

**THE COST OF
NOISE REDUCTION FOR
POWERED-LIFT
STOL AIRCRAFT**

by **Wesley F. Moore**

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THE COST OF NOISE REDUCTION
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ABSTRACT

The direct operating costs and noise impacts of a wide variety of Externally Blown Flap and Augmentor Wing STOL short haul transport aircraft designs were evaluated to study the costs of noise reduction for these types of aircraft.

The "two-stream" Augmentor Wing designs were found to be capable of the greatest noise reductions, and to have the lowest direct operating costs at all levels of noise impact. Sideline noise levels of 81 PNdB at 500 feet were attainable for an 80 seat aircraft with an 8 to 15 percent increase in direct operating cost over an aircraft designed with no constraints on noise.

ACKNOWLEDGEMENTS

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I. INTRODUCTION

During the past several years, the design of aircraft able to operate in the very restrictive environments of city centers has occupied the attention of the manufacturers, airlines, and agencies that make up the world's aviation community. These V/STOL (for Vertical and/or Short Takeoff and Landing) aircraft would be able to use landing fields less than 2000 feet in length, fly steep takeoff and landing flight paths, and be maneuverable enough to avoid the numerous obstructions encountered in an urban environment.

During the past few years, however, the problem of "noise pollution" has become the urban citizens' most frequent contact with air transportation. It now appears that no aircraft will be able to operate from small fields (with the closeness to other urban activities that the small size implies) without meeting very restrictive noise standards which have yet to be promulgated.

It is possible for the designer to make these aircraft very quiet, just as it is possible to make aircraft fly at the low speeds necessary to operate from small fields. There is a cost for noise reduction, however, just as there is a cost for V/STOL performance.

This report studies two STOL concepts of current interest,

the Externally Blown Flap, and the Augmentor Wing. These "powered-lift" concepts enable flight at low speeds with high wing loadings, but tend to result in noisy aircraft (compared to conventional aircraft).

In order to make these powered-lift aircraft quiet enough to meet or exceed proposed noise standards for V/STOL aircraft, it is necessary to sacrifice economy for quietness, through the use of wing sizes and engine parameters that are non-optimum from the direct operating cost outlook.

Through the use of a computer program that synthesized aircraft designs, and evaluated their operating costs and noise "footprints", it was possible to study a large number of designs using both concepts. For each level of noise generation, a design of minimum operating cost was found, resulting in a pattern of cost-versus-noise for both STOL concepts.

II. POWERED LIFT STOL AIRCRAFT DESIGN

A. POWERED LIFT FLIGHT CONCEPTS

The two types of STOL aircraft discussed in this report rely on the "jet flap" principle for flight at low speeds. In a jet flap, the momentum of a jet of air flowing from the trailing edge of an aerofoil increases its effective lift, both by the direct thrust of the vertical component of momentum, and by enhancing the circulatory flow about the aerofoil. Suitable secondary jets or slots can further enhance the circulation by sucking or blowing on the boundary layers, thus keeping the flow attached and well behaved. Figure 1a shows the essentials of the concept.

The Externally Blown Flap (EBF) concept creates the jet flap at the trailing edge by immersing large double-or triple-slotted flaps in the exhaust flow of high bypass turbofan engines (see Figure 1b). This arrangement looks like a typical underwing arrangement for conventional aircraft; in those, however, cut-outs are provided so that the flaps do not deflect the thrust. Furthermore, the EBF engines are as close to the wing undersurface, and grouped close together as practical. This concept has the advantage of simplicity, but suffers from problems of engine-out controllability and efficiency; for example,

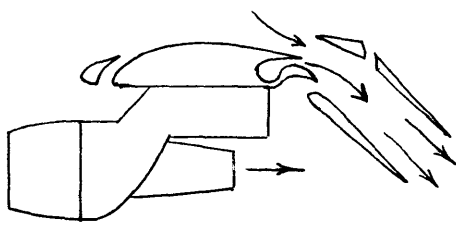
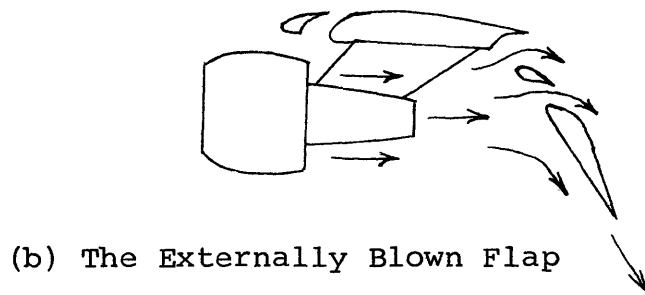
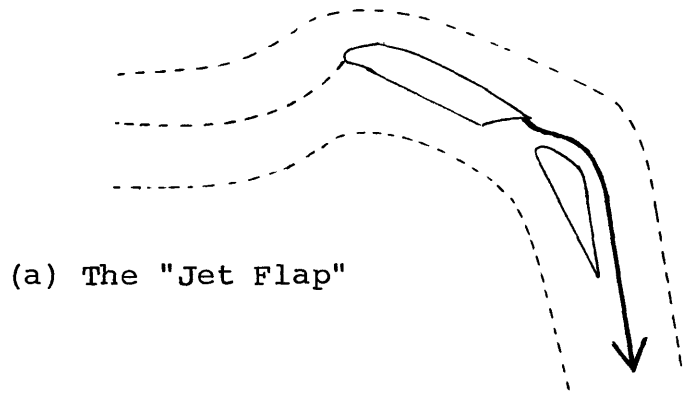
at 60 degree flap deflection, only 65% of the gross engine thrust is effectively deflected (see References 1, 2, 3).

When high lift is not required, as when the aircraft is in cruising flight, the flaps are raised out of the engine exhaust stream, and the aircraft flies like a conventional airplane.

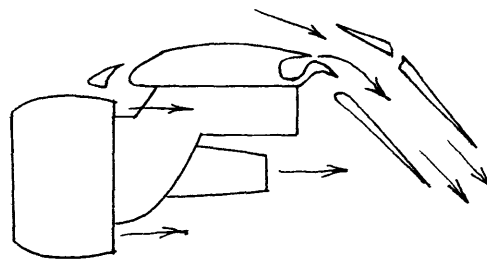
The Augmentor Wing (AW) concept is more complicated, but more efficient (see Figure 1c,d). A pair of trailing edge flaps form an "ejector" slot for a slot nozzle. This nozzle is supplied with high pressure air from the aircraft power-plants through ducts in the wings. This arrangement derives its efficiency from the efficient turning of the nozzle flow and the thrust augmentation of the ejector slot (see References 2, 4, 5).

Augmentor wing systems may be further divided into sub-types. The first is the so-called "two stream" system where the only engine exhaust which does not pass through the duct and flap system is that of the gas generator, or "core". When the thrust of the system is increased beyond that necessary for lift generation, the excess thrust must still be used by the flap system; the angle of the flap, however may be reduced for the same lift. See Figure 1c.

The other system is the "three-stream" type, where a second bypass duct exhausts to the rear along with the core. When thrust beyond that required for lift is added, it may be added



(c) The Two-Stream Augmentor Wing



(d) The Three-Stream Augmentor Wing

FIGURE 1. Powered Lift Concepts.

by this "third stream"; thus, the duct and nozzle system need not be enlarged. See Figure 1d.

In both types, the wing nozzle flow is diverted to "cruise nozzles" when the flaps are retracted and the aircraft is flying as a conventional airplane.

B. POWERED LIFT FLIGHT MECHANICS

In order to estimate the performance of powered lift aircraft in the takeoff and landing configurations, it is necessary to use flight mechanics which account for the deflection of power-plant thrust and the dependence of lift upon thrust.

In general, the most convenient way to handling these flight mechanics is through the use of dimensionless parameters, as in Reference 6. In conventional flight, for example, there is the coefficient of lift

$$C_L = \text{Lift} / \frac{1}{2} \rho v^2 \quad (2.1)$$

In powered lift flight, the C_L is dependent upon the angle of attack (α) and flap deflections (δ_f), but also upon the coefficient of thrust

$$C_T = \text{Thrust} / \frac{1}{2} \rho v^2 \quad (2.2)$$

The gross thrust of the nozzle is used in this calculation.

For the use of the computer program of section III F, the dependence of C_L on C_T , α , and δ_f was taken from wind tunnel data (see References 1 and 4) and reduced to form empirical functions for $C_{L_{\max}}$ (as a function of C_T and δ_f), α (as a function of C_L , C_T , and δ_f), required C_T (as a function of the required $C_{L_{\max}}$, and the δ_f), and required δ_f (as a function of the required $C_{L_{\max}}$, and the C_T), for both the EBF and AW types.

The dependence of C_L upon C_T introduces some unusual features to powered lift flight. The $C_{L_{\max}}$ available increases monotonically with C_T , and so, conversely, decreases with decreasing C_T , for a constant δ_f . Thus, as the speed of flight increases, a constant power-plant gross thrust results in a decreasing C_T (through the increase of the denominator in Equation 2.2 above) and, therefore, a decreasing C_L .

The peculiar result of this is the decoupling of speed and load-factor margins. While a conventional airplane can "pull" a normal load factor of up to 1.44 g's before stalling when flying at a speed 1.2 times stalling speed, the powered-lift airplane at 1.2 times stalling speed without changing thrust will find itself able to "pull" a normal load factor only a little greater than 1.0, the exact value depending on other variables. This

should be born in mind in the discussion of the operational requirements in section II.C.

The calculation of excess specific thrust (thrust available for climb and/or acceleration) becomes a very involved process. For a conventional aircraft, the horizontal net force coefficient is simply the net thrust minus the total drag:

$$C_X = C_{T_{net}} - C_D \quad (2.3)$$

where the total drag coefficient is simply the sum of the profile and parasitic drag plus the induced drag plus the drag of deflected flaps (if any):

$$C_D = C_{D_o} + C_L^2 / eAR\pi + C_{D_{flap}} \quad (2.4)$$

where C_L =coefficient of lift, e="Oswald" efficiency factor, and AR =aspect ratio.

For the powered lift aircraft, however, two factors require different analyses. First, part of the lift is from the deflected thrust, so that the induced drag is reduced:

$$C_{D_i} = (C_L - C_T \eta \sin\theta)^2 / eAR\pi \quad (2.5)$$

where θ is the total thrust deflection ($\alpha + \delta_f$) and η is the deflection efficiency. Secondly, the thrust is deflected, so that the net thrust is no longer the net thrust of the power-plants. Instead, the net thrust becomes

$$C_{T_{net}} = C_T \eta \cos \theta - C_T (V_o / V_{ex}) \quad (2.6)$$

where V_{ex} = exhaust velocity and V_o = flight speed.

For the externally blown flap aircraft, the complete equation becomes, in terms of coefficients (see Figure 2a)

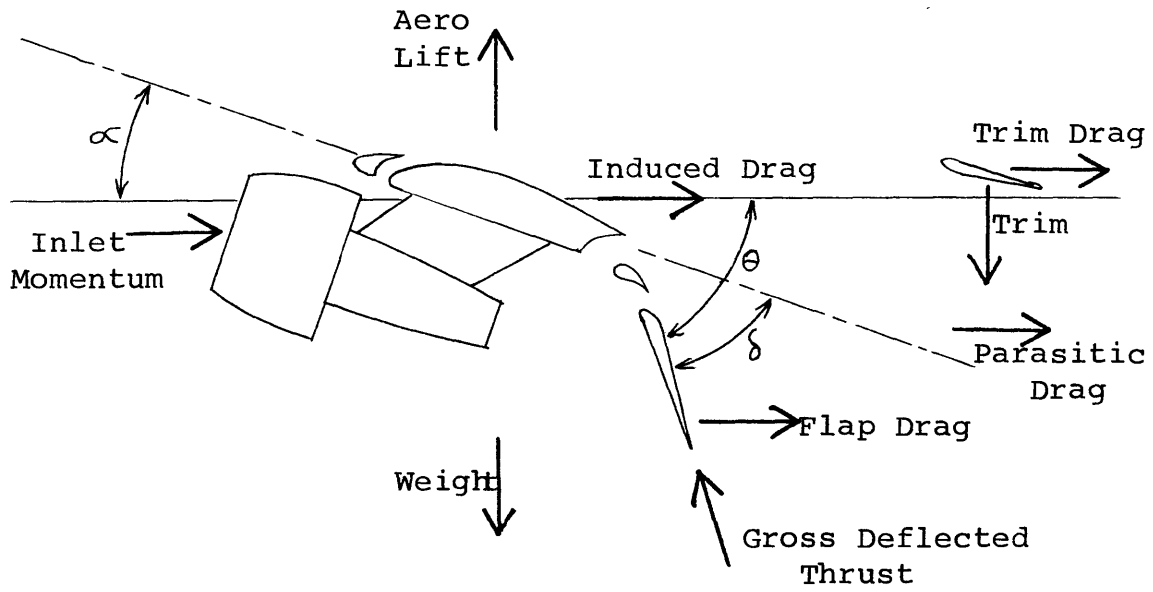
$$C_X = C_T \eta \cos \theta - C_{D_i} - C_{D_o} - C_{D_{trim}} - C_{D_{flap}} - C_T (V_o / V_{ex}) \quad (2.7)$$

where $\eta = f(\delta_f) = 1 - .00583 \cdot \delta_f$

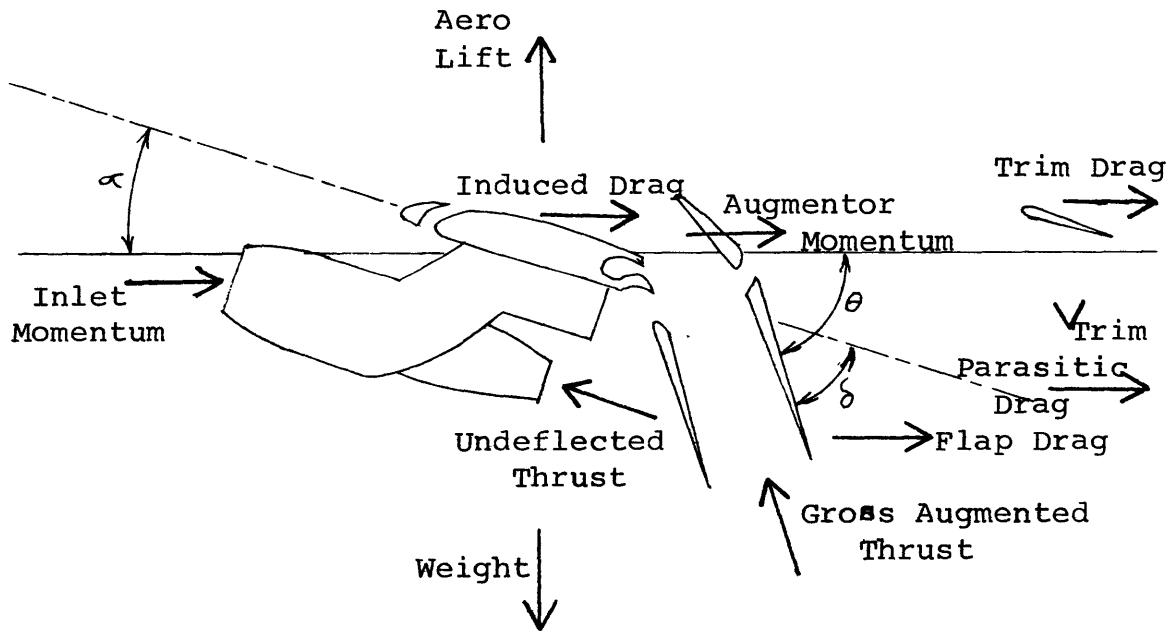
$$C_{D_{trim}} = f(C_L) = .0025 \cdot C_L$$

$$C_{D_{flap}} = f(\delta_f) = .0018 \cdot \delta_f \quad (\text{for the assumptions used in the Appendix})$$

For the augmentor wind aircraft, the coefficient of thrust is related to the isentropic (ideal) thrust of the wing nozzles (mainly for the convenience of wind tunnel experimentors); thus, if the static augmentation ratio (the ratio of the augmentor wing thrust to the nozzle thrust) is ϕ_o , and the efficiency of the nozzle is η_n , the augmentation ratio at forward speed V_o is



(a) EBF



(b) AW

FIGURE 2. Net Force Components.

$$\phi_{V_o} = \left[\phi_o^2 (\eta_n^2 + v_o^2/v_{ex}^2) \right]^{1/2}$$

relative to the ideal thrust of the nozzle.

