THE COST OF NOISE REDUCTION IN HELICOPTERS

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ABSTRACT

The relationship between noise reduction and direct operating cost was studied for transport helicopters. A large number of helicopter preliminary designs was generated with the help of a computer program. Vehicles were selected to meet certain noise goals with minimum direct operating cost; this was repeated for several payloads and technology time frames. The effect of changes in the assumed mission profile was studied.

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1.0 Introduction

The helicopter has become an important means of transportation in densely populated regions. Land is scarce and surface transportation is slow in these regions. Here the higher operating costs of the helicopter can be offset with its small land requirements and the resulting ability to locate numerous terminals. In the next decade helicopter transportation is expected to expand rapidly if noise abatement constraints are met.

In this report the emphasis is on helicopters for intercity transportation, covering stage lengths of 50 to 400 miles. However, the results could be applied to intraurban helicopters operating on shorter stage lengths.

In recent years a strong adverse public reaction to aircraft noise has developed. Noise reduction is now an important, if not dominant, objective in air transportation planning. The helicopter is inherently one of the quietest types of transport aircraft. However, it is likely to operate closer to a greater number of listeners than other types because of the small size of the terminals it operates from and the greater number of them. Therefore it is essential to the success of helicopter transportation that its potential for low noise operations be exploited as far as possible.

There are two methods of reducing the noise exposure due to aircraft operations. One is to change the flight profile. The aircraft trajectory can be moved further from the listeners, the amount of noise generated can be reduced by changing thrust, or the speed can be increased in order to reduce noise exposure time. This method of noise reduction is explored in references 8 and 9. The second method is to change the design of the aircraft to reduce the noise generated at a given distance, thrust level, and speed. The second method is given primary emphasis here. However, substantial changes in the flight profile, which affect the design, are also considered.

It is worth remembering that existing helicopters were not designed with noise reduction as a design objective at the outset. Modifications have been made to a few existing helicopters to reduce noise, often accompanied by a significant loss of payload. This does not indicate, however, that new helicopters cannot be designed to achieve substantial noise reduction with a moderate increase in direct operating cost. It is also worth remembering that all existing large helicopters were designed for military use, and hence a decrease in direct operating cost might be achieved by designing primarily for a civilian transport role.

The purpose of this work is to identify those design changes which can reduce noise with the minimum cost penalty and to develop the relationship between the amount of noise reduction and the resulting cost penalty.

Miller (Reference 10) performed an initial study of these questions. By developing a series of helicopter preliminary designs, he explored the relationship between design parameters, direct operating cost, and noise generated. A computer program was used to aid in the design iterations. Curves of hover noise versus hover tip Mach number and direct operating cost (DOC) versus hover noise were developed for a series of 80 passenger helicopters. These curves were generated by varying either the hover tip Mach number, or the thrust coefficient to solidity ratio, while holding other parameters constant.

In this work a different approach is taken using a more sophisticated helicopter design computer program. Takeoff and cruise noise objectives were set along with size, technology time frame (year of first flight), and operational constraints. Then all other parameters were varied to produce a vehicle with minimum direct operating cost which met the noise objectives. This was then repeated for three other levels of noise objectives to find the relationship between noise level and direct operating cost. This basic variation was then extended to different sizes and time frames. Finally, the effect of different operational constraints on noise and direct operating cost was examined.

2.0 Helicopter Design Procedure

2.1 General

The process for preliminary design of air vehicles can be computerized such that parametric variations can be obtained rapidly. These computer programs are now a design tool used to find the optimal configuration for a given vehicle performance requirement in terms of size, speed, range, direct operating cost, etc. Estimated noise generation is now included as one of the performance measures of the vehicle. Other design objectives can be met at varying levels of noise, or an optimal design can be found for a specified noise level.

2.2 Description of the Helicopter Design Computer Program

a) The Design Logic

The helicopter computer design program is fully described in Flight Transportation Laboratory Technical Memo 71-3 (Reference 1). This program considers only conventional pure helicopters. (See Appendix for helicopter terminology.)

The program begins by reading input data such as cabin size, range, speed, etc. and generating constants, including atmospheric data, for later use. Calculations regarding hover performance are done for a hot day; all other calculations assume a standard day.

Then the program goes into a design procedure which is an iteration on gross weight. Initially a gross weight is estimated based on the design payload; on succeeding iterations the previous gross weight is used. The rotor is then designed considering both cruise and hover. It is assumed that there are two rotor angular velocities: the rotor turns at hover rpm when the advance ratio is less than 0.325 and cruise rpm otherwise. Next the fuselage is sized and parasite drag is calculated. Then the power plant and drive system is sized to the maximum of cruise and hover requirements. If hover rpm is less than cruise rpm then the installed power required for hover is increased to account for reduced engine output below rated rpm. This completes the selection of design parameters.

The vehicle is then flown through the design mission to find the fuel consumed. Nine phases in the mission profile are considered: hover, vertical climb, acceleration to climb advance ratio, unaccelerated climb to cruise altitude, acceleration to cruise, cruise, undecelerated descent, deceleration to hover, and vertical descent. The time, distance and fuel consumed in each phase is calculated. An input table of rotor lift-to-drag ratio as a function of advance ratio and thrust coefficient to solidity ratio is used to estimate performance above advance ratio .325.

Then the component weights are calculated, resulting in a new gross weight. If the difference between new and old gross weights is greater than 10 pounds, the design procedure goes through another cycle. When the iteration is complete the parameters describing the final design are printed.

b) <u>Vehicle Operating Cost</u>

The vehicle then is flown through various stage lengths that are less than the design range, with appropriate cruise altitudes and speeds. The time, distance, and fuel consumed for each phase of each stage is calculated, printed, and stored for use in the calculation of direct operating cost (DOC).

Then the program calculates DOC's for each stage length, breaks them down by categories, and prints this out. The DOC is calculated according to the Lockheed VTOL formula. (References 3 and 4).

c) Vehicle Noise Generation

As the last step, the program calculates the noise generated by the vehicle. There are three principal noise sources in a helicopter: the rotors, the engine, and the transmission. Modern commercial helicopters are powered by turboshaft engines. The methods used to quiet these engines and the transmission are quite straightforward and have a relatively small effect on DOC. This effect is accounted for by assuming a weight penalty in the engine. Above 90 dB perceived noise level at 500 feet no penalty is assumed. The input horsepower/weight ratio is decreased approximately 20% for each 10 dB reduction below 90 dB. The weight penalty for quieting the tail rotor on single rotor ships is assumed to be insignificant. Thus noise sources other than the main rotor(s) are assumed to be quieted singificantly below the main rotor(s).

Overall sound pressure level for the rotor(s) at 300 feet distance is calculated using the following well established formula taken from reference 5 for vortex noise. This formula is applied to all flight conditions. In cruise, the advancing blade tip speed is used.

$$L_p = 10 \log_{10} \left[\frac{7.62 \times 10^{-10} \, T^2 (V_{tip})^2}{\rho^2 A_B} \right]$$

where L_p = overall sound pressure, db

T = thrust, lb.

V_{tip} = rotor tip speed, ft/sec

ρ = air density, slugs/ft^3

A_B = total rotor blade area, ft^2

Rotational noise was hand-calculated for a sample case using both the method of Ollerhead and Lowson (Reference 6) and a method developed in the Flight Transportation Laboratory (Reference 7). Both results indicated that rotational noise was not significant for helicopters with low tip speeds, and thus it has not been included in the program. Recent research (Reference 12) has indicated that a large part of what used to be thought of as vortex (broadband) noise may in fact be largely composed of rotational noise. This does not affect the accuracy of empirical predictions of overall sound pressure level, however.

