. — SUPPRESSION WEIGHT SCHEDULE WITH PNdB OF SUPPRESSION.

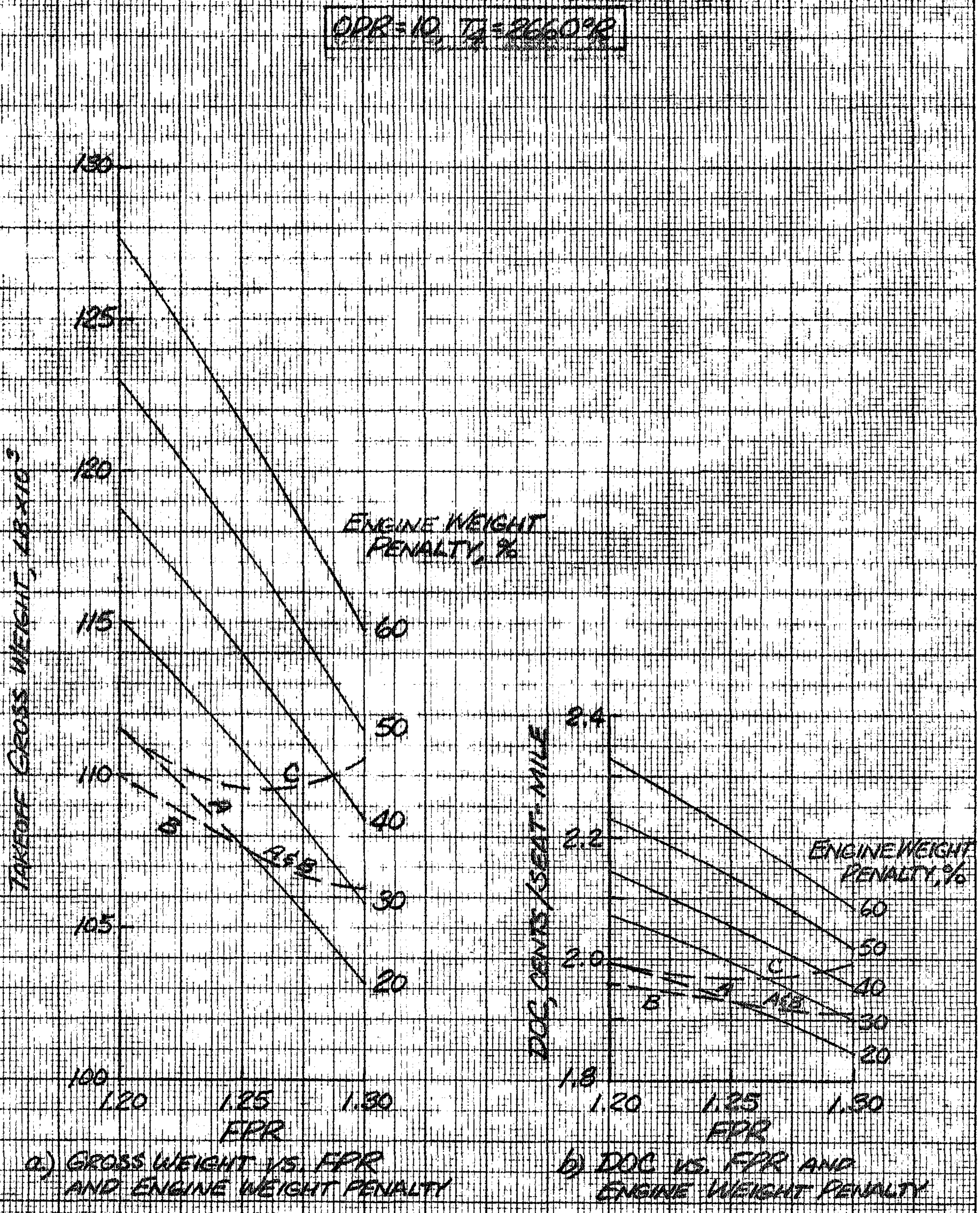


FIGURE 16. — GROSS WEIGHT AND DOC VARIATION WITH FAN PRESSURE RATIO FOR THREE SUPPRESSION SCHEDULES.