The momentum drag of the augmentor inlet is then

$$C_{D_{mom}} = (\phi_o^2 - 1) (v_o/v_{ex}) C_T$$

Thus the horizontal force coefficient becomes

$$C_X = C_{T/v_o} \phi \cos \theta + C_{T_e} \cos \alpha - C_{D_i} - C_{D_o} - C_{D_{trim}} \quad (2.10)$$

$$- C_{D_{flap}} - \phi_o^2 C_T (v_o/v_{ex}) - C_{T_e} (v_o/v_e)$$

where C_{T_e} is the engine coefficient of thrust, which is not deflected by the flap. See Figure 2b.

C. COMMERCIAL STOL DESIGN REQUIREMENTS

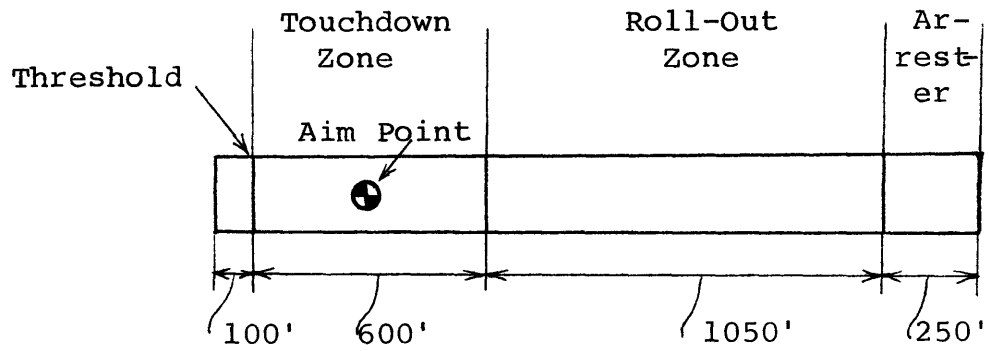
Both the FAA in this country and the ARB in the United Kingdom have issued provisional airworthiness requirements for "Powered Lift" aircraft (see References 7 and 8). The requirements and field length definitions set forth, however, are not yet adequate to define fully the performance required of aircraft

operating into urban or suburban "STOLports".

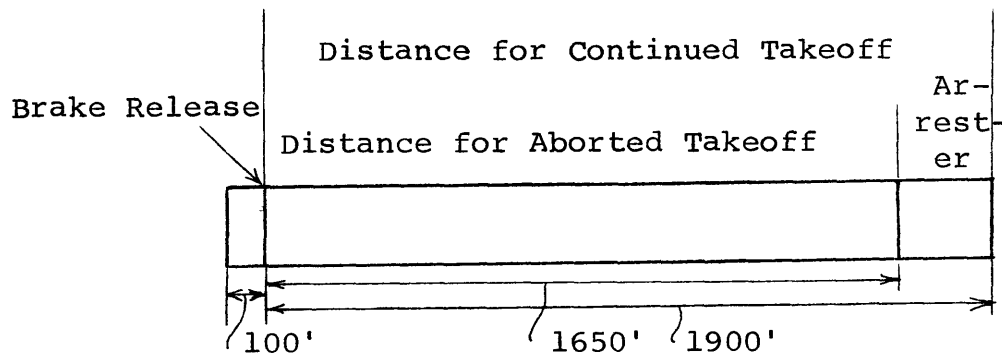
These STOLports may be characterized by extremely restricted space — both in terms of field lengths and air maneuvering space. In reference 9, R. K. Ransone has set forth operational requirements for 2000 foot STOL fields which go far beyond either of the airworthiness standards. We have generally adopted these requirements.

Those operational requirements established in Reference 9 which directly affect those parameters used by the design analysis program (see section II.F) are as follows (see Figure 3):

- a) For takeoff, that the distance required to clear a 35 foot obstacle be not more than 1900 feet, assuming one power-plant failed at a decision speed, V_1 , such that the aborted takeoff (accelerate-stop) distance be not more than 1650 feet (given a two second delay at V_1 for pilot-power-plant response).
- b) The gradient of climb with one power-plant failed must be no less than 7.7% at the second segment climb speed, V_2 , which is taken to equal the lift-off speed in the analysis program.
- c) For landing, the distance available for rollout from touchdown to a complete stop is 1050 feet. Passenger comfort requirements limit the normal operational



LANDING →



TAKEOFF →

FIGURE 3. STOL Field Allowances.

landing deceleration to .33 g's (well below that available for emergency deceleration), giving a normal touchdown speed of 76 knots (130 feet/sec), assuming a two second delay for pilot-airframe-power-plant response.

- d) If a (conservative) assumption is made that there is no deceleration in the flare maneuver, then the approach speed is also 76 knots (89 mph). The flare maneuver, with one power-plant failed, must require a horizontal distance (measured from the glide slope aim point) of not more than 300 feet (while including the adverse effects of "ground-effect").
- e) At this approach speed, on a 7.5 degree glide-slope, the aircraft must be able to clear the STOL field surface after breaking out of a 100 foot ceiling (given a two second delay for pilot-airframe-power-plant response), with one power-plant failed. This requires a normal (perpendicular to the flight path) acceleration of .3 g's; i.e., a load factor capability of 1.3, with one power-plant failed.

Due to the peculiarities of powered-lift flight pointed out in the previous section, this requirement is the critical one for sizing the thrust acting on the flaps. The takeoff requirements (a) and (b)

generally size the total thrust requirement.

- f) Furthermore (in an extension of the requirements of Reference 9), after leveling off, and accelerating over a horizontal distance of 1000 feet, the aircraft must be able to climb at a gradient of 6.7%.
- g) Consistent with the procedures of Reference 6 the combinations of speed, flap settings and available thrust must be such that the flight speed is at least 1.2 times the one power-plant failed stall speed for the configuration; in addition, the flight attitude must be (approximately) 5 degrees below the stall angle of attack.

It should be noted that the above requirements do not include any factors on landing or takeoff distances. In conventional operations (e.g., operations under F.A.R. Part 25), distance factors are used to allow for variations in pilot skill and technique, aircraft-power-plant performance, etc. Instead of distance factors, the STOLports of Reference 9 are equipped with arresting devices capable of stopping an aircraft at landing speed with no damage to the aircraft or injury to its occupants. Their use, which would be alarming to the average passenger, is restricted to unusual and severe emergency incidents, such as multiple engine failure. Single engine failure incidents should be handled without recourse to these devices.

The normal landing deceleration, as mentioned before, is limited by passenger comfort to .33 g's. The deceleration in an abort situation is limited by brake and thrust reverser effectiveness.

In the takeoff calculations performed by the computer program, a coefficient of friction of .40 was assumed for the maximum braking case (abort). Reference 8 gives this as lower limit for the coefficient on wet grooved runways at high speed. As the coefficient increases with decreasing speed, the assumed constant is very conservative, especially since the effects of aerodynamic drag are completely ignored in the abort calculations.

The runway coefficient of friction on takeoff (with no brakes) is assumed to be .025.

Similarly, the thrust reverser effectiveness is assumed to be 40%; i.e., the maximum reverse thrust is 40% of the maximum forward thrust. This is optimistic, except for variable pitch fan engines; however, the inlet drag of the moving engine is ignored. Since this reverse thrust is to be used in the abort case with one engine failed, the full reverse thrust is assumed to operate on only two out of the four engines to avoid asymmetrical forces.

Although Reference 9 requires that the aircraft should be able to land at the approach rate of descent (implying a aircraft-

carrier-type no-flare touchdown), this seems too conservative in terms of landing gear requirements, and too liberal in terms of passenger comfort. Therefore, the landing gear is sized so that it can withstand loads 1.66 times that of conventional transport aircraft.

The computer routines which compute landing and takeoff performance compute the flap angles required for all flight conditions, as set by stall margins, etc. The maximum setting for the EBF aircraft is 60 degrees, and the maximum for the augmentor wing is 70 degrees.

In accordance with the practices of Reference 6, during the takeoff roll, the flap angle is assumed to be zero, to allow all of the thrust to accelerate the aircraft. This means that the flaps must be reasonably fast acting, with deflection rates of about 15 degrees/sec.

The effects of the ground plane on the landing flare maneuver are taken from Reference 10. All designs are assumed to be high wing configurations.

The requirements for STOL operation given in this section must be met at a hot day temperature of 30° C, 15° C above the standard day ground temperature. Cruise calculations, however, are made at standard day conditions.

D. POWER PLANT PERFORMANCE

The chief parameters which characterize each aircraft design are those of the power plants. The selection of power-plant type has profound effects on the gross weight, cost, and noise of each design.

The most basic parameter that characterizes each power-plant is the velocity of the exhaust. The EBF power-plants are assumed to have only one exhaust velocity (the average of the bypass and core exhaust velocities), while the AW engines have two (one for the flap nozzle, and one for the engine nozzle(s)).

The thrust-to-weight ratios and static specific fuel consumptions are shown in Figure 4 for the exhaust velocities used in this report. These values assume an advanced gas generator (engine "core") with compressor ratios of about 28 and turbine inlet temperatures of 2800°R , as will be available in the next generation of civil engines, available in the late 70's (see Reference 11).

E. OTHER DESIGN ASSUMPTIONS

No arbitrary lower limit was set on wing-loading, as is sometimes thought necessary for ride comfort considerations. References 12 and 13 have demonstrated the feasibility

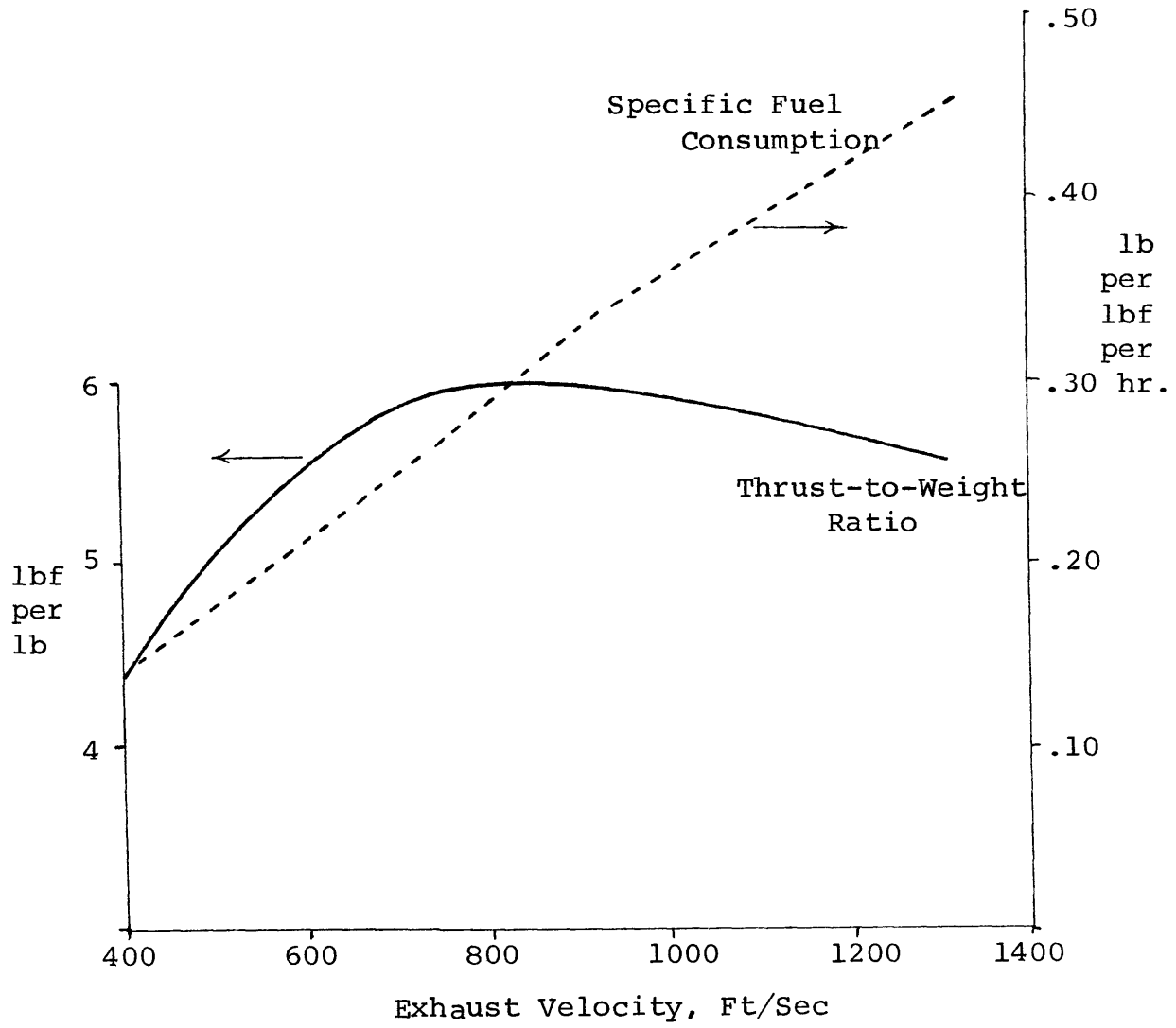


FIGURE 4. Power-Plant Performance.

of ride smoothing by the aircraft control surfaces and fast acting flaps. Thus, wing loadings as low as 60 lb/ft^2 were investigated.

There is no allowance made in the design program for asymmetry in engine-out situations. The AW aircraft have cross ducts in the wings to neutralize part of the effects, but the EBF aircraft must rely completely on differential flap and aileron control to counteract the resulting rolling moments.

With this in mind, there is an allowance in the design process of both types for the weight of equipment to "blow" all of the control surfaces.

No limit is placed on the ground angle on takeoff or landing, but the stall margin requirements coincidentally limit this to about 15 degrees.

All of the aircraft in this study have similar wing geometry. The rather low aspect ratio of 6.5 was assumed, as it was required for the AW aircraft (due to duct volume requirements), and EBF aircraft were not sensitive to aspect ratio changes, due to the low wing loadings of interest, and the short range of the aircraft.

The thickness-chord ratio (measured perpendicular to the wing structural axis) was 17% and a supercritical wing profile was assumed. The taper ratio was constant at 0.5.

The wing sweep angle for the AW aircraft was the minimum which

resulted in a critical Mach number for the wing greater than the design cruise speed (0° - 20°). For the EBF aircraft, the sweep angle was 25° , which is required to spread the power-plant exhaust over a larger area of flap.

Cruise speed was not an input to the design process, the cruise speed being that which resulted from the thrust-loading necessary to meet the STOL requirement. An upper limit of Mach .8 was imposed, however, to prevent excessive fuel consumption by those designs which were capable of very high speeds.

As most of the designs were limited in cruise speed, the operating cruise altitude was set at 25,000 feet, the optimum altitude for maximum speed for most designs.

The design range for all of the designs was 500 statute miles. STOL aircraft will be used for short-haul operations, implying a average stage length of less than 300 miles. Such a low design range reduces the flexibility of the aircraft for ferrying and semi-conventional operations, but the sizes of these designs are sensitive to the required range, so it is of great advantage to keep the range as short as practical.

A passenger weight factor (the assumed average weight of one passenger and his baggage) was 200 pounds for these designs. This is a bit generous for short-haul "commuter" type operations, but no allowance was made for cargo or mail carriage.

The weight of a standard airline interior was allowed for, although no galleys were assumed, and only a very small weight allowance was made for inflight services.

Most of the aircraft in this report carried 80 passengers. This size was chosen for two reasons. First, the initial sizes of all the short-range jets (the DC-9-10, the BAC-111, the F-28, and the Caravelle) were all in that size range. Second, the success of a short-haul transportation system in competition with other modes depends very strongly on frequency of service. For high frequency, it is necessary to have small aircraft.

Two other sizes were studied less broadly, however, to investigate the effects on cost and noise of other sizes; 50 seat and 110 seat aircraft were chosen, mainly for compatibility with another FTL study of the cost of noise reduction in short haul helicopters, Reference 14.

The fuselage dimensions chosen for the three sizes were as follows: For the 50 seat aircraft, the seating arrangement was four abreast, resulting in a fuselage diameter of 9.5 feet and a length was 68 feet. For the 80 seat aircraft, the seating was also four abreast, and the length was 85 feet. For the 110 seat aircraft, the seating was six abreast with two aisles, resulting in a fuselage diameter of 14 feet, and a length of 92 feet.

All aircraft carry a flight crew of two, and either two (for

the 50 and 80 seat aircraft) or three (for the 110 seat aircraft) cabin attendants.

The reserves carried by the aircraft are sufficient to fly at the cruise speed for best milage (at the design cruise altitude) for one-half hour, plus that required to abandon a missed approach and fly to an alternate airfield 110 statute miles away.