Simple inverse square law attenuation is used to for observer distance other than 300 feet from the vehicle. No directivity in azimuth is assumed. The method given in Reference 6 for vortex noise is used for directivity in elevation. A factor DF is added to the overall sound pressure level:

$$DF = 10 \log_{10} \left[\frac{\cos^2 \phi + 0.1}{\cos^2 70^\circ + 0.1} \right]$$

where ϕ = angle at the rotor hub between the rotor shaft axis an a line joining the rotor hub and the observer.

This factor varies from +7.0 along the shaft axis to -.34 in the rotor plane. The overall sound pressure level is converted to perceived noise level using an assumed frequency distribution from Reference 11.

The standard takeoff profile assumed in the helicopter design program is shown in Figure 1. Climb power (1.2 times normal rated power) is used throughout the takeoff profile. During the acceleration phase, the vehicle tries to accelerate horizontally at a given maximum acceleration, and if it has more than enough power to do this, it uses the excess power to climb. Hence the

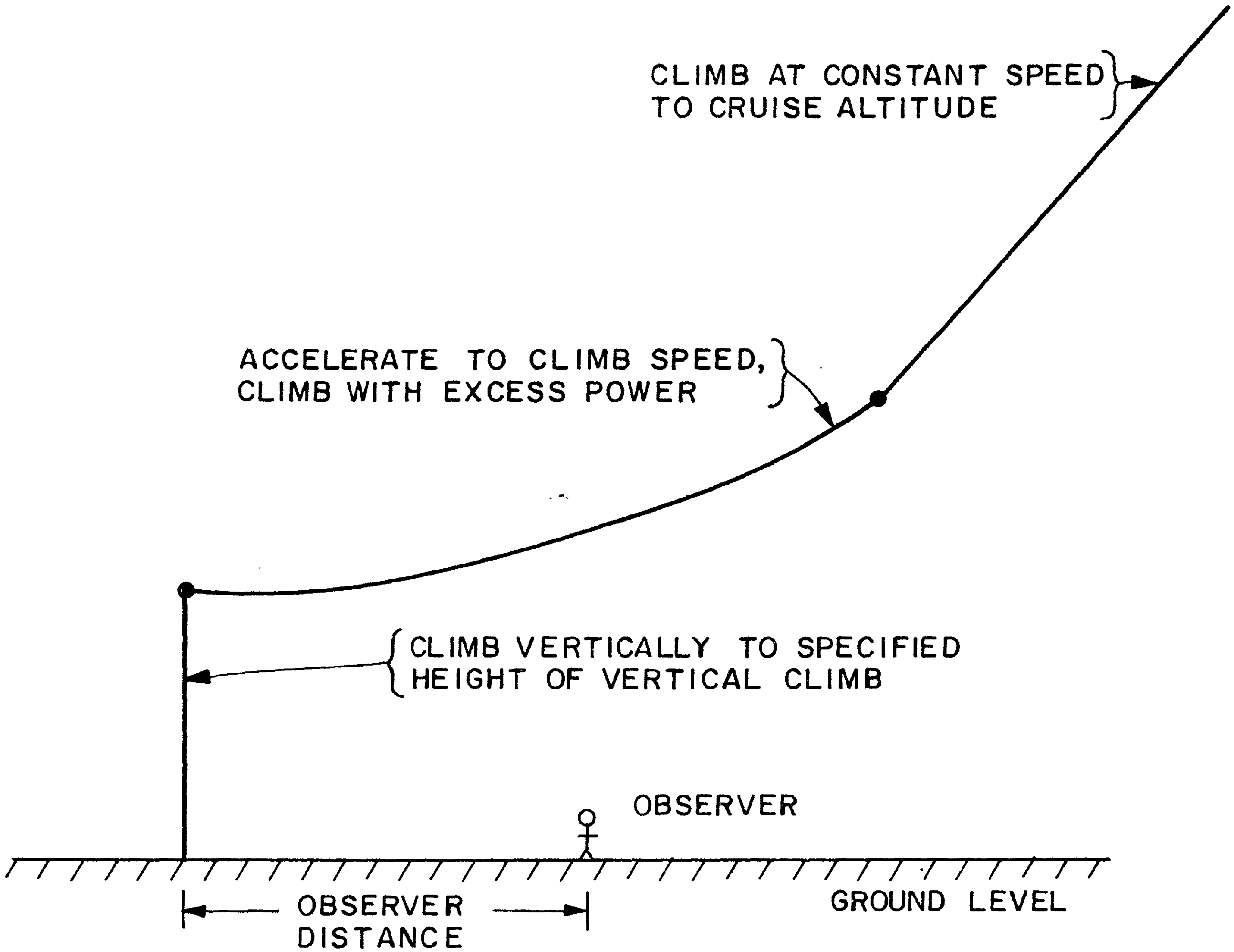

FIGURE 1 SCHEMATIC OF TAKEOFF PROFILE

profile varies depending on how much power is available and the maximum acceleration allowed. During the climb phase the vehicle climbs at a constant forward speed. The observers are always in the plane of the takeoff profile. Varying the height of vertical climb has the effect of shifting the flight profile up or down. Reducing the maximum acceleration causes greater excess power to be available for climb and hence has the effect of tilting the path upward during acceleration.

As the vehicle accelerates from rest to its vertical rate of climb, thrust is greater than weight and hence more noise is generated. The noise resulting from maximum thrust is calculated and assumed to represent the noise in the first few seconds of the takeoff profile. This is called noise at liftoff.

The noise is calculated for a single observer at 15 points during the takeoff profile and output along with the time, altitude and horizongtal distance corresponding to each point. This can be repeated for observers at different distances from the takeoff point. Noise on the ground due to the vehicle passing directly overhead at cruise altitude (peak flyover noise) is also calculated.

Noise is not calculated for the landing profile. However, the landing profile is nearly the reverse of the takeoff profile. Idle power is used for descent and deceleration. Most of the descent is made at cruise speed. Then the vehicle decelerates at a specified allowable deceleration using excess drag for the rest of the descent to a specified height of vertical descent. Vertical descent to touchdown is made at a specified maximum vertical descent rate to avoid entering the vortex ring state. Thus the landing profile does not differ fundamentally from the reverse of the takeoff profile, but somewhat different distances and speeds may be involved in each phase.

2.3 Noise Reduction

This section describes the procedures involved in varying design parameters to achieve low levels of rotor noise for a given mission. (See Appendix for helicopter terminology).

The formula for overall sound pressure level for vortex noise from section 2.2 may be rewritten as follows:

$$L_p = 10 \log_{10} \left[3.04 \times 10^{-9} \, (C_L)^2 \cdot A_B \cdot (V_{tip})^6 \right]$$

where C_L = average blade lift coefficient $\approx 6 \, (CT/\sigma)$

Of the three variables, it is clear that since V_{tip} is raised to the sixth power, it is dominant in reducing the overall noise level.

Consider cruise noise reduction first. Since the tip speed in the formula above is taken to be the advancing blade tip speed, cruise noise is reduced by reducing the advancing blade tip Mach number, M_{at}.

Now consider hover (or low speed flight) noise reduction. The rotor thrust in hover, which must remain constant, is given by the following relation:

$$T_h = \rho_h A_B \, V_{th}^2 \, (C_T/\sigma)_h$$

where ρ_h = air density for hover conditions

V_{th} = rotor tip speed in hover

$(C_T/\sigma)_h$ = thrust coefficient to solidity ratio in hover

A small decrease in hover tip speed from normal practice can be obtained by increasing $(C_T/\sigma)_h$ to a maximum of 0.10. Beyond this value blade stall becomes critical and blade area must be increased either by increasing solidity, σ, or decreasing disc loading, DL.

However, changes in cruise parameters must accompany the increase in blade area because cruise thrust, T_{cr}, must also remain approximately constant. The following relations apply:

$$T_{cr} = \rho_{cr} A_B V_{tcr}^2 (C_T/\sigma)_{cr}$$

and $$V_{tcr} = \frac{a_{cr} M_{at}}{1 + \mu}$$

where ρ_{cr} = air density for cruise conditions

V_{tcr} = rotor rotational tip speed in cruise

$(C_T/\sigma)_{cr}$ = thrust coefficient to solidity ratio in cruise

a_{cr} = speed of sound for cruise conditions

μ = advance ratio

Thus an increase in blade area would be accompanied by a decrease in $(C_T/\sigma)_{cr}$ or V_{tcr} for constant thrust. A decrease in V_{tcr} means an increase in advance ratio or a decrease in advancing tip Mach number. Conversely, a decrease in M_{at} for cruise noise reduction must be accompanied by an increase in μ, and increase in A_B, or an increase in $(C_T/\sigma)_{cr}$. To reduce noise for hover and cruise conditions simultaneously, A_B and μ would be increased and both $(C_T/\sigma)_{cr}$ and Mat would be reduced.

The noise prediction formula used here was developed from a correlation of design parameters with measurements of noise from helicopters and rotors (Reference 2). These helicopters and rotors had solidities and disc loadings typical of designs which are unconstrained by noise considerations. As the solidity is increased and disc loading reduced to reduce noise, this empirical noise prediction formula becomes less valid. Further experimental data on the noise generation of low disc loading high solidity rotors is required to develop a more generalized formula. Until this is available, prediction of large noise reductions based on this formula must be regarded as preliminary. The same argument can be applied to the method of predicting high speed rotor performance. Experimental performance data is also needed for high solidity, low disc loading rotors.

Variations in detail rotor blade geometry are not considered here. New tip planforms and twist distributions can reduce noise somewhat beyond the levels shown here. These changes do not generally result in a significant weight or performance penalty, and hence do not affect DOC. Therefore they do not change the nature of what is said here.

2.4 Design Constants

All of the helicopters in this report, except E70-50, are designed to be able to hover on a hot day with one engine out.

A number of inputs to the helicopter design computer program were kept constant throughout the work reported here. The values of these are presented in Table 1.

Table 1 : Input Constants

Climb Advance Ratio	= 0.30
Rotor Equivalent Lift/Drag	= See Table 2.
Standard Temp.	= 59° F
Hot Day Temp.	= 95° F
Reserve	= 20 min. at cruise power
Rate of Vertical Descent	= 600 feet/minute
Allowable Deceleration	= .20 g
Utilization	= 2300 hour/year
Depreciation Period	= 12 years
Airframe Cost	= 70 $/pound
Engine Cost	= 50 $/hp.
Insurance Rate	= 2%/year
Labor Rate	= 5 $/hour

Table 2 : Rotor Equivalent Lift/Drag Ratio as a Function of Advance Ratio, μ, and Thrust Coefficient to Solidity Ratio, C_T/σ

C_T/σ \ μ	.60	.55	.50	.45	.40	.35	.30
.075			9.0	9.4	9.6	9.3	8.4
.070			9.2	9.6	9.8	9.5	8.6
.065		8.3	9.1	9.5	9.7	9.4	8.5
.060		8.1	8.8	9.2	9.4	9.1	8.3
.055	7.3	7.6	8.3	8.7	8.9	8.6	7.8
.050	6.9	7.1	7.8	8.1	8.3	8.1	7.3
.045	6.2	6.5	7.1	7.4	7.5	7.3	6.6
.040	5.6	5.8	6.4	6.6	6.8	6.6	5.9
.035	4.9	5.1	5.6	5.8	6.0	5.8	5.2
.030	4.2	4.3	4.7	5.0	5.1	4.9	4.4

Note : This table was derived using the performance of existing helicopters and preliminary rotor performance prediction studies in the Flight Transportation Laboratory.

3.0 Results

3.1 Nomenclature

The helicopter designs described here are designated by codes consisting of a letter and two numbers. The letter indicates the noisiness class according to the following mnemonics:

C - Cheap - unconstrained

M - Medium - moderately quiet

Q - Quiet - very quiet

S - Silent - extremely quiet

The first number indicates the technology time frame. Here the time frame is the year in which a production prototype could be flying, using the latest technology both in design and manufacturing. The second number indicates the size as measured by passenger seats. For example, Q75-50 is a very quiet helicopter, designed using 1975 technology and carrying 50 passengers. An exception to this is E70-50, which represents an approximation of a helicopter existing in 1970, the Vertol 347.

Other nomenclature is shown below:

$(L_{PN})_{to}$ = perceived noise level at liftoff

$(L_{PN})_{cr}$ = perceived noise level in cruise overhead

GW = gross weight

V_{cr} = cruise speed

NRP = normal rated power

$(L/D)_{cr}$ = overall lift to drag ratio in cruise

GBH = gear box factor in hover (factor used to determine the drive system limited power at hover rpm)

D = Rotor diameter

C = Rotor blade chord

H_{vc} = height of vertical climb and descent

$(a/g)_{max}$ = maximum allowable forward acceleration

H_{cr} = cruise altitude

3.2 Basic Variation

The basic variation consists of four tandem helicopters, designed to meet four different noise level objectives. The payload, design time frame, and operational constraints were kept constant as shown in Table 3. All other parameters were varied to produce vehicles which met the noise objectives with minimum direct operating cost, as shown in Table 4.

The fifth vehicle shown in Table 4, called E70-50, is an approximation to a helicopter existing in 1970, the Vertol 347. This vehicle was included to add perspective by showing what is available now. It has the same payload, range and operating constraints as the other machines, but lacks engine-out hover capability.

The C and S vehicles were chosen to represent the extremes of the noise level spectrum for this kind of aircraft. It is unlikely that any future civilian transport helicopter would be designed without regard for noise reduction, as the C vehicle is. The S vehicle, on the other hand, carries the noise reduction techniques described above to the fringe of practicality.

One of the features of the S vehicle which is most likely to be impractical is the high solidity. The rotors for the four basic helicopters are shown to scale in Figure 2. A solidity of 0.25 has generally been considered the maximum practical in previous work in the Flight Transportation Laboratory. However, it can be argued that the practical limit is somewhat less. The sensitivity of DOC to solidity was studied to determine how serious this problem was for the Q and S vehicles. The result, shown in Figure 3, is

Table 3 : Parameters Held Constant for Basic Variation

Seats	=	50
Time Frame	=	1975
Design Range	=	400 miles
Height of Vertical Climb	=	500 feet
Cruise Altitude	=	5000 feet
Maximum Acceleration	=	0.25