These reserves are only half of those carried in conventional airline operations, but should be adequate considering the short-haul nature of the operations, and the ease with which STOL aircraft could operate into conventional airports in the worst of weather.

F. THE AIRCRAFT DESIGN COMPUTER PROGRAM

As mentioned in previous sections, the actual synthesis and evaluation of aircraft designs was done by a computer program. This program enabled a large number of designs to be studied.

The program is described in detail in the Appendix. Its general design features are described here, and the evaluation features in Chapter III.

The design process used in the program is that of the preliminary design engineer: without detailed analysis of the design, find the required weight of each component, and solve for the gross weight.

The inputs to the program include the required payload and range, the wing aerodynamic and structural geometry, the basic engine parameters, the required reserves, the required takeoff and landing performance, and the desired wing loading.

The first step is to solve for the necessary engine size (in terms of the thrust-to-weight ratio) to satisfy the takeoff and landing requirements. A rough estimate of the zero-lift drag coefficient (C_{D_0}) of the aircraft is made, and all of the other necessary coefficients are found from the known wing and engine parameters. An initial value for the thrust-to-weight ratio is found from the missed approach pull-up requirement. The missed approach maneuver is tested to see if the gradient of climb at its completion is adequate. If it is not, the thrust-to-weight ratio is increased in steps of .01 until the requirement is met. The same process is repeated for the other requirements.

The resulting thrust-to-weight ratio now defines the engine size in terms of the unknown gross weight. Starting with an estimate of the gross weight, the routine uses the fuselage, wing and engine parameters to find a better estimate of the aircraft zero-lift drag coefficient. Conventional aerodynamics are then used to calculate the climb, cruise, and descent performance of the aircraft, in order to find the fuel weight required for completion of the design mission with the proper reserves.

The program then uses formulae taken from regression analysis studies to find the weights of the various components of the airframe. These formulae (taken from Reference 15) are consistent with conventional all-metal design and construction techniques of the late sixties. Due to the uncertainty of the availability and costs of composite construction in the time frame under consideration (the late seventies), only conventional techniques could be considered.

The power-plant weights were taken from the previously found thrust-to-weight ratios and the specific weights shown in Figure 4.

The airframe, engine, fuel, payload, and other miscellaneous weights are then summed to find a total weight. This is compared to the old estimate of gross weight that was used in all of the computations. If these two weights agree to within ten pounds, the design is considered solved, and the design features are output in detail.

If the two weights do not agree, a new gross weight is solved for by the "weight fraction" method. In this method, the weight of each component whose size depends on the gross weight (wings, engines, fuel, etc.) is assumed to be a fixed fraction of the gross weight. All other weights are assumed to be fixed. A new gross weight is found by setting the fraction of the gross weight

remaining for the fixed weights equal to the total weight of these fixed weights.

This new estimate is used to repeat the process, starting with the detailed estimate of C_{D_0} , until successive iterations converge to within ten pounds.

III. EVALUATION OF AIRCRAFT

Two figures of merit were chosen to compare the results of the design activities. One was a cost measure; the direct operating cost at 500 miles. The other was a noise measure; the footprint area within the 85 PNdB contour for a takeoff and landing.

A. DIRECT OPERATING COST

The results of the design process of section II.F are used to estimate the direct operating cost (DOC) of the design. A set of stage lengths (along with a cruise altitude and speed for each one) is input, and the aircraft drag and thrust performance used to calculate the block time and block fuel consumption for each one.

Next, a set of cost parameters is input and used to estimate the value of the aircraft. The power-plants for both types of aircraft were priced at \$25 per pound of thrust. The airframe (empty weight minus engine weight) was priced at \$70 per pound for the EBF aircraft, and \$80 per pound for the AW aircraft (this is to reflect the added complexity of the duct and flap equipment). Assumed utilization is 3000 hours per year, with a useful life of twelve years.

These values are put directly into the ATA (1967) Standard Method for DOCs (Reference 16), and used, along with the block data, to compute the direct operating costs (in 1967 dollars) for each stage length.

B. NOISE EVALUATION

It has been shown (Reference 17) that for uniform population density, the noise footprint area is proportional to a measure for total community annoyance. Therefore, the footprint area within the 85 PNdB contour has been selected as a measure of the noise impact of these aircraft.

The value of 85 PNdB was chosen for two reasons. First, it was neither so high a value that the approximations involved in the contour calculations would break down, nor so low that the contour areas would be strongly dependent on the flight profile followed on takeoff or landing. Second, it approximates the exterior noise level which most people will tolerate repeatedly when they are inside of any reasonable building (see Reference 18). This contour can be assumed to encompass all of the land that will be effected by noise from a STOLport operation. This area must be either utilized by the STOLport, owned by the STOLport operators, zoned for activities not influenced by aircraft noise, or covered by easements from landowners. The cost and

difficulty of STOLport location will be proportional to the value of this figure of merit.

In order to compute the 85 PNdB contours used as the noise figure of merit, it is of course necessary to know something of the noise generation of powered-lift STOL aircraft.

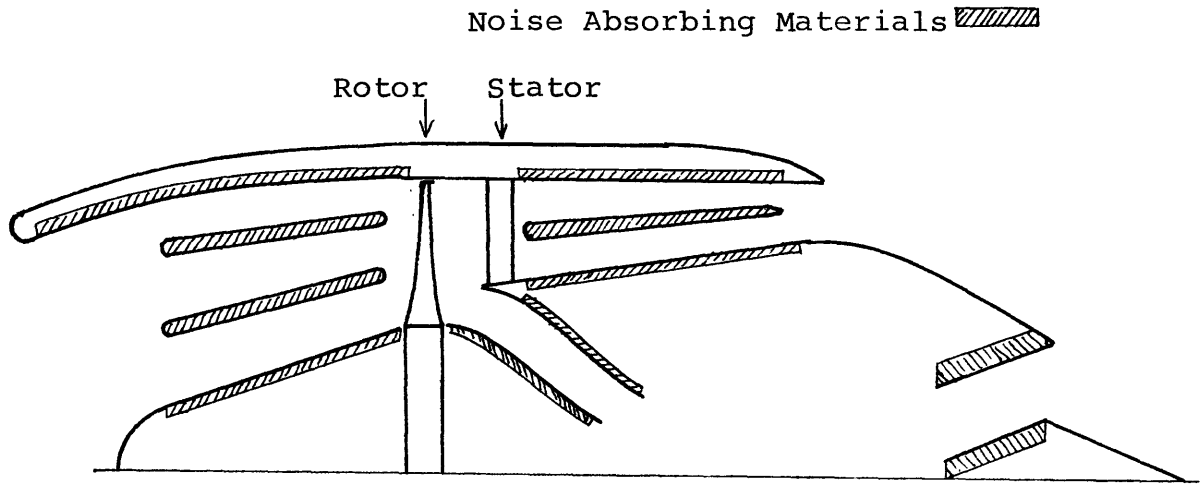
There are three sources of noise from powered-lift aircraft operating in takeoff or landing configurations: the noise generated by the fan, compressor and turbines ("fan" noise); the noise generated by the exhaust ("jet" noise); and the noise generated by the operation of the powered-lift system ("flap" noise).

Fan noise is generated by the fluctuating pressures on rotor and stator blades as the machinery rotates. This noise is emitted out of the inlet and exhaust nozzle of the engine. The level depends on both the basic parameters and the detailed design of the engine.

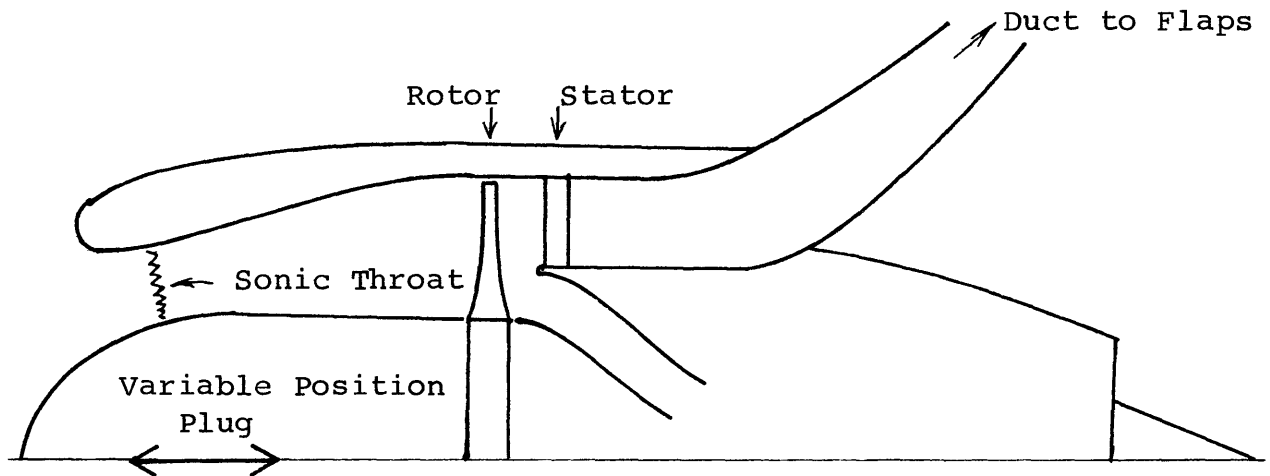
Fan noise may be reduced by three means. The first is to select the details of design, such as fan rotor tip speed and rotor-stator spacing, to minimize the noise, with only a small performance penalty. The second is to choose the basic parameters of the power-plant for lower noise levels; this has very profound effects on the design of both the engine and the airplane that carries it. The third method is to absorb the noise which is generated before it leaves the engine nacelle.

There are two ways of absorbing fan noise (see Figures 5a and 5b). The simplest is to line the inlet and ducts with porous materials which damp out the sound waves which impinge upon them. The second method is called "sonic choking", which involves constricting the inlet, forcing the velocity of the entering air to approach the speed of sound. The sound waves propagating out through the inlet are slowed as they pass through the sonic zone, and are absorbed by the flow nonuniformities associated with the sonic flow (the mechanism of absorption is not completely understood). This method only works for noise propagating out of the inlet, so it is useful only for the two-stream type of Augmentor Wing aircraft, where all of the bypass flow goes into ducts leading to the flap nozzles. The fan noise propagating down these ducts can be easily absorbed with no performance penalty. The sonic inlet has performance penalties only while it is in use; during cruise, or in an engine-out emergency, the inlet can be unstricted, allowing the engine to develop full thrust.

Three fan noise levels were used for each engine type (from References 11 and 19) as shown in Figure 6. The loudest is for a moderate amount of suppression — absorptive lining of the inlet and exhaust duct walls, with no engine performance penalties, but with a 20% nacelle weight penalty. The middle curve is for the maximum amount of absorptive suppression, with linings on the



(a) Suppression By Acoustic Linings



(b) Suppression By Inlet Choking (On an Augmentor Wing Engine)

FIGURE 5. Fan Noise Suppression.

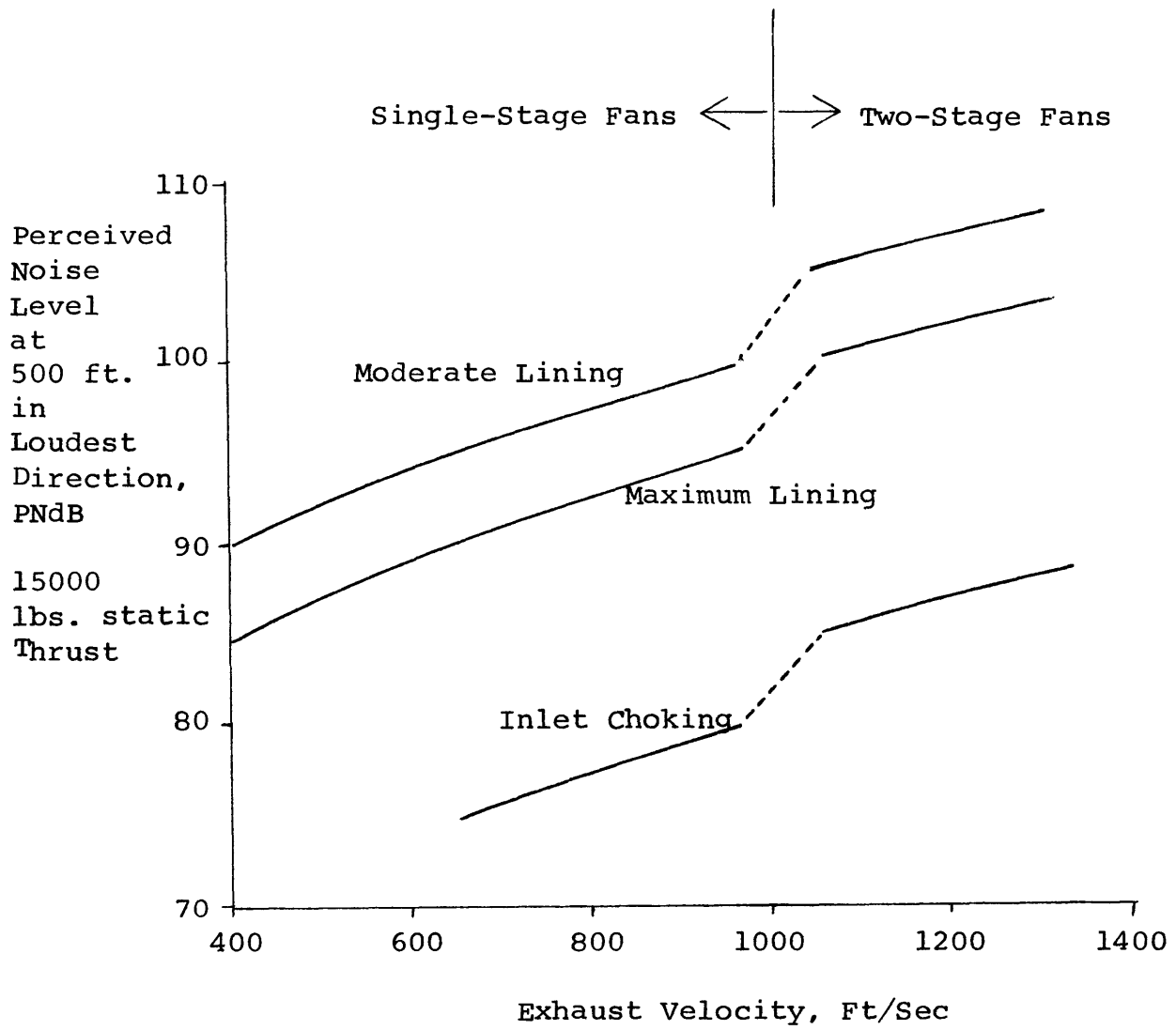


FIGURE 6. Reference Fan Noise Levels

walls and on splitter rings inserted in the inlet and ducts. This can involve substantial performance losses, and adds 60% to the weight and 10% to the drag on the engine nacelles. The quietest curve is that for sonic choking. It involves severe performance losses when in operation, and also adds 60% to the weight and 10% to the drag of the nacelles.

The performance penalties of the maximum lining suppression and sonic choking (from References 11 and 20) are shown in Figure 7. As the suppression devices cause constant pressure losses, the penalties increase with decreasing fan pressure ratio. (The penalties for the sonic choking are probably too severe in the light of results published after the studies in this report were completed.)

Jet noise is caused by the turbulent mixing of the exhaust flow with the ambient air. This mixing is completely external to the engine, and cannot be absorbed. Reducing the exhaust velocity by choice of engine parameters is the most effective way to reduce the noise, as slight reductions in the exhaust velocity can have marked effects of the noise generation. This, of course, has basic effects on the engine and aircraft design.

The only other way to reduce the noise is to use a "mixer" nozzle (see Figure 8), which improves the mixing of the exhaust, reducing the noise levels. This sort of suppression has little

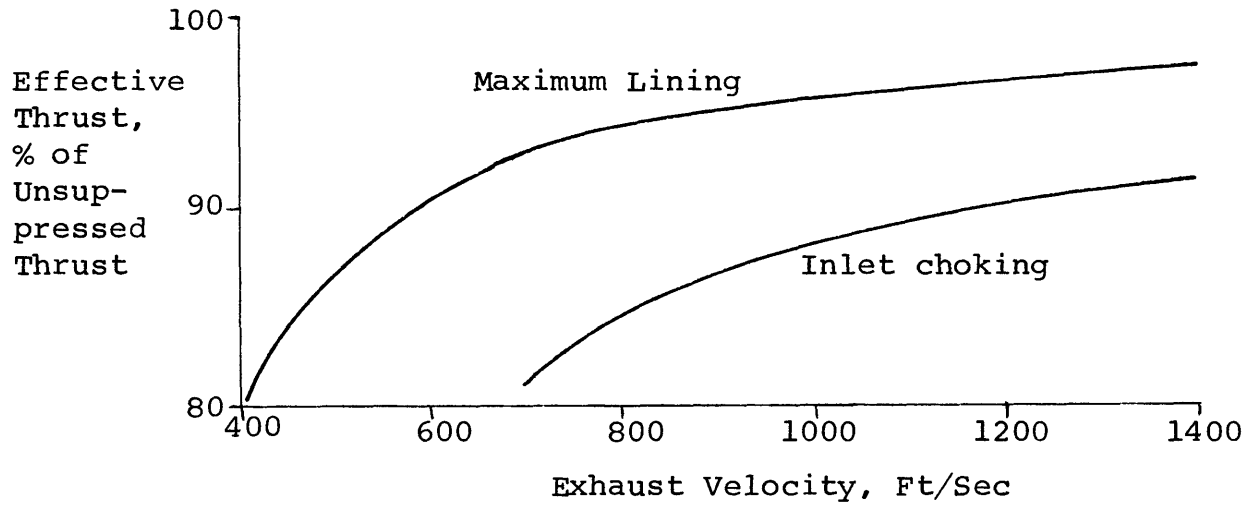


FIGURE 7. Fan Suppression Penalties.

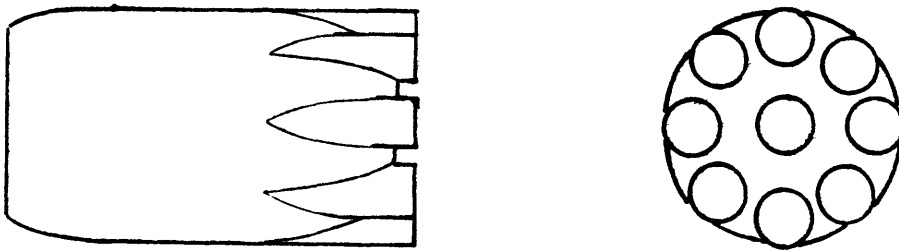


FIGURE 8. Mixer Nozzle

effect on the noise levels of engines with low exhaust velocities, as are used for the aircraft studied in this report. The jet noise levels for the velocities of interest (from References 11 and 21) are shown in Figure 9.

Flap noise must be discussed for the two types of aircraft separately, for they have completely different mechanisms of generation.

In the EBF system, the exhaust of the engines blows directly on the flaps. The turbulence caused by the exhaust mixing "scrubs" across the flaps, causing pressure fluctuations, which, in the presence of a solid boundary, become efficient noise sources. The noise of the exhaust blowing on the flap is louder than the noise of the exhaust by itself.

As with jet noise, the external nature of the noise does not allow the use of absorbing devices. Similarly, reductions in the blowing velocity allow marked reductions in the noise level. The use of a mixer nozzle on the engine, which results in lower peak velocities on the flap (reducing the intensity of the turbulence) can help reduce the noise of the flap scrubbing slightly. The noise levels for EBF flaps (from References 2,3, and 21) are shown in Figure 10. The upper curve is for plain nozzles, and the lower is for mixer nozzles. It can be seen that the difference is slight.

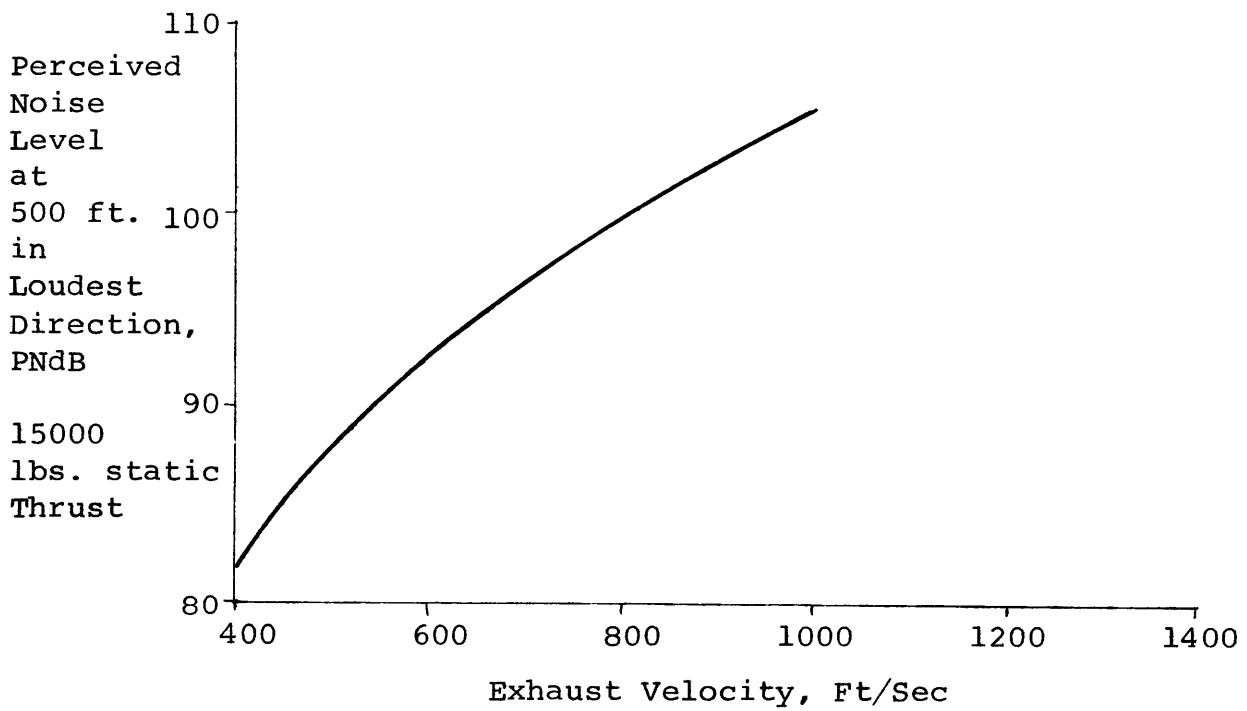


FIGURE 9. Static Reference Jet Noise Levels

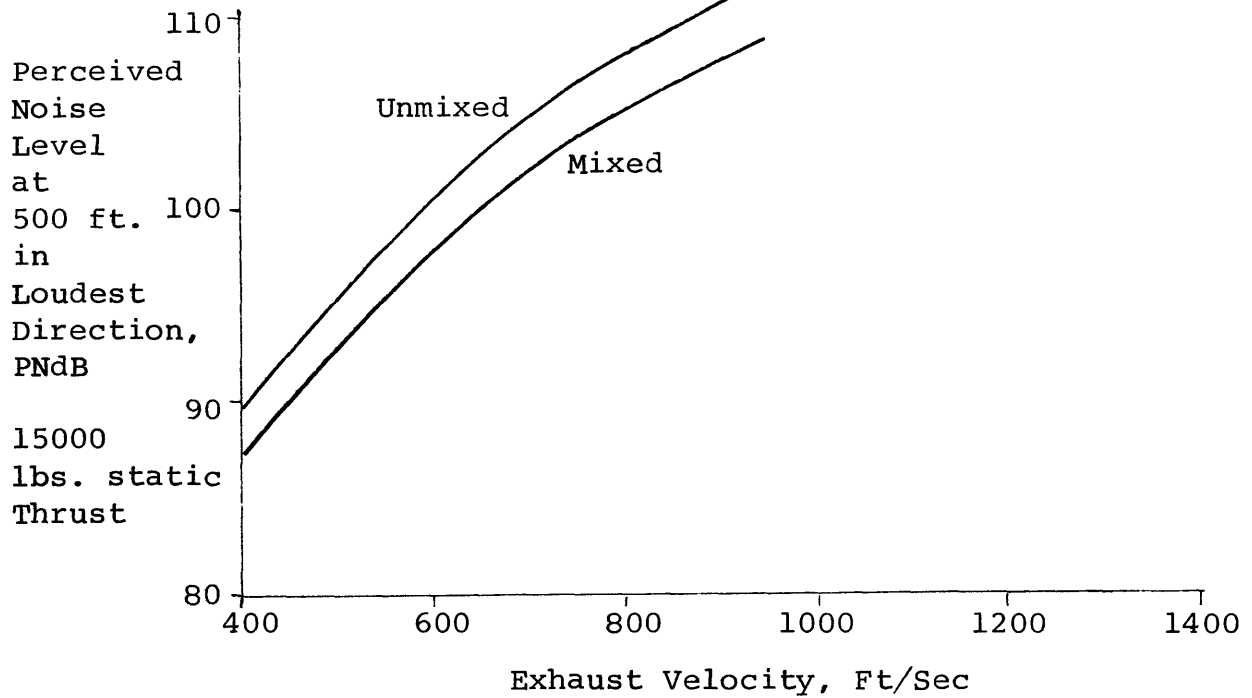


FIGURE 10. Reference EBF Flap Noise Levels

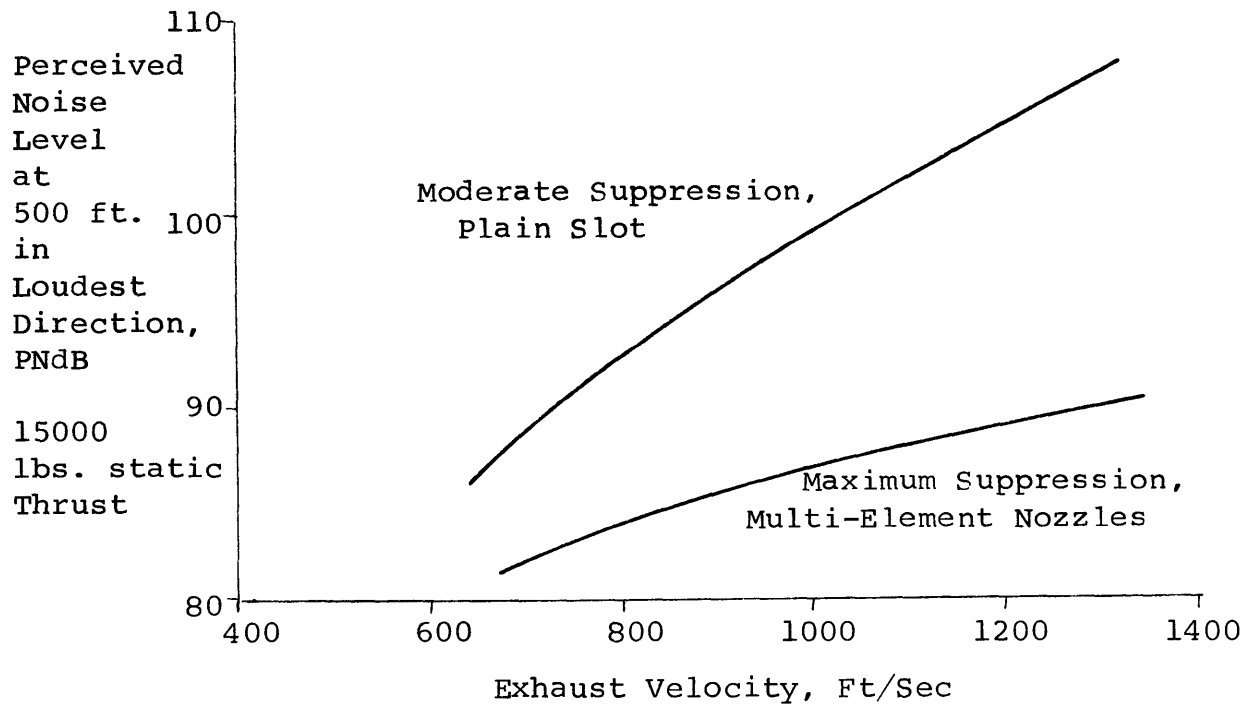


FIGURE 11. Reference AW Flap Noise Levels

The nozzle of an AW system exhausts between two flaps without impinging directly upon them. Thus, the noise source for a properly design system is a jet mixing noise identical to that described earlier.

The fact that this jet is in the form of a slot instead of a circular orifice increases the mixing efficiency of the system, resulting in a lower noise level relative to an ordinary jet of the same velocity and thrust. The advantage is further enhanced by the ejector slot, which constricts the flow around the jet, raising its speed, and lowering the relative mixing velocity, resulting in a still lower noise level.

The noise level can be lowered even further by two techniques. The first is the changing of the narrow slot into an array of multiple slots, further increasing the mixing efficiency, and lowering the noise level in a way identical to the mixer nozzle on a circular jet. The second is to line the areas on the inner faces of the ejector flaps with absorptive material, which absorb the noise internally.

The noise of an AW system is also amenable to reduction by lowering the nozzle exhaust velocity. However, since the air for the nozzles must be carried by ducts inside the wing, and lower exhaust velocities require higher mass flows for the same thrust, a lower limit is placed on the nozzle velocity by the

cross sectional area of the wing profile.

Under certain conditions the noise of an AW system may be increased by such mechanisms as supersonic nozzle "screech" and flap "scrubbing". With proper design, however, these problems can be avoided.

Noise levels for the AW systems (from References 2,5,11 and 21) are shown in Figure 11. The upper curve is for plain slot nozzles with a moderate amount of suppression material applied to the interior surfaces of the flaps. The lower curve is for the maximum amount of suppression, with mixer nozzles, etc. The increased mixing efficiency of the maximum suppression flaps results in increased thrust augmentation, so there is no performance penalty. As with mixer nozzles for circular jets, the suppression is less effective for lower nozzle velocities.

The noise levels shown in Figures 6,9,10, and 11 were used as reference noise levels for the power-plants used in the noise-contour process. After the design process has found the required thrust levels, the noise levels are scaled up or down to account for the difference between the reference 15000 lb-thrust engines and the actual design engines.

The noise levels of the fan and jet vary with flight condition and thrust level in ways which are easily approximated. The noise levels of the flap systems are considered to remain constant with

flight condition, changing only with thrust level and flap setting (see References 5 and 22).

Each of the noise sources has its own directivity pattern; i.e., the noise sources are not isotropic, but put out different noise levels in different directions. The program uses empirical curves adapted from data taken from References 2,3,11,19, and 21.

The thrust and drag performance is used to calculate take-off and landing trajectories for each design, with time histories of thrust, aircraft angle, and noise generation.

When the takeoff path is computed for the noise evaluation, the aircraft climbs at maximum power. No "noise-abatement" power reductions are used, as they are ineffective for the power-plants under study, and would require awkward changes in flaps, etc. On landing, the aircraft follows a 7.5° glide slope down to the threshold.

These paths and time histories are then used, along with the directivity assumptions, to compute time histories of perceived noise level at each point on a ground level grid below and along side of the flight path (see Figure 12a).