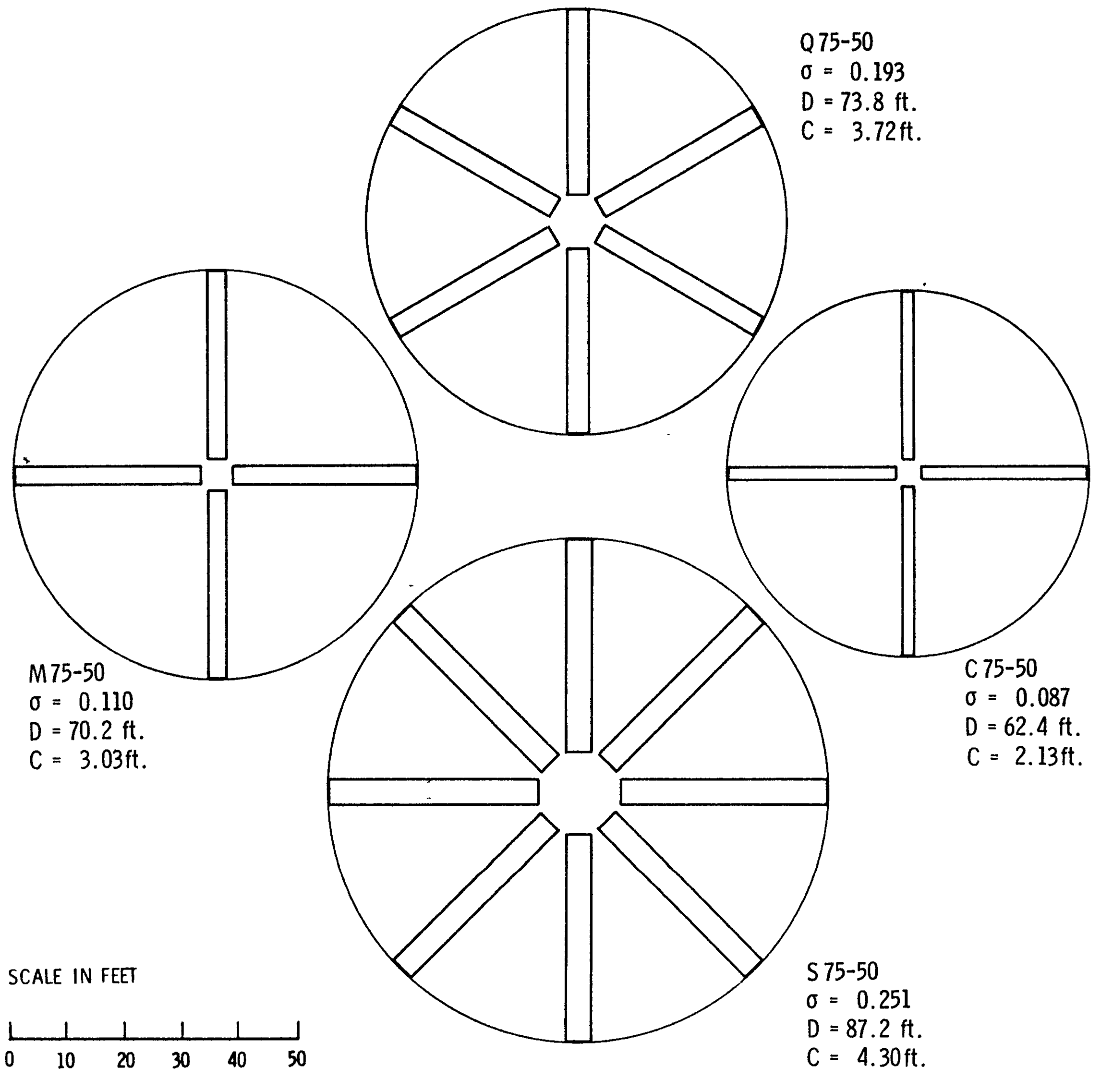

Fig. 2 Rotors for basic helicopters.

Table 4 : Parameters Describing Four Basic Helicopters and E70-50

	E70-50	C75-50	M75-50	Q75-50	S75-50
$(L_{PN})_{to}$, dB	95.0	93.6	85.2	79.2	74.9
$(L_{PN})_{cr}$, dB	84.1	82.5	77.6	73.2	69.4
DOC @ 100 mi, $/seat trip	4.36	3.36	3.65	4.25	5.23
GW, lb	46,186	36,774	38,637	42,739	47,855
NRP, hp	7133	6280	5964	6593	6570
DL, lbs/ft^2	8.2	6.0	5.0	5.0	4.0
σ	0.089	0.087	0.110	0.193	0.251
V_{cr}, mph	189	237	219	196	168
$(L/D)_{cr}$	4.31	4.89	5.00	5.07	5.42
μ	0.40	0.50	0.55	0.60	0.60
M_{at}	0.883	0.95	0.825	0.70	0.575
$(C_T/\sigma)_{cr}$	0.093	0.070	0.065	0.055	0.050
V_{tcr}, ft/sec	692	694	584	480	411
GBH	1.05	1.20	1.10	1.02	1.00
$(C_T/\sigma)_h$	0.081	0.063	0.100	0.100	0.100
V_{th}, ft/sec	692	680	437	330	270

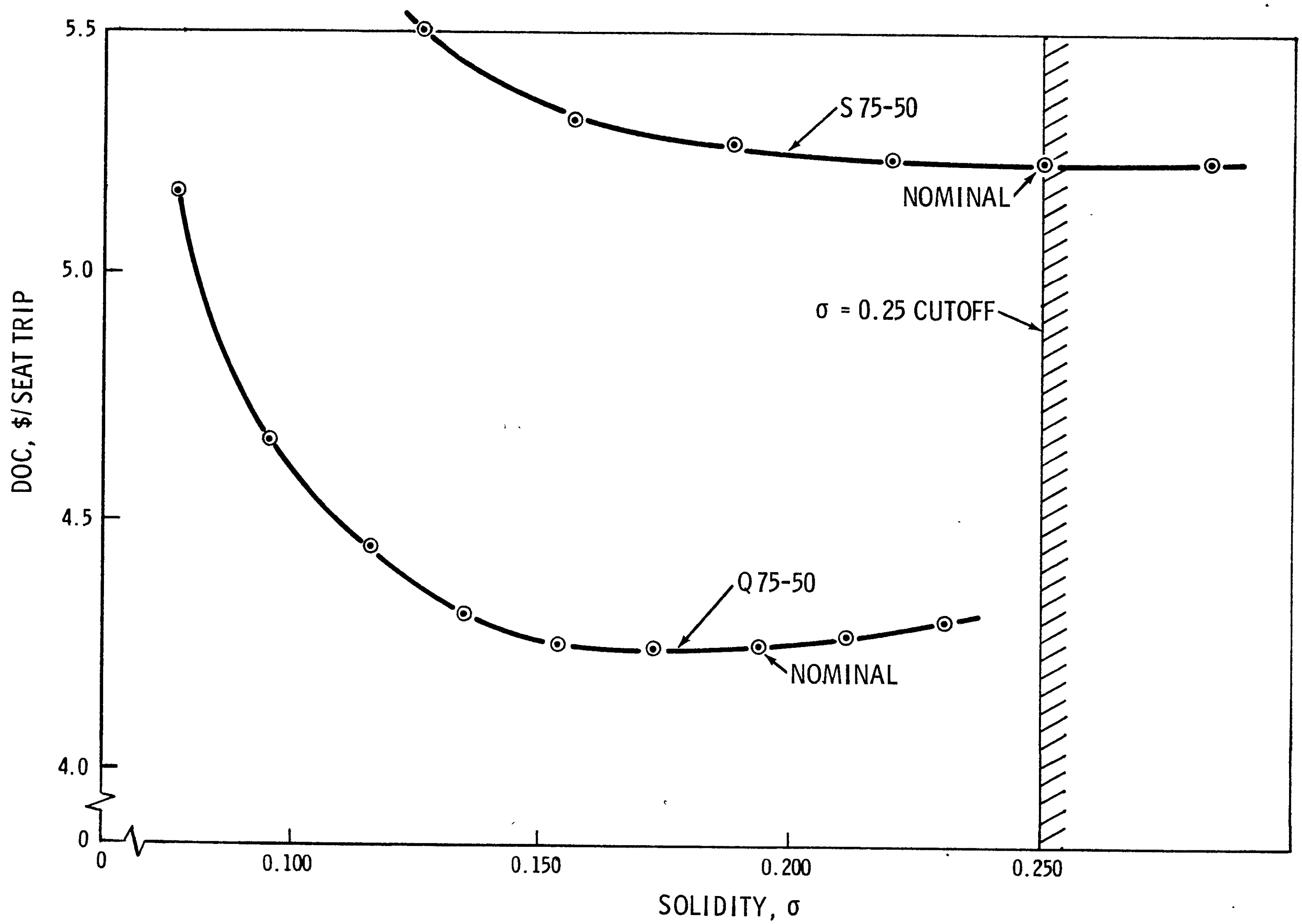

Fig. 3 DOC @ 100 mi. vs solidity for Q75-50, S75-50.

that the solidity can be reduced to about 0.15 before DOC begins to rise significantly.