The noise level from the aircraft drops off with distance, not only due to the spherical attenuation (the "inverse square" law), but also due to the absorption of sound energy by the atmosphere. The absorption is a function of the temperature, the

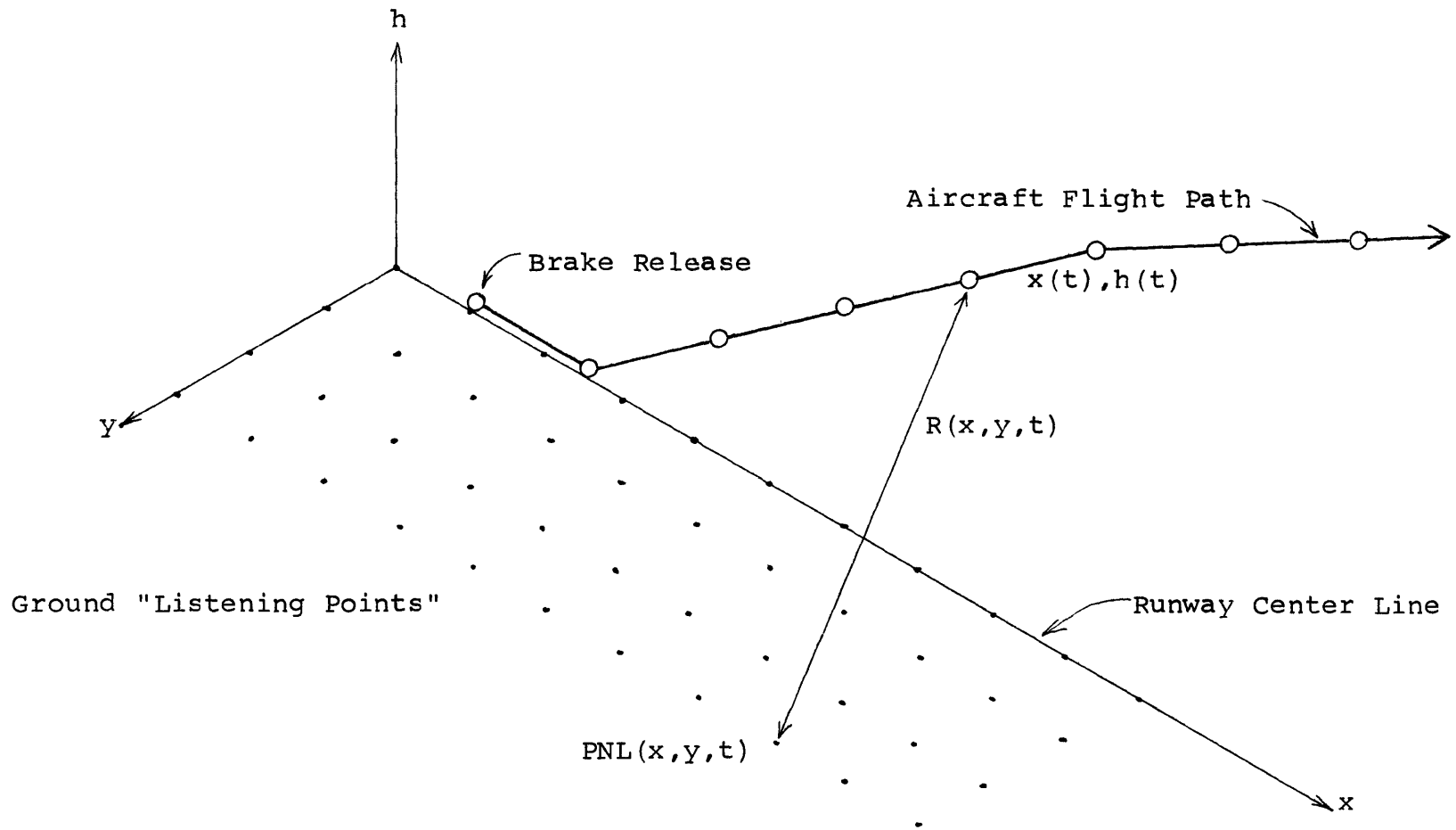


FIGURE 12a. Noise Evaluation Computations (Takeoff)

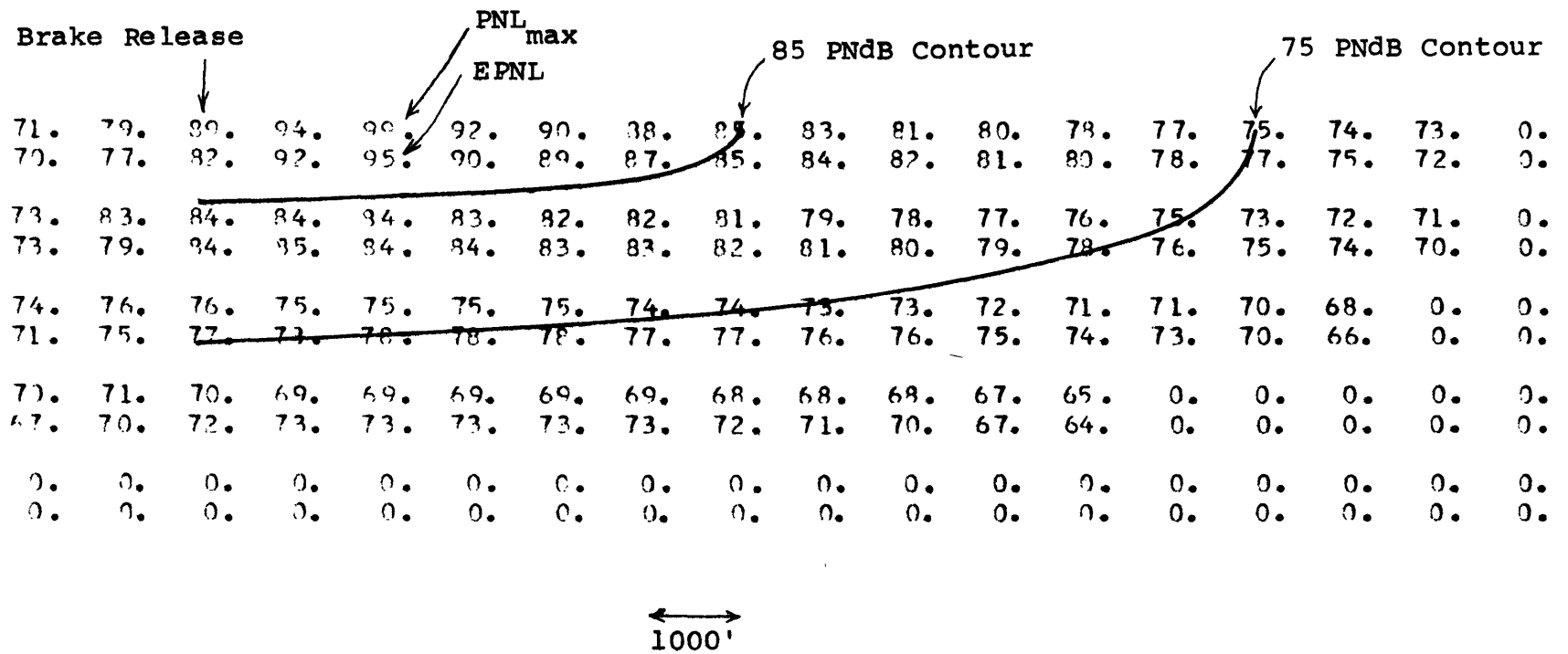


FIGURE 12b. Sample Computation Output (Takeoff).

humidity, and the wave-length of the sound. In this report, the conservative assumption of a dry standard day was used. The attenuations were 3 dB per 1000 feet for the fan noise (due to its high frequency content), 1 dB for the jet, and 1.5 for the flaps.

The time histories of perceived noise are then used to find a Maximum Perceived Noise Level, and an Effective Perceived Noise Level for each ground point, by the methods of Reference 23. The results are output in a grid format (see Figure 12b), from which the contours of PNL_{\max} and EPNL can be estimated.

IV. RESULTS AND DISCUSSION

The use of a computer program to design and evaluate the different types of aircraft allowed a large number of different wing loadings and engine types to be investigated for the Externally Blown Flap, the two-stream Augmentor Wing, and the three-stream Augmentor Wing types of STOL transports.

A. PRELIMINARY RESULTS

Some preliminary results covered wing loadings from 60 lb/ft² to 100 lb/ft². A clear pattern of cost and noise emerged for all types. As shown in Figure 13, both cost and noise were reduced by reduction of the wing loading. For the AW types, this trend was reduced or reversed below 80 lb/ft². The trend continued down to below 60 lb/ft² with the EBF types.

The final results for the AW types, therefore, covered only 70 and 80 lb/ft². The final results for the EBF types covered 60, 70, and 80 lb/ft². It was considered outside the scope of this work to examine wing loadings below 60 lb/ft², as powered lift becomes unnecessary at lower loadings.

The results shown in Figure 13 are at variance with the results in Reference 6. That study found that the optimum EBF aircraft (AW types were not studied) had a wing loading of about

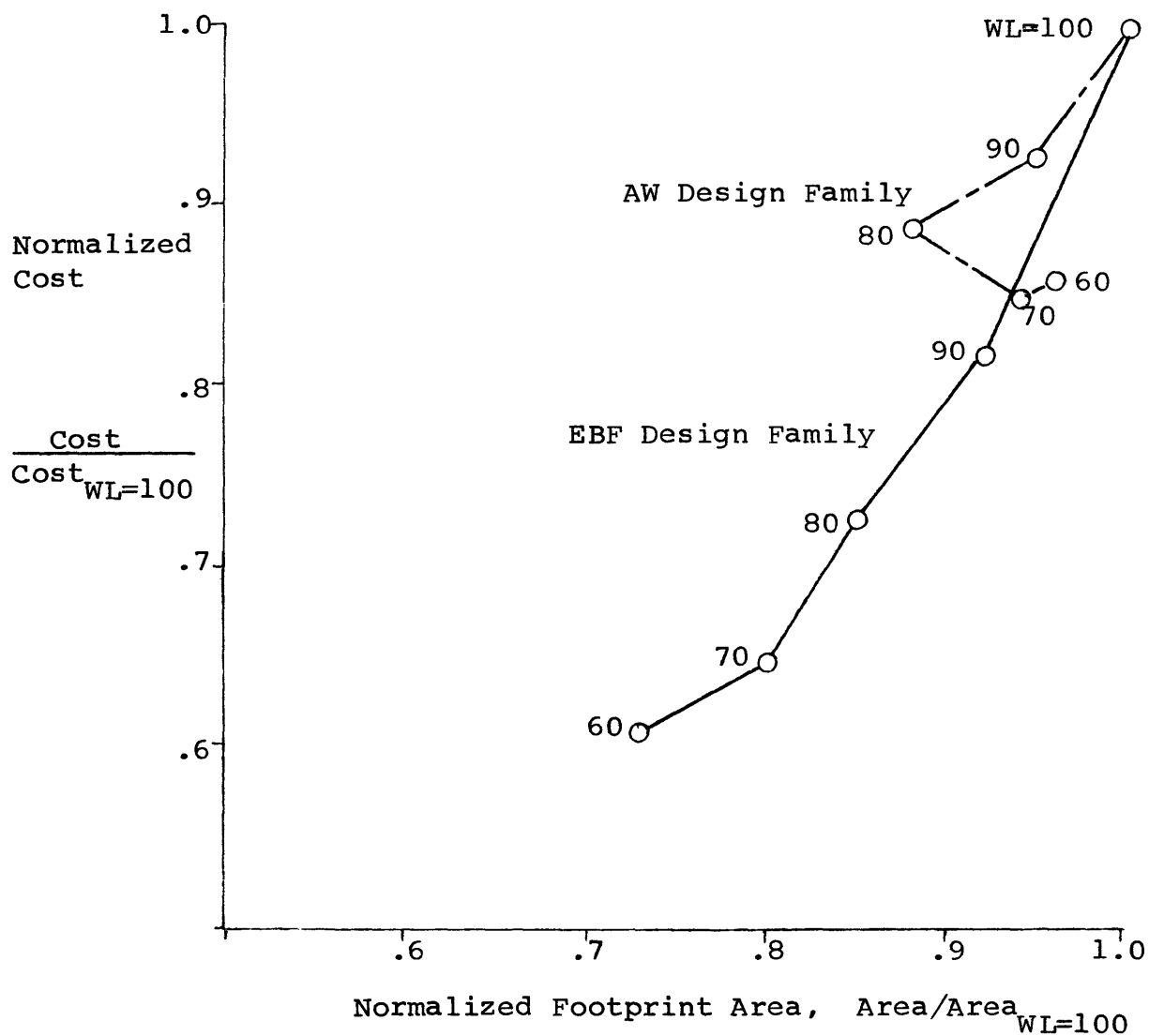


FIGURE 13. Typical Cost and Noise Trends With Wing Loading.

80 lb/ft², using military design rules (with no noise constraints). The rules use in the present study resulted in decreasing costs for aircraft with decreasing wing loadings, down to, and presumably beyond, 60 lb/ft². The use of sophisticated high-lift devices permits non-powered-lift STOL operations (in accordance with the rules of section II.C) with wing loadings as high as 55 lb/ft². The conclusion must therefore be made that low wing loading STOL aircraft would be both cheaper and less noisy (having no flap-blowing noise) than EBF aircraft. This assumes, of course, that passenger comfort can be maintained at low wing loadings by means of ride-smoothing devices.

B. FINAL RESULTS

The final results for the three different types of aircraft are shown in Figures 14,15, and 16. The horizontal scale on each plot is the area of the 85 PNdB contour for a takeoff and landing operation in acres (note that the scale is logarithmic). The vertical scale is the seat-trip cost at 500 miles in 1967 dollars (note that the scale is not logarithmic).

The lines labelled "95" in each plot are an attempt to compare the noise figures to the single noise rating number that is most often seen — "95 PNdB at 500 feet". If an isotropic source which gives a noise level of 95 PNdB at 500 feet starts at the

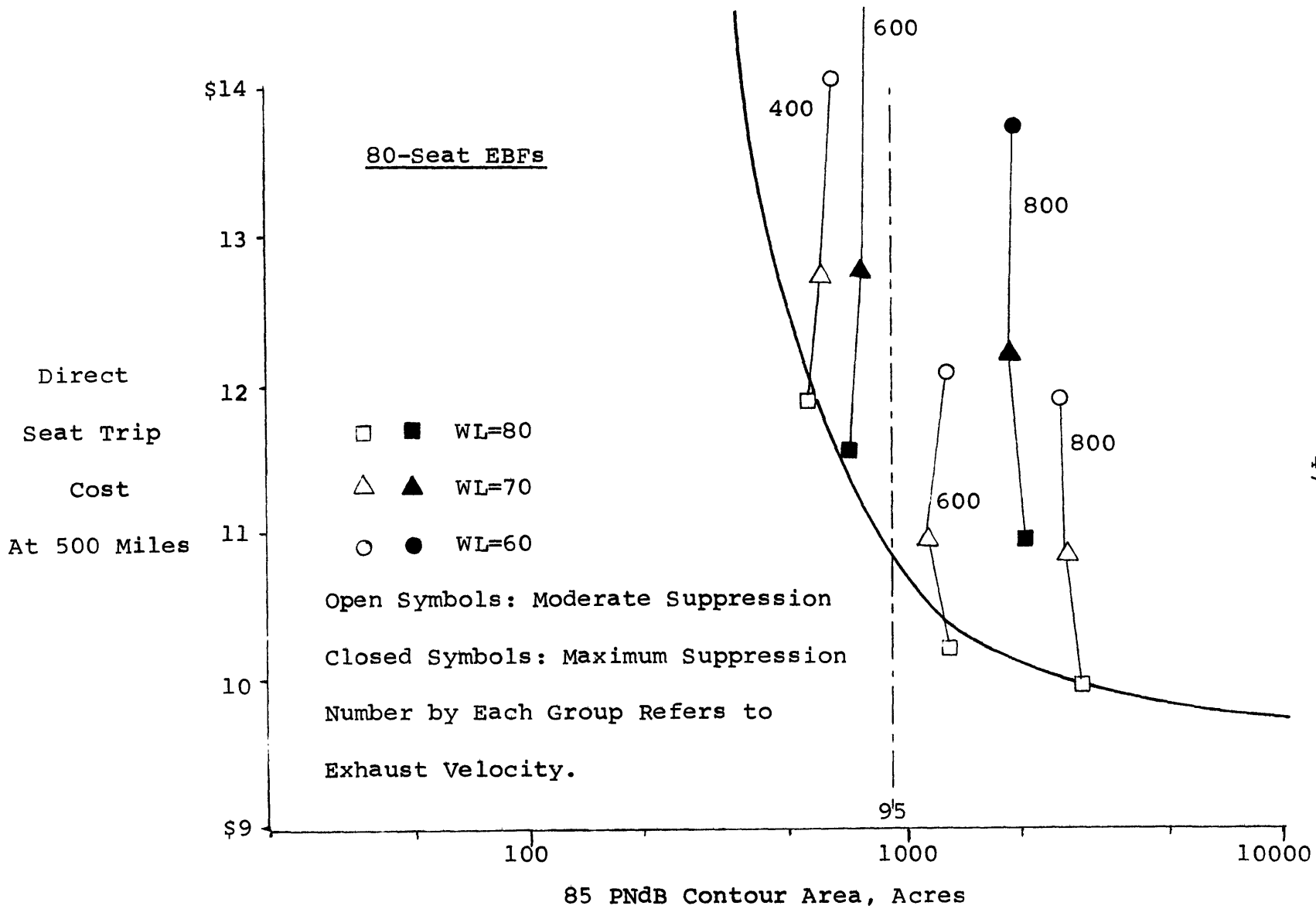


FIGURE 14. Cost and Noise of EBF STOL Designs (80 Seats).

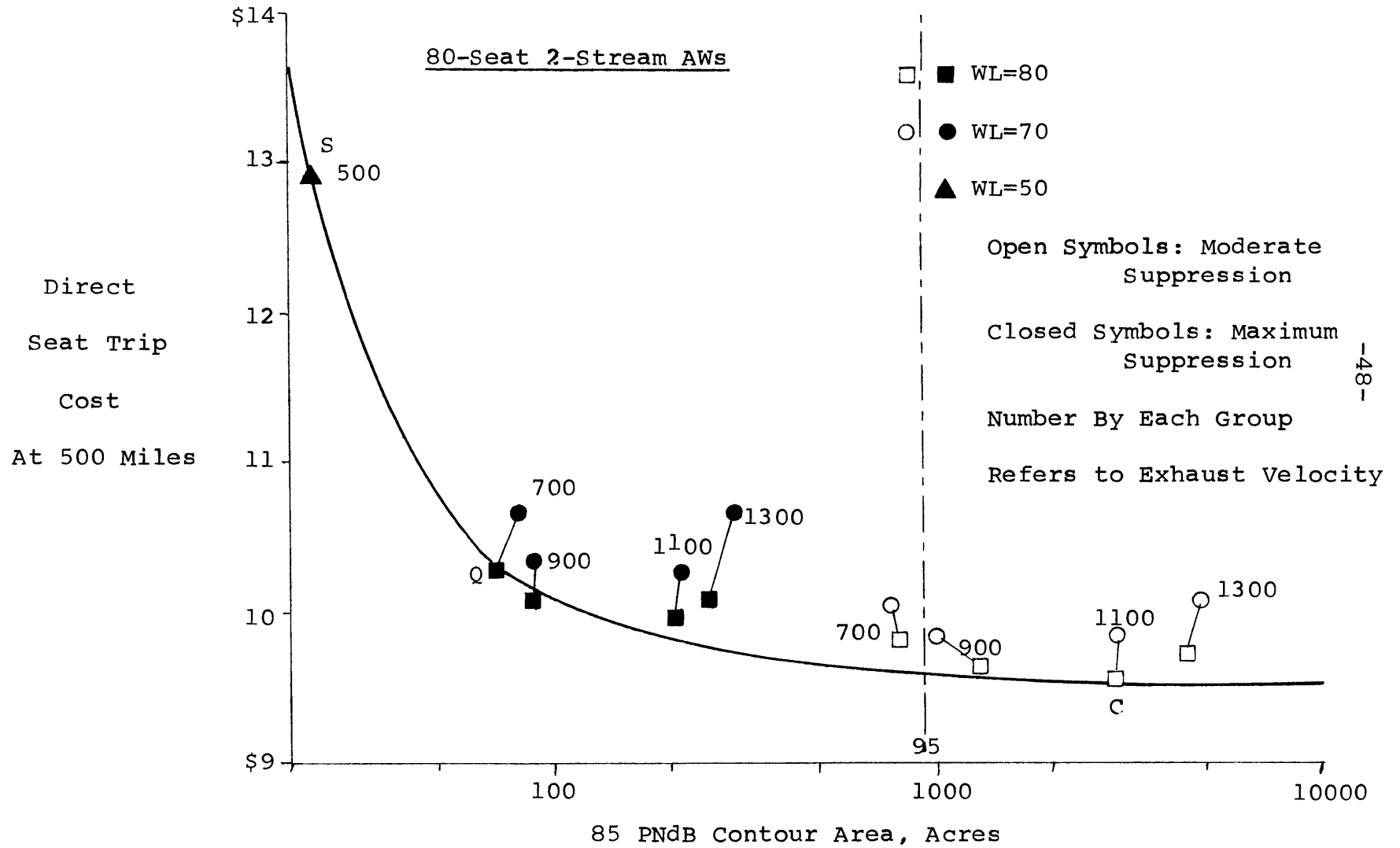


FIGURE 15. Cost and Noise of Two-Stream AW Designs (80 Seats).

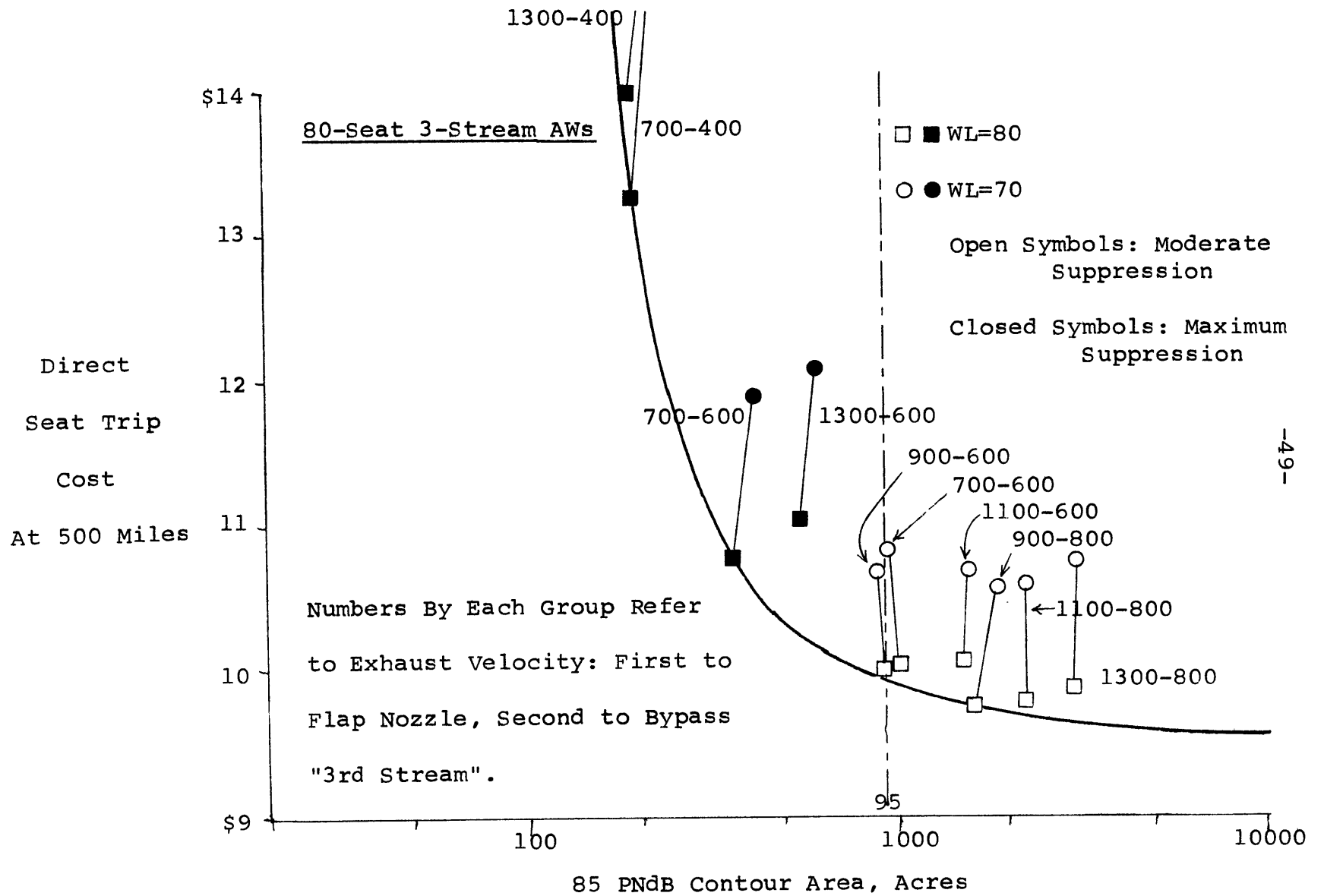


FIGURE 16. Cost and Noise of Three-Stream AW Designs (80 Seats).

brake release point and climbs away on a 10% gradient, and lands on a 7.5° glide slope putting out half as much noise (92 PNdB), the mythical airplane will have a 85 PNdB contour area of 900 acres. The relation between real designs and this baseline is affected not only by the relative noise levels at 500 feet, but also the actual flight path, and the actual noise directivity (the actual designs are very non-isotropic).

The curves fitted to each plot are hyperbolic; that is, they are of the form

$$C = C_0 + d/(A-A_0) \quad (4.1)$$

where C is the cost, and A is the footprint area. A hyperbolic form implies that the cost increases as the area diminishes. Furthermore, there is a lower asymptote on cost, C_0 (no matter how noisy the aircraft, it can cost no less), and an asymptote on area, A_0 (no matter how expensive the aircraft, it can be no quieter). The factor d governs the curvature of the trend curve as it goes from one asymptote to the other.

Thus, the results show, as expected, that quieter aircraft are more expensive aircraft. There are four basic reasons for this.

First, the addition of noise suppression devices to the

engines or nacelles has performance and weight penalties as discussed in section III.B.

Second, the noise of any design is strongly dependent on its basic engine cycle. As the engine exhaust velocity is reduced, resulting in lower noise levels, the required aircraft thrust-to-weight ratio increases. Figure 17 shows a plot of required thrust-to-weight ratios versus exhaust velocity for the two-stream AW designs. Figure 18 shows a similar plot of EBF designs. As can be seen, the lower the exhaust velocities, the higher the thrust-to-weight ratio becomes. This is mainly due to the effect of the inlet momentum drag terms in the expressions for net horizontal force, as given in section II.B. The lower the exhaust velocity, the higher the momentum drag as a fraction of gross thrust at landing or takeoff speed. This is an important penalty of low exhaust velocity power-plants used for deflected thrust which has not received adequate emphasis.

Third, as the exhaust velocity of the engines is lowered, the net thrust drops off more quickly with speed (this is the phenomenon of the previous paragraph in another guise). When the installed thrust levels are set by landing and takeoff requirements, this results in a slower cruise speed, as the decrease in thrust at high speed more than offsets the increase in thrust referred to above. Slower cruise speeds result in

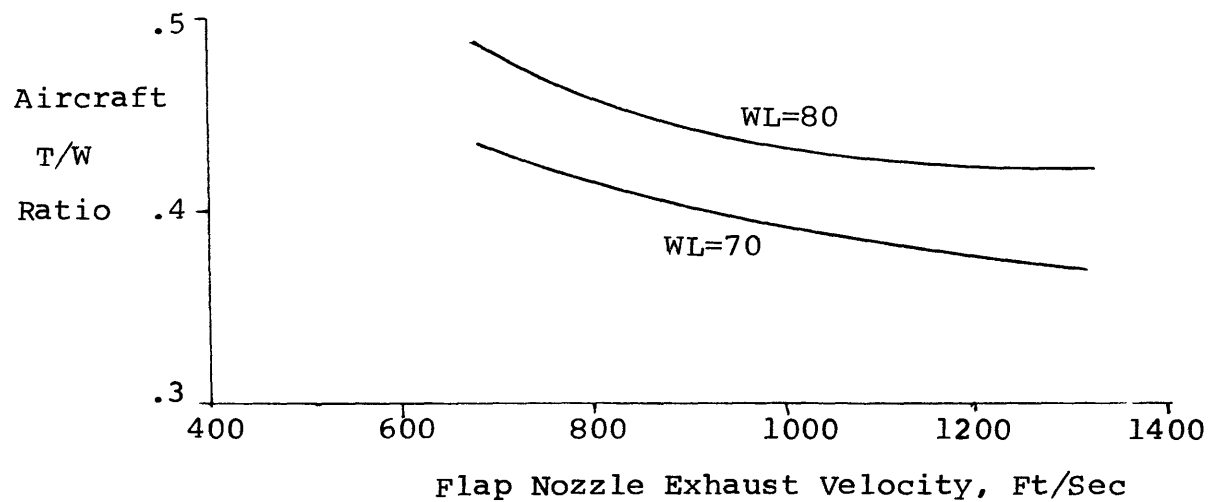


FIGURE 17. Required Thrust-to-Weight Ratios, Two-Stream AW Designs.

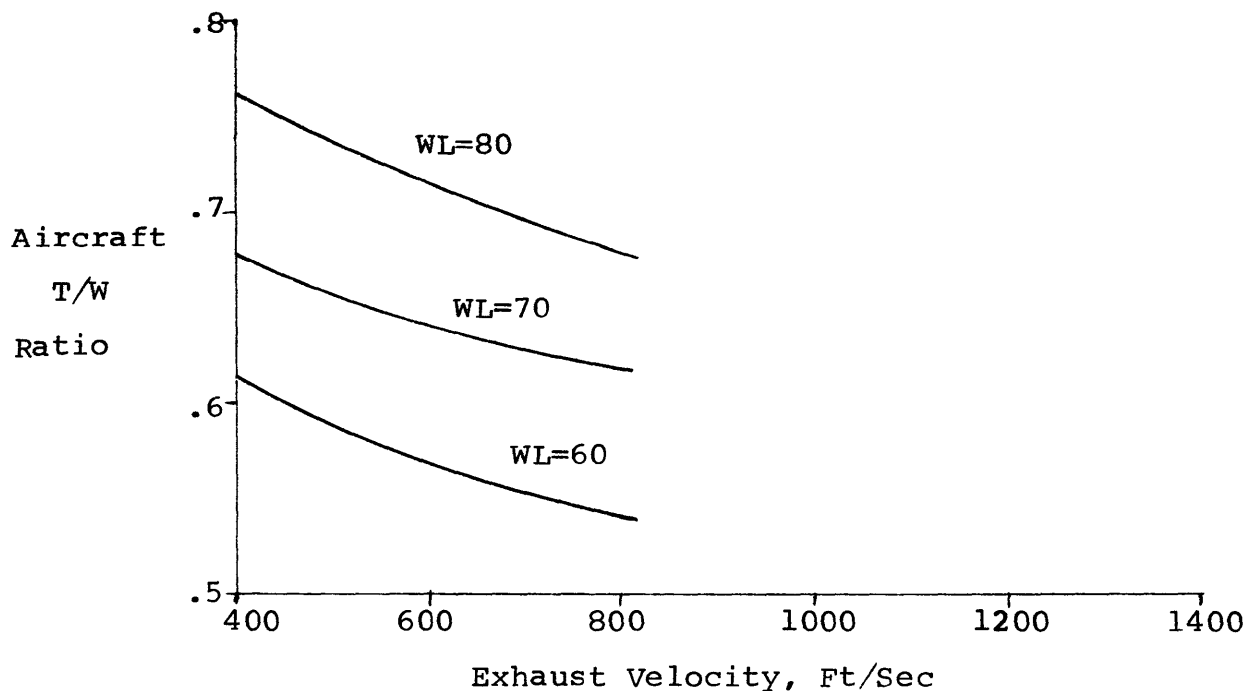


FIGURE 18. Required Thrust-to-Weight Ratios, EBF Designs.

longer block times and higher flight crew, maintenance, depreciation, and insurance costs.

Fourth, below a certain exhaust velocity, the thrust-to-weight ratios of the power-plants become worse (see Figure 4), resulting in heavier, more expensive aircraft.

C. TWO-STREAM AUGMENTOR WING DESIGNS

As can be seen, the two-stream AW types do the best in terms of both cost and noise. The EBF types lose out; in noise, because of the difficulty of making quiet flaps, and in cost due to the high thrust-to-weight ratios of the aircraft. The three-stream AW types lose out too; the suppression of fan noise by absorptive linings is not as effective as sonic choking, the sonic inlets enjoy the advantage of an "on-off" capability, and the high momentum drag penalties mean high thrust-to-weight ratios.

Up to a point, the curve of cost versus noise for the two-stream AW designs is surprisingly flat. The aircraft labelled "Q" (for "quiet") is only 8% more expensive than the aircraft labelled "C" (for "cheap"). This result is surprising in view of the fact that the cheap aircraft is still at least 30% more expensive than a conventional (non-STOL) aircraft of the same capacity.

There are some points that must be borne in mind when comparing the cheap aircraft and the quiet aircraft. Due to the low wing loading and low exhaust velocity of the quiet design, the cruise speed is only 445 mph (385 knots); this compares with the cheap designs's cruise speed of 515 mph (450 knots). From the point of view of the ATA Standard Method, this results in a block time penalty of 9 minutes at 500 miles (9.5%). The ATA Method assumes a ground time of fifteen minutes in its block time calculations. STOL operators could hope for better than that, but even if no ground time at all is assumed, the nine minute difference still means only 11%.

Airlines in the "jet age" tend to be prejudiced against slow airplanes, although passengers will apparently accept anything without propellers. The quiet design is still faster than the Lockheed 188 Electra, which was turning in short-haul block times comparable to McDonnell-Douglas DC-9s as recently as 1972. It would appear that the airlines could be sold on a slow-but-quiet airplane, if they can indeed be sold on STOL.