In general, the parameters in Table 4 show a monotonic variation with noise level. It is interesting to note that overall lift to drag ratio increases with decreasing noise level because cruise speed is decreasing. The variation of NRP seems slightly erratic. It decreased from C75 - 50 to M75 - 50 because cruise power required is decreasing. In Q75 - 50, however, NRP is set by hover power requirements which have increased from M75 - 50 to Q75 - 50. The hover power requirements would increase still further in S75 - 50 except that disc loading was decreased.

It should be borne in mind that cruise noise (peak flyover noise) and liftoff noise are being reduced simultaneously here. A particular proportion of cruise noise reduction to liftoff noise reduction has been assumed, about 3 dB in cruise for each 5 dB at liftoff. The interrelationships between cruise and low speed parameters were discussed in section 2.3. Different proportions between the noise reduction goals would produce different optimum vehicles.

DOC is plotted against liftoff noise and cruise noise in Figures 4 and 5 respectively. These are the basic cost vs. noise reduction relationships that were sought. As expected, noise reduction returns, per unit increase in DOC, diminish as we move toward quieter vehicles.

DOC at 100 miles stage length was taken as representative of typical intercity operations. DOC at other stage lengths can be found in Figure 6.

Liftoff noise was chosen here as a measure of terminal area noise because it is independent of the takeoff path. However, it is clearly only one dimension of the terminal noise picture.

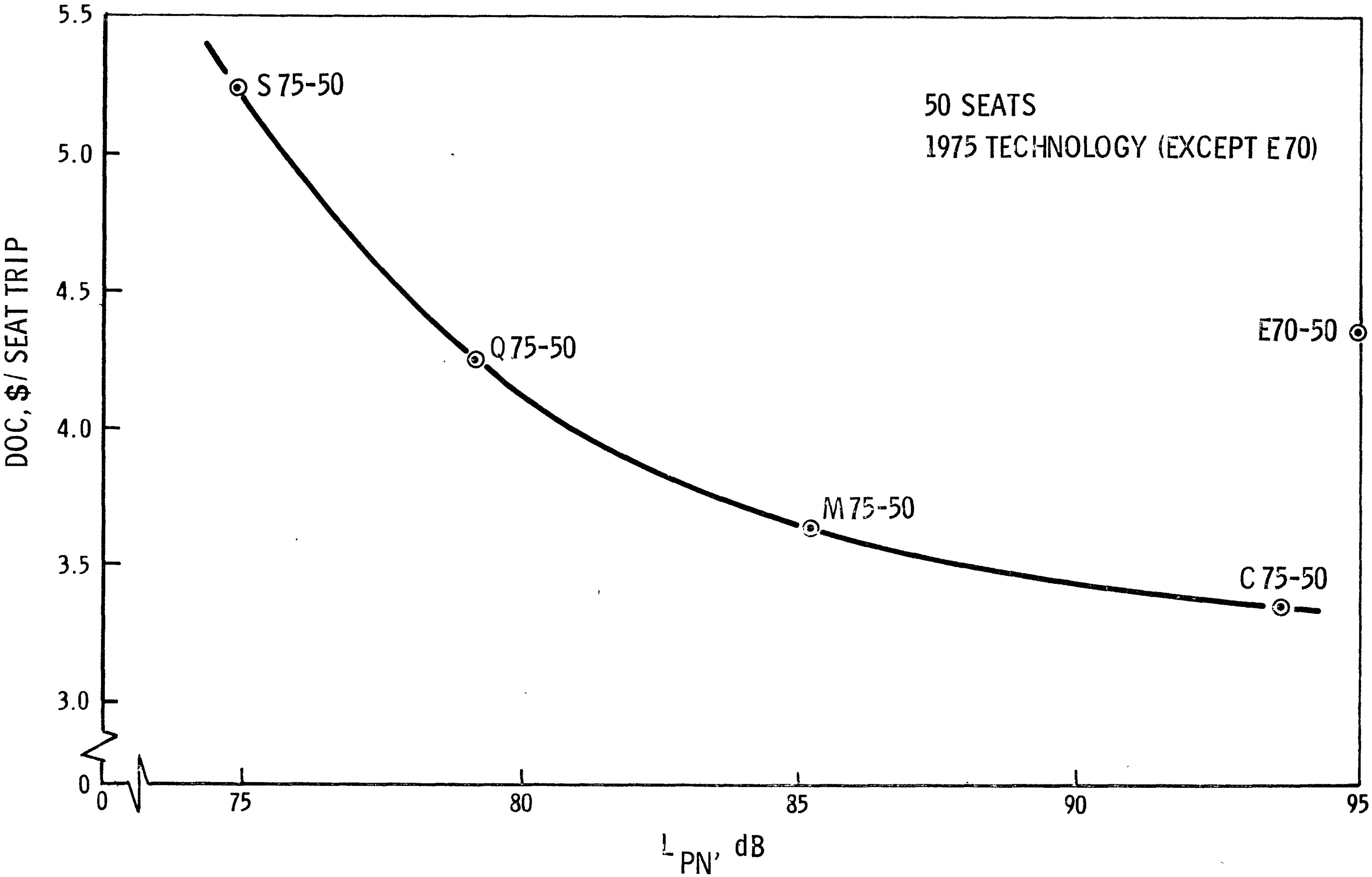

Fig. 4 DOC @ 100 mi. vs liftoff noise @ 500 ft for basic helicopters.