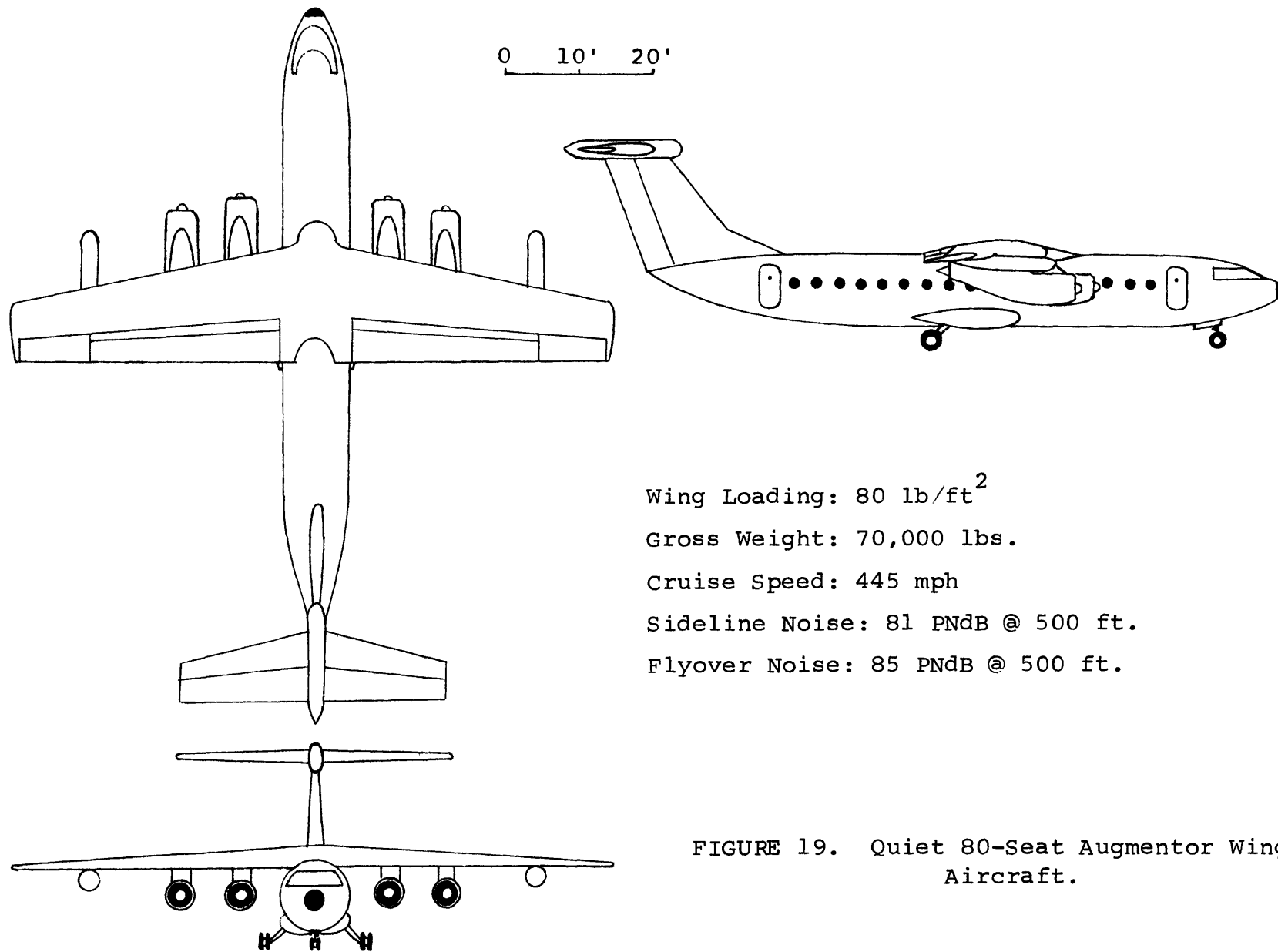
Another point is the wing design of the quiet airplane. Its low exhaust velocity flap nozzles require high mass flows, which in turn require large ducts. In order to fit these ducts inside of the wings, they must occupy part of the volume in front of the rear spar, which of course requires that they pass through

this spar in order to reach the nozzles. Structural designers are quite loath to allow anything to penetrate the webs of the wing spars, due to the adverse effects this has on torsional stiffness. (For example, the tradeoff studies in Reference 5 assume that all of the ducts must fit in the space available behind the rear spar; this severely restricts the usable range of flap nozzle pressure ratios.) Thus, although a 25% weight penalty for the wing was assumed, some uncertainty in size and cost must remain. If the penalty were increased to 50%, the gross weight of the aircraft would increase 5%, and the cost by 4%.

Lastly, external fuel tanks are necessary as much of the wing volume is taken up by ducts. Certain airliners have used similar tanks, so their acceptance should pose no problem.

All in all, there are enough uncertainties to influence the cost of the quiet airplane by around 7%; thus the penalty for using the quiet aircraft instead of the cheap aircraft will range from 8 to 15%.

Figure 19 shows the approximate appearance of a quiet design. Figure 20 shows the cheap design for comparison. They look quite similar, due to their identical wing loading.



Wing Loading: 80 lb/ft²
 Gross Weight: 70,000 lbs.
 Cruise Speed: 445 mph
 Sideline Noise: 81 PNdB @ 500 ft.
 Flyover Noise: 85 PNdB @ 500 ft.

FIGURE 19. Quiet 80-Seat Augmentor Wing Aircraft.

D. "SILENT" AUGMENTOR WING DESIGNS

Aircraft "S" (for "silent") in Figure 15 was designed after examination of the results shown in the rest of the figure. It is an attempt to meet the stiffest noise requirements which are ever likely to be promulgated for STOL aircraft using 2000 foot strips.

An aircraft landing on the STOL runway of Figure 3 will be at an altitude of 100 feet when it is 500 feet from the end of the primary surface. Aircraft passing overhead at altitudes less than 100 feet will probably influence land use and value, no matter what the noise levels involved. One could guess, then, that if the 85 PNdB contour does not extend more than 500 feet beyond the ends of the runway, then the noise of the aircraft will have no effect on the STOLport cost.

On takeoff, this requires noise levels of 80 to 83 PNdB at 500 feet (flyover) if steep climbouts can be used, resulting in altitudes of 300-400 feet over the 500 foot point. On landing, the 100 foot altitude requires a noise level of 71 PNdB at 500 feet. The corresponding sideline noise levels result in the 85 PNdB contour being only 200 feet from the runway centerline. This will obviously require a very quiet airplane.

It is difficult to estimate the noise levels of very low exhaust velocity augmentor wing systems, as this requires ex-

FIGURE 20. Cheap 80-Seat Augmentor Wing Aircraft.

Wing Loading: 80 lb/ft^2
Gross Weight 69,000 lbs.
Cruise Speed: 515 mph
Sideline Noise: 102 PNdB @ 500 ft.
Flyover Noise: 104 PNdB @ 500 ft.

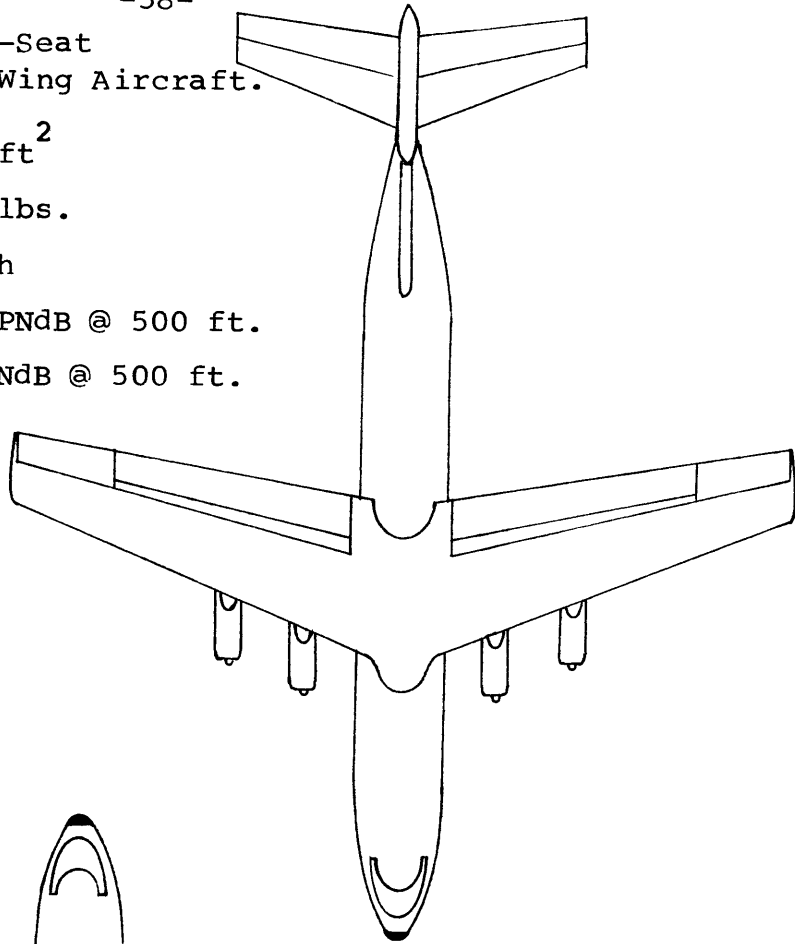
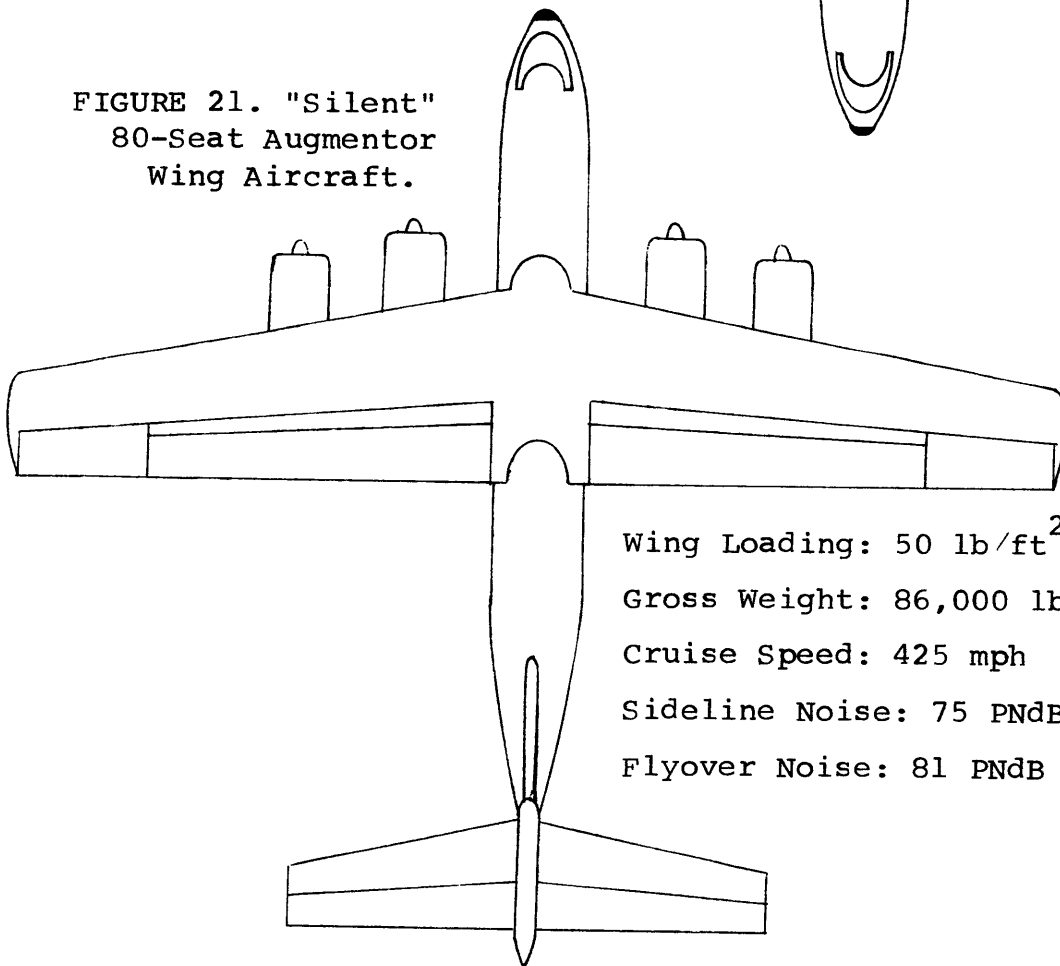


FIGURE 21. "Silent" 80-Seat Augmentor Wing Aircraft.



Wing Loading: 50 lb/ft^2
Gross Weight: 86,000 lbs.
Cruise Speed: 425 mph
Sideline Noise: 75 PNdB @ 500 ft.
Flyover Noise: 81 PNdB @ 500 ft.

trapolations of extrapolations, but it appears that an aircraft with a nozzle velocity of 500 ft/sec and a thrust to weight ratio adequate enough to climb out very steeply would have a fly-over noise level of about 81 PNdB. A wing loading of 50 lb/ft² is required, to accommodate the wing ducts, and to allow very low glide slope thrusts and noise levels. The resulting contours should fit inside the 500-foot limit, but the lower wing loading, the higher aircraft thrust-to-weight ratio, the lower engine thrust-to-weight ratio, and the lower exhaust velocity result in an aircraft with a 23% higher gross weight, a 4% slower cruise speed, and a 25-30% higher DOC when compared to the "quiet" design (a 35-50% higher DOC than the "cheap" design). The appearance of the aircraft is shown in Figure 21.

E. THE EFFECT OF SIZE

When the size of the aircraft (that is, the number of seats) is increased, the resulting aircraft is cheaper to operate per seat, but noisier, due to its higher thrust. When the design evaluations of the two-stream augmentor wing types were repeated for the 50- and 110-seat aircraft, the results were as shown in Figure 22.

At high noise levels, the size advantage (in DOC per seat) outweighs the penalty for quieting the higher noise levels. Be-

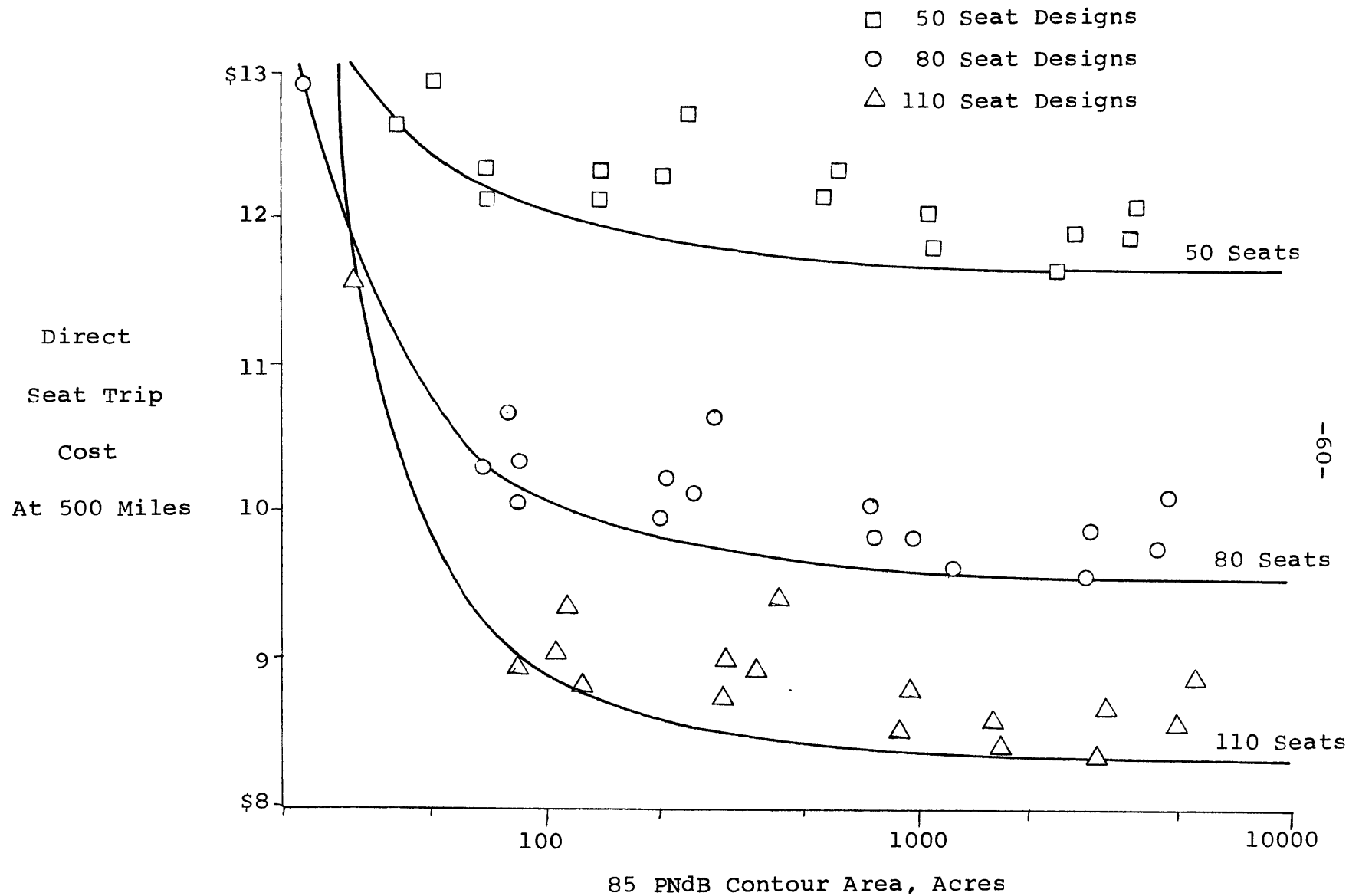


FIGURE 22. The Effect of Size On Cost and Noise (Two-Stream AW Designs).

low certain noise levels, the reverse is true. These crossover noise levels, however, are very low.