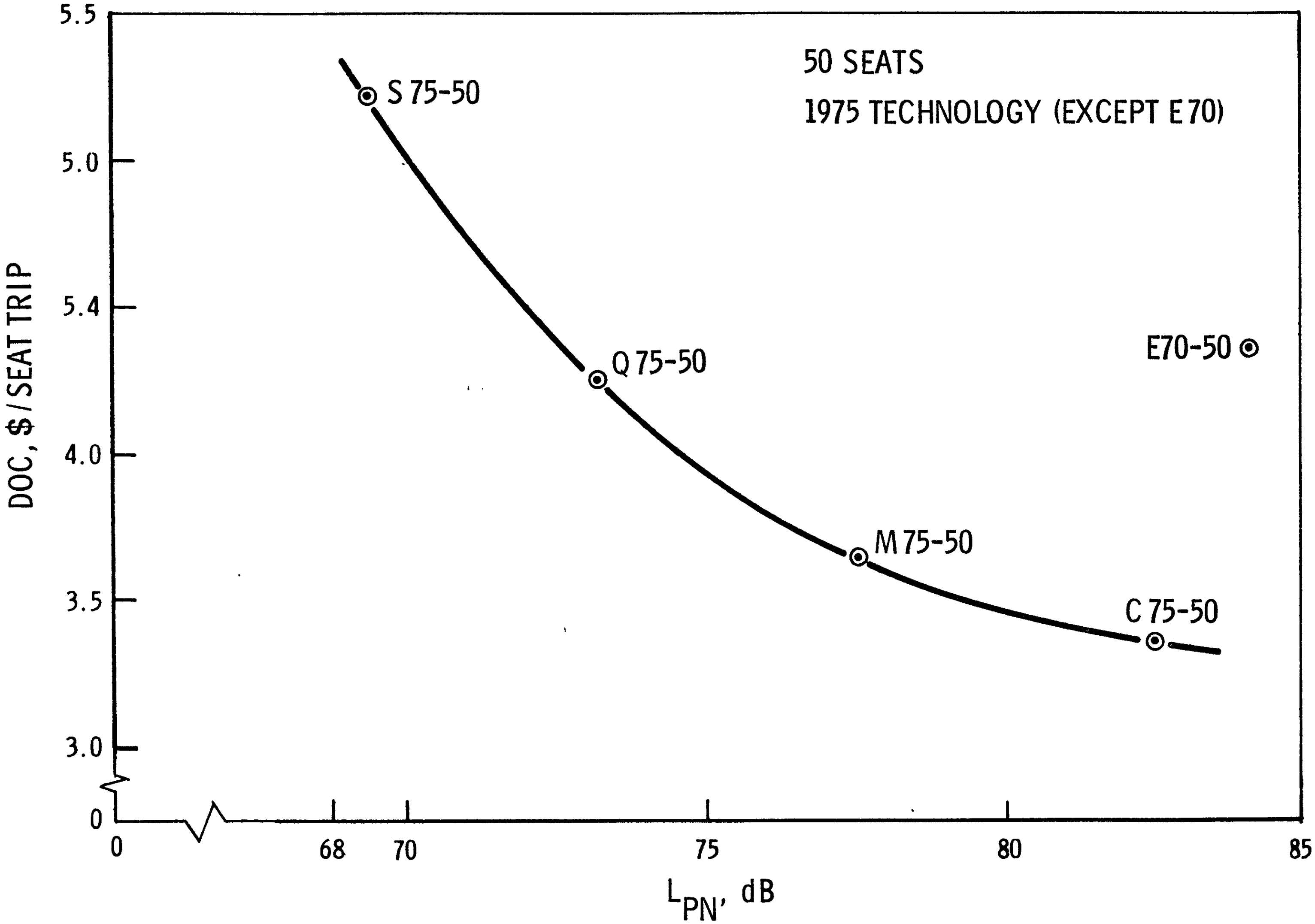

Fig. 5 DOC @ 100 mi. vs cruise noise @ 5000 ft altitude for basic helicopters.

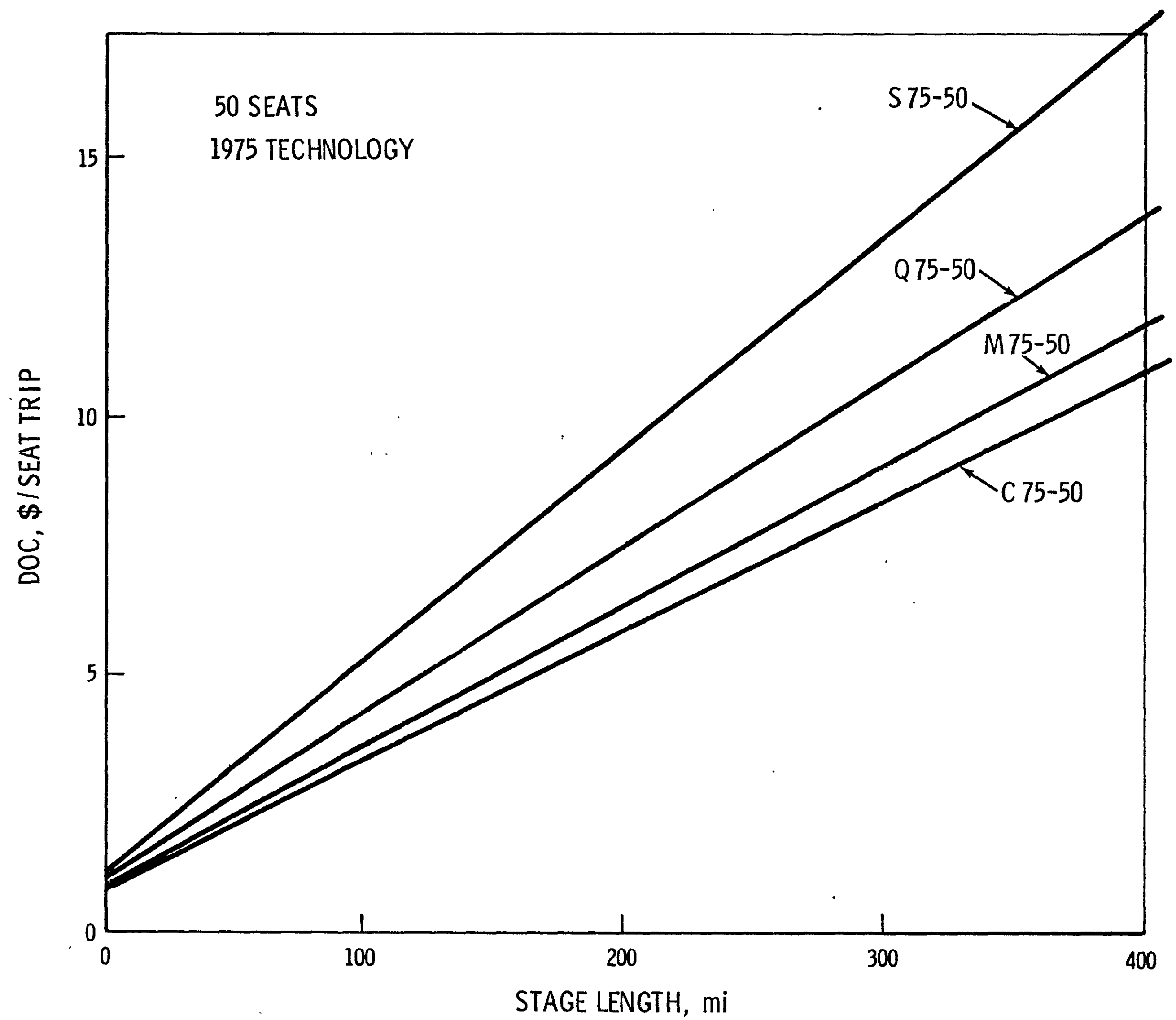

Fig. 6 DOC vs stage length for basic helicopters.

Therefore noise vs. time histories were found for three of the helicopters, as heard by observers at three distances from the liftoff point. These are shown in Figures 7, 8 and 9. The directivity function gives maximum noise along the rotor shaft axis, and minimum noise in the rotor plane. Thus the maximum noise occurs when the aircraft is overhead in all cases. The first dip in the noise vs. time curves is due to the decrease in thrust as the vehicle moves from vertical acceleration into steady state vertical climb. Noise increases again at the top of vertical climb because the rotor plane has moved further from the observer and thus the directivity function is stronger. Noise decreases again as the rotor plane is tilted toward the observer at the start of horizontal acceleration. Then, as the vehicle moves toward the overhead position, noise builds rapidly toward the peak.

To clarify the space-time relationships, the takeoff profiles for C75-50 and Q75-50 are plotted in Figure 10. Notice that C75-50 moves through its profile much more rapidly, but the acceleration phase (curved portion of the profile) takes up much more space. Q75 - 50 is 9 - 13 dB quieter than C75 - 50 at corresponding points in space, but Q75 - 50 reaches these points later in time. This is due to the lower gear box limited power for Q75 - 50.

The curves show that, for any given vehicle, the peak noise heard by each of the three observers is about the same. This is an indication that a greater height of vertical climb should be considered. This will be discussed further in section 3.5.