F. A COMPARISON WITH THE HELICOPTER

A comparison with the results of a helicopter study (Reference 14) is interesting. Figure 23 shows the costs for 50 seat advanced helicopters and Augmentor Wing STOLs of various degrees of quietness. The cost derivation for the STOLs has been modified from the ATA Method to a method comparable with the DOC method used in Reference 14. The labor costs have been increased by 25%, and the ground time was reduced from fifteen to five minutes (the helicopter costs make no allowance for ground time). Furthermore, a cruise delay time of six minutes was eliminated, leaving a 20 mile terminal maneuver allowance included in the cruise segment of each flight. The sideline noise levels of the aircraft can be compared directly to each other; however, the footprint areas cannot, due to the helicopters' superior takeoff and landing trajectories. Also, the helicopters were designed for a maximum range of only 400 miles.

It is seen that the STOL designs become cheaper at distances over 200 miles for corresponding sideline noise levels.

Figure 24 shows a comparison between STOLs and advanced helicopters of various sizes, all with sideline noise levels in

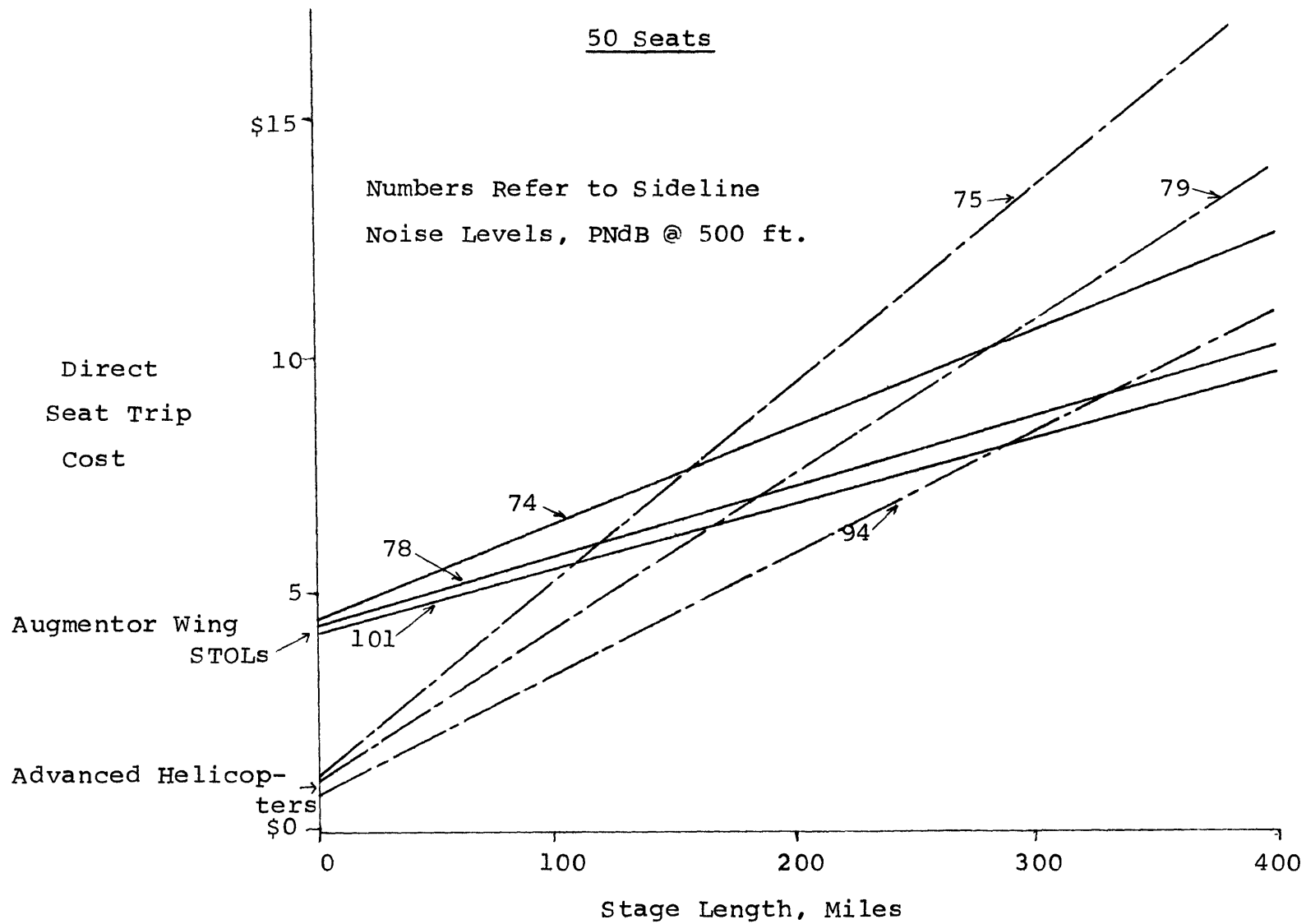


FIGURE 23. Comparison Between Augmentor Wing STOLs and Advanced Helicopters.

the 80 PNdB range at 500 feet. Again the DOCs for the STOLs have been modified from the ATA Method. With these modifications, the crossover in cost was below 200 miles for all sizes (the crossover in blocktime was at 75 miles).

These comparisons indicate that the market for STOL aircraft would be for the longer short-haul stages, that is, over stage lengths of more than 200 miles. This niche for STOLs can be maintained based on any sideline noise requirement, no matter what the size of the aircraft being operated.

If one makes a more logical comparison on the basis of footprint areas, the results are much less conclusive, although it becomes reasonably clear that the STOL must suffer. A helicopter should be able to takeoff vertically to a height where the noise level on the ground is less than 85 PNdB, and land vertically from the same height. Thus the takeoff and landing footprint would be a circle caused by the liftoff noise. The low noise levels of some of the helicopter designs would result in very small circles.

However, from this point on, the comparison of STOL and helicopter footprints becomes very dependent on other operational considerations. For example, the STOL aircraft all take off and land from the same runway, but helicopters might operate from several pads, possibly as many pads as there are gates.

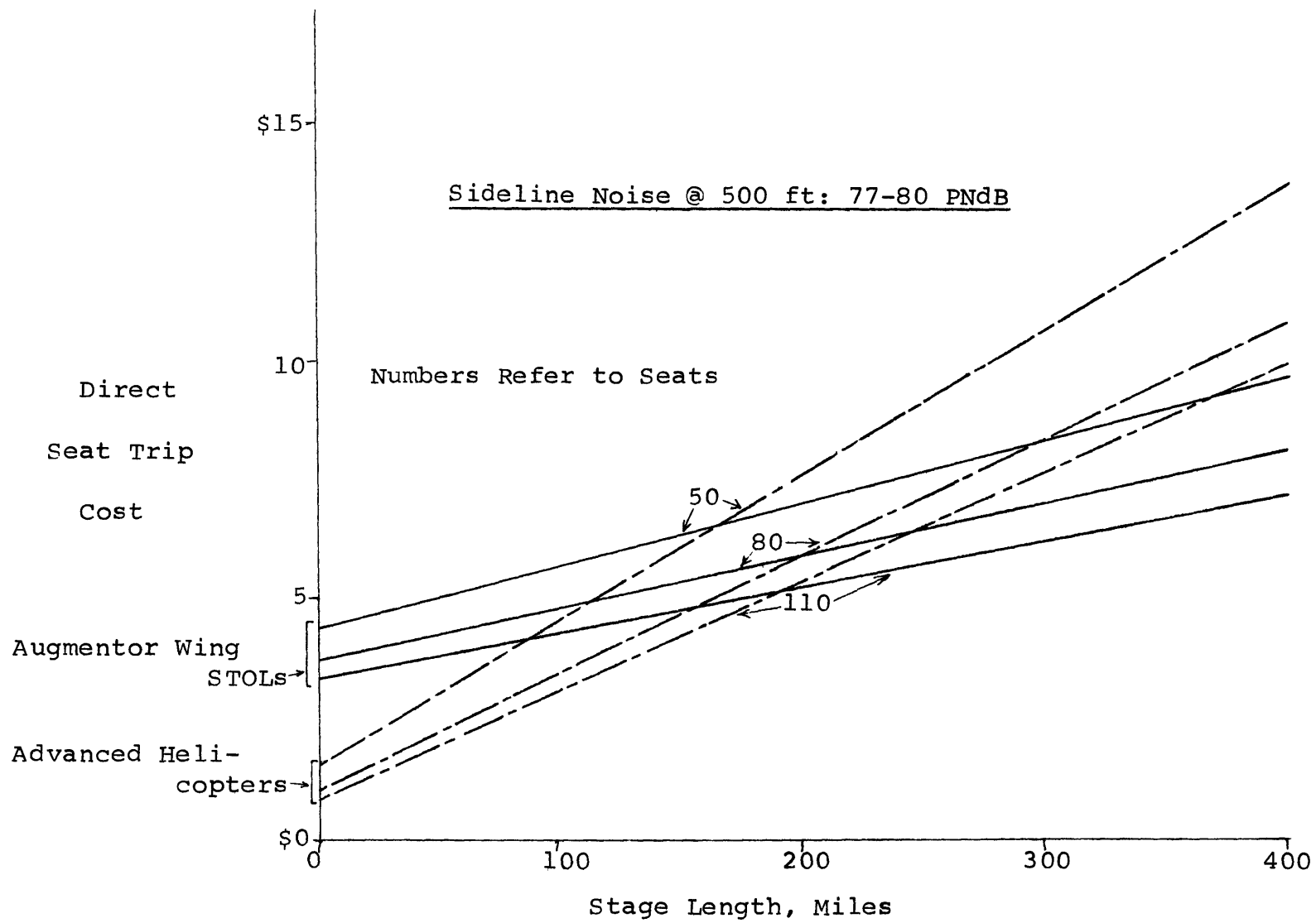


FIGURE 24. Comparison Between Augmentor Wing STOLs and Advanced Helicopters of Various Sizes.

This has the effect of spreading out the helicopter noise depending on the number and spacing of the pads.

Another confusing consideration is the definition of the airport boundary. Obviously, noise within the airport boundary has no effect on operations. The "silent" augmentor wing was designed so that the levels at 500 feet from the runway ends would be no greater than 85 PNdB. Should a similar 500 foot allowance be made for a heliport? On all sides?

In view of the above considerations, it is difficult to say which STOL corresponds to which helicopter on the basis of footprint area. The crossover in DOC between STOLs and helicopters of the same noise impact could range from 200 to 500 miles, depending on assumptions.

V. CONCLUSIONS

The results of the design evaluations show that the two-stream Augmentor Wing STOL transports are less expensive to operate for any permitted footprint area than either the three-stream Augmentor Wing STOLs, or the Externally Blown Flap STOLs.

At high noise levels the superiority is marginal, and could be eliminated if certain assumptions are changed. At low noise levels, however, the superiority is quite marked. No other other type, including low-wing-loading (non-powered-lift) types, can approach their quietness, due to the effectiveness of the sonic inlets and the long flap nozzle ducts, which are peculiar to these types. The cost of achieving these low noise levels is small (up to a point), due to the efficiency of the augmentor flaps, and the "on-off" capability of the sonic inlets.

It is difficult to make a direct comparison between Augmentor Wing STOLs and advanced helicopters, but even a comparison of aircraft of equal noise which is probably over-favorable to the STOLs shows that the helicopters would have lower operating costs on stage length of less than 200 miles. Operational considerations which are beyond the scope of this study might change the relative noise requirements to the advantage of the helicopters, making them more economical on stage lengths of up

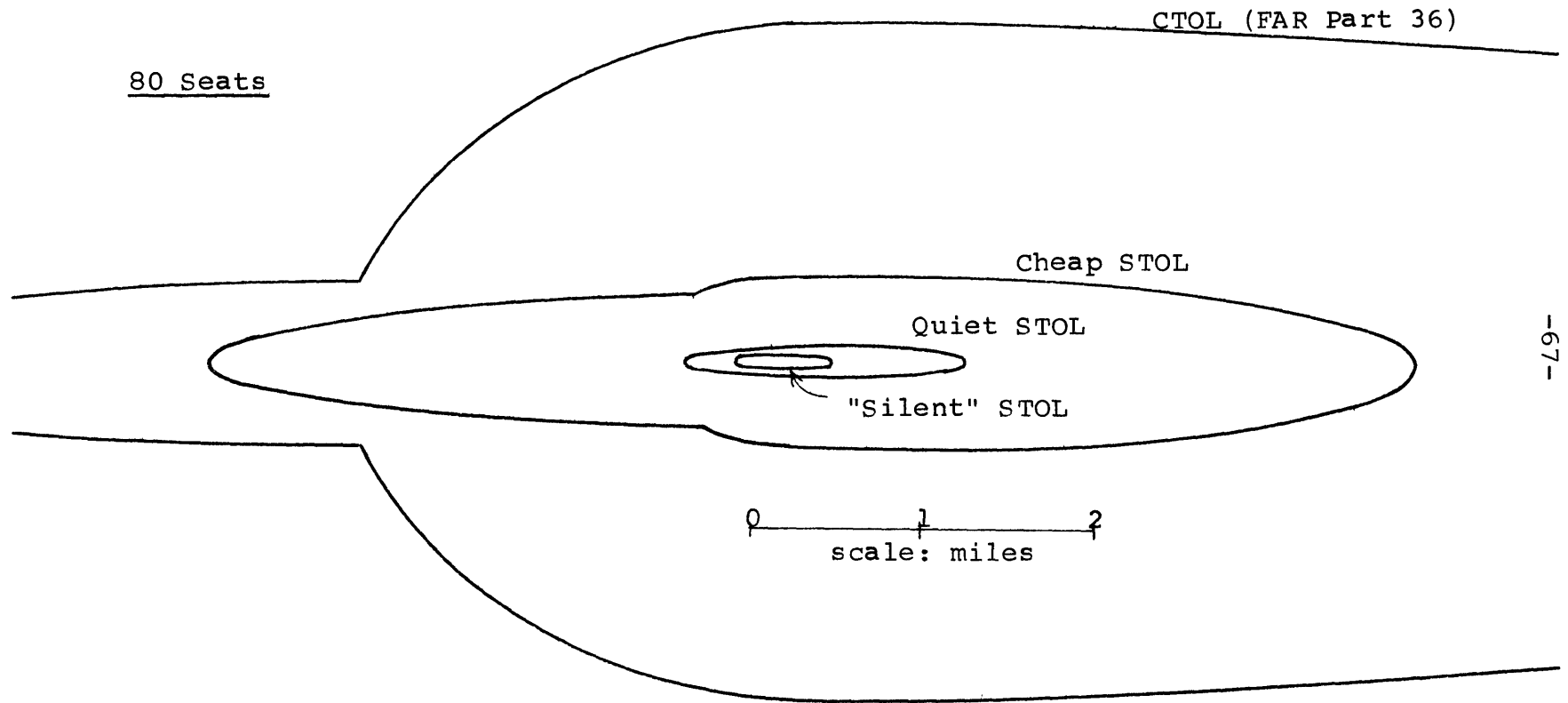


FIGURE 25. 85 PNdB Contours for Various 80 Seat Aircraft.

to 500 miles.

Figure 25 show the 85 PNdB contours for the "cheap", "quiet;" and "silent" Augmentor Wing STOLs, compared to the contour for a conventional 80 passenger aircraft meeting FAR Part 36. The figure shows that STOL transport aircraft, if one is willing to pay the price, can be made so quiet so as to be practically un-noticable in an urban environment.

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