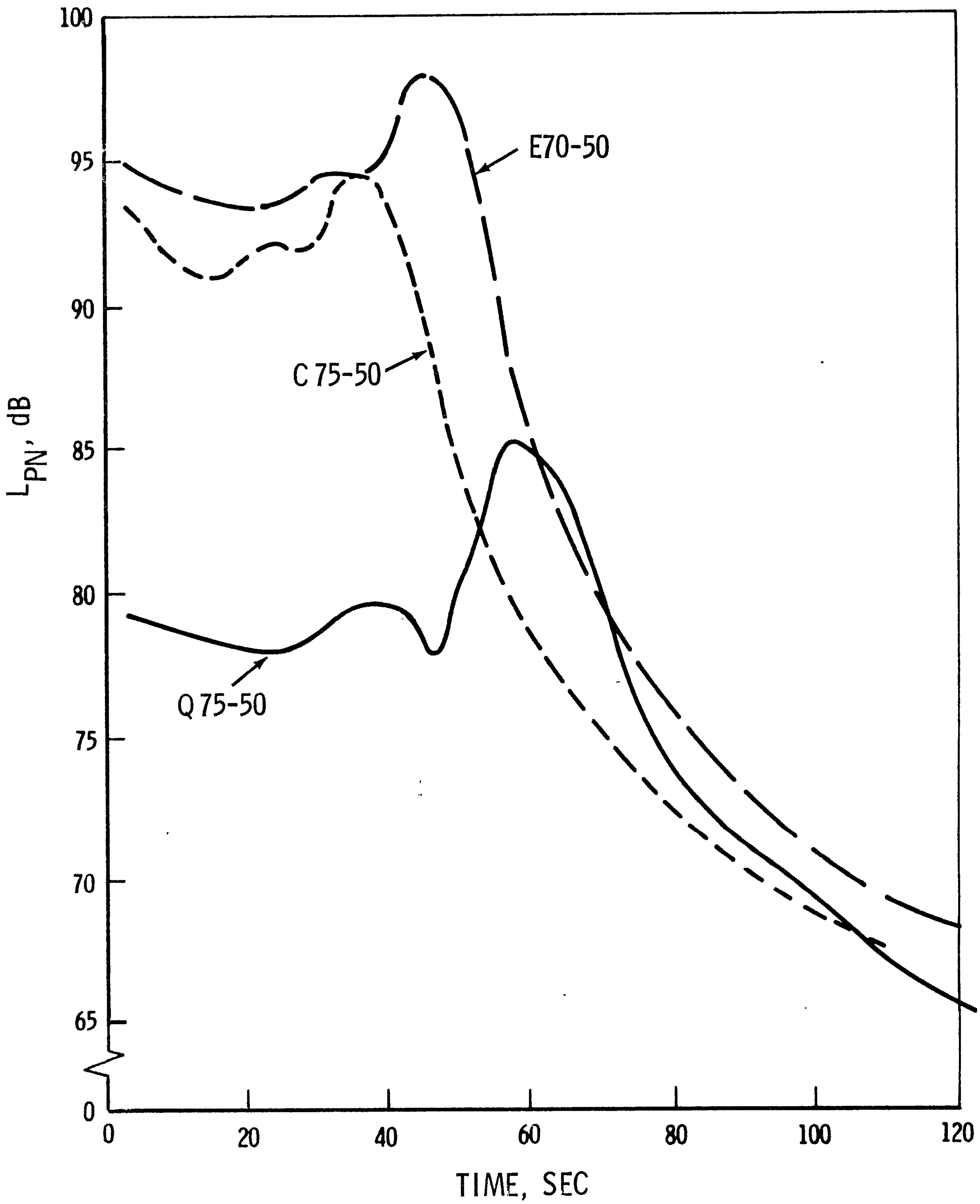

Fig. 7 Noise vs time from liftoff for 3 helicopters with observer at 500 ft.

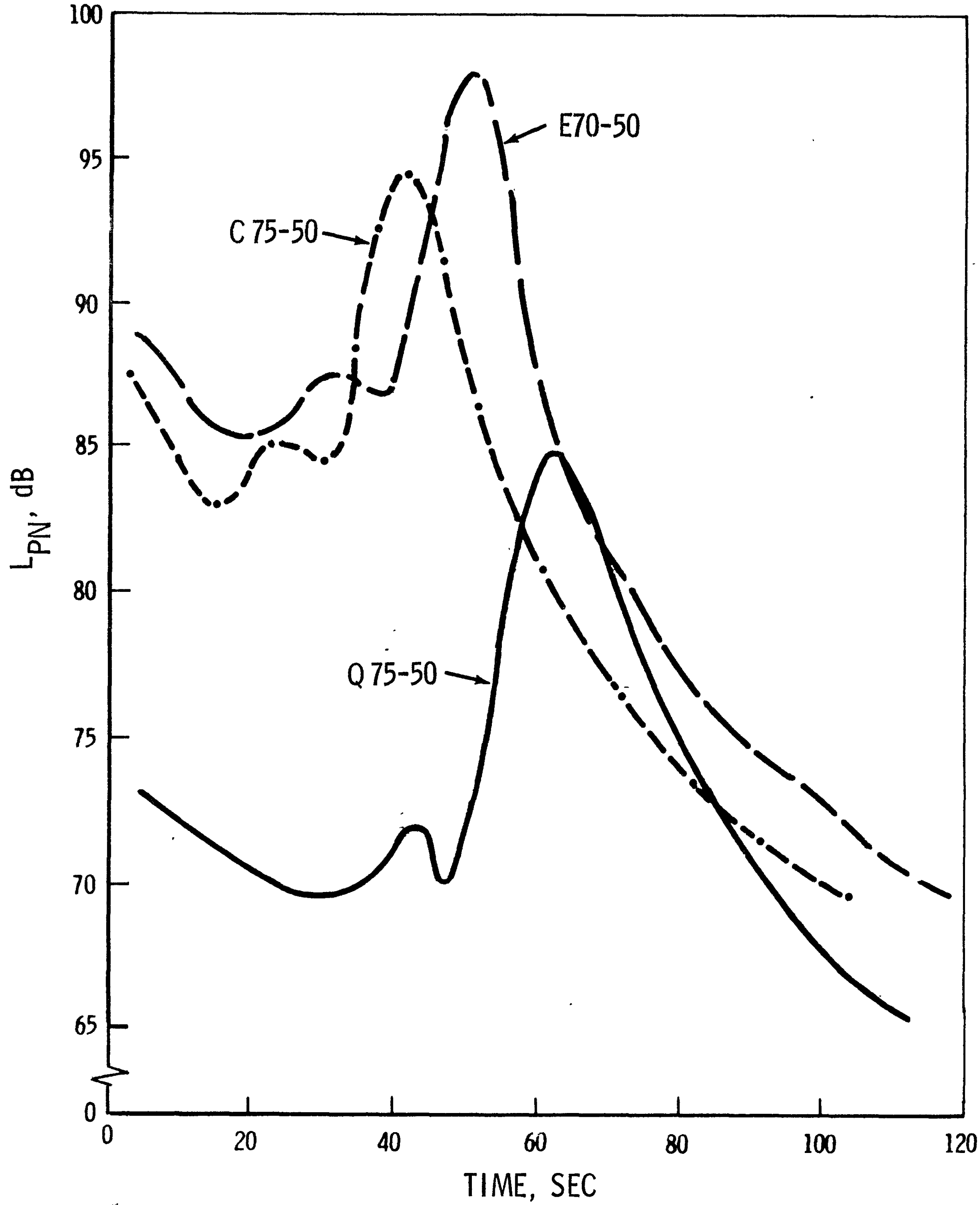

Fig. 8 Noise vs time from liftoff for 3 helicopters with observer at 1000 ft.

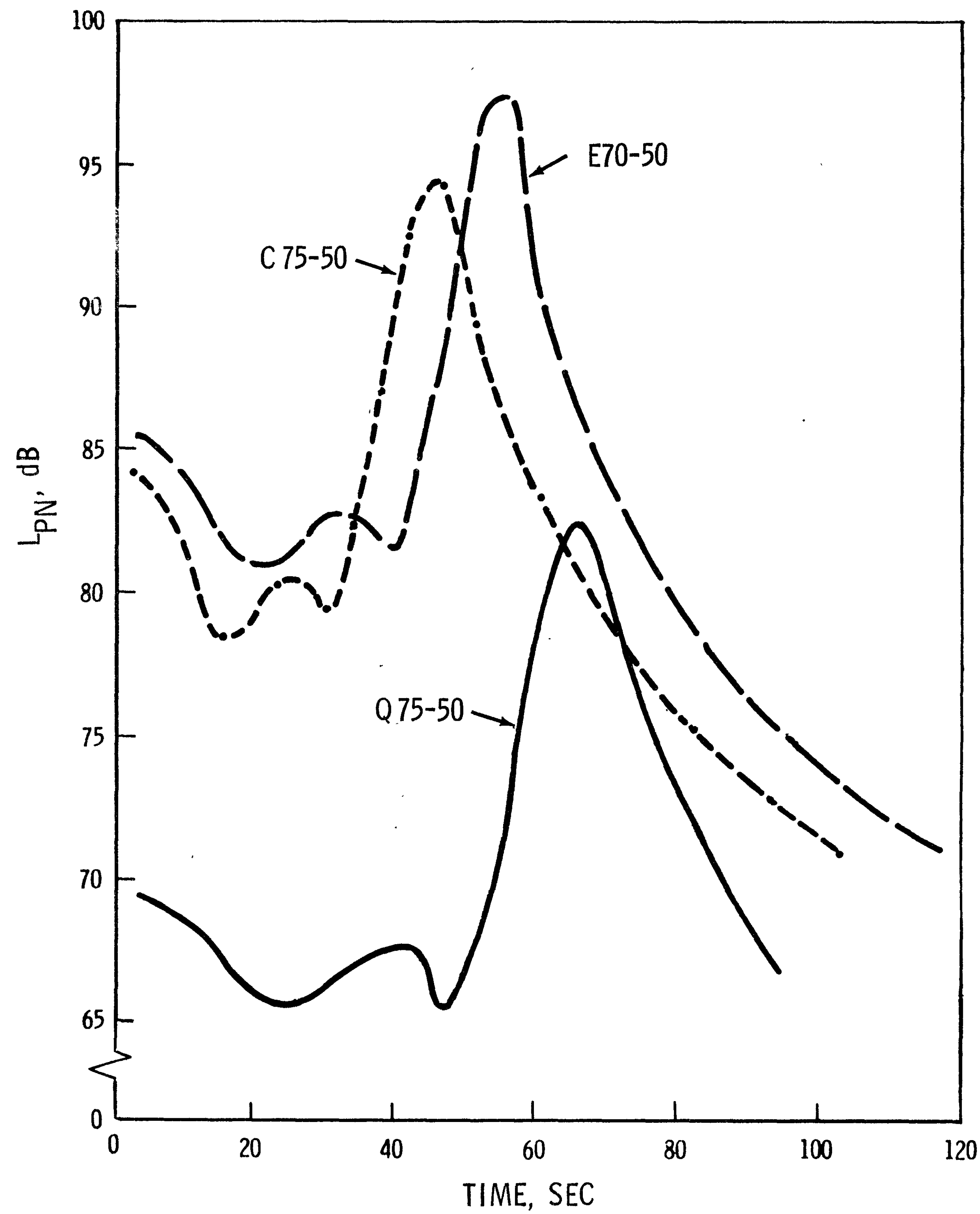

Fig. 9 Noise vs time from liftoff for 3 helicopters with observer at 1500 ft.

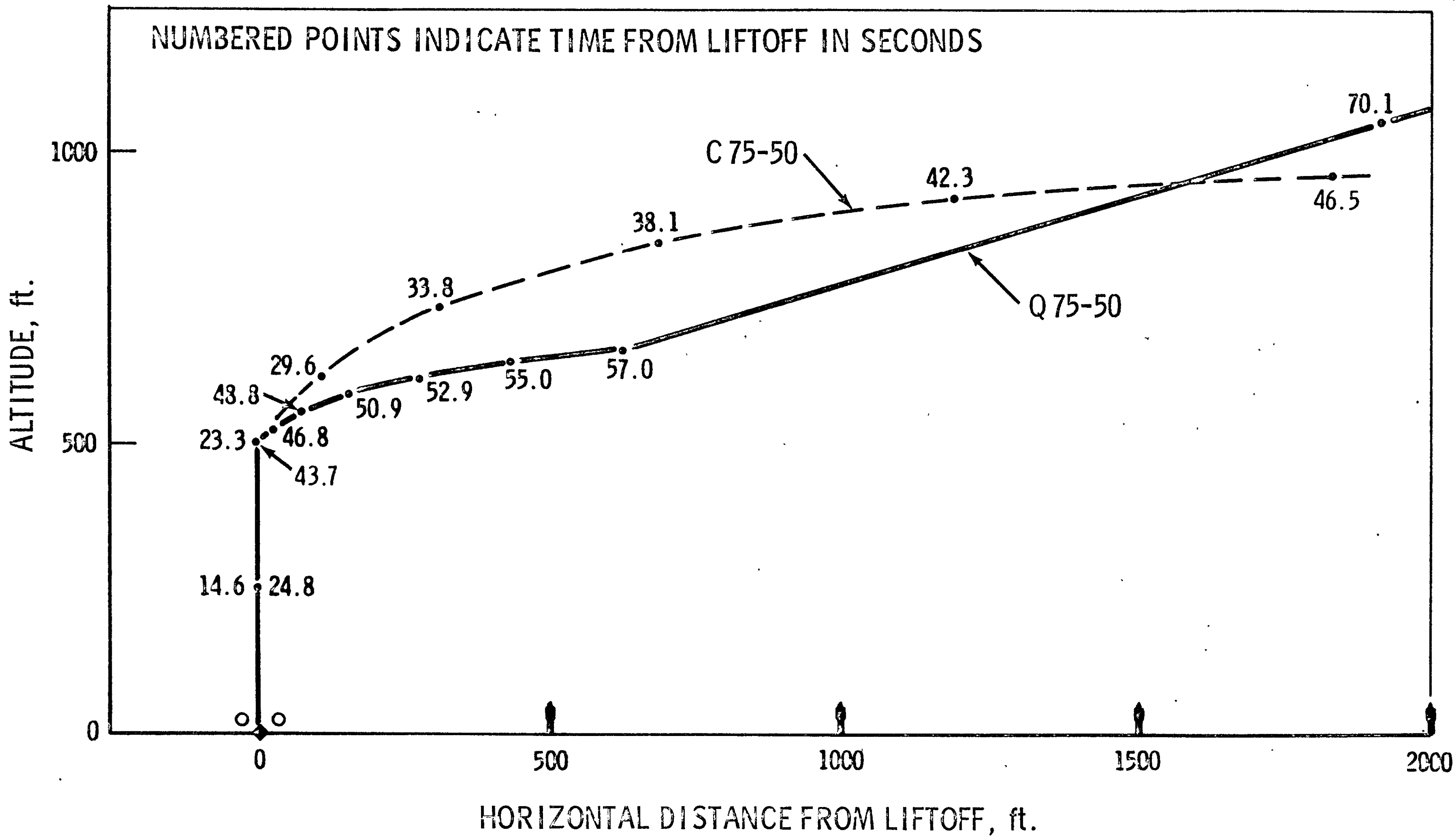

Fig. 10 Altitude vs distance from liftoff for 2 helicopters.

3.3 Size Variation

In the basic variation the payload was kept fixed at 50 seats. This was then extended by developing equivalent variations for 20, 80, and 110 seats.

Except for seats, the parameters in Table 3 were kept the same. Table 5 contains a portion of Table 4 which was used again for the variation of noise for a given size. Note that noise levels are not shown since a larger vehicle will be noisier, other parameters being kept constant. Keeping these parameters constant assumes that the optimal values for 50 seats are optimal for the other sizes as well.

The parameters shown in Table 6 were changed along with size. Fuselage planform outlines are shown in Figure 11. A large proportion of the planform area is devoted to the cabin. This is possible because these aircraft are assumed to be unpressurized and the short design range should allow lavatory and galley space to be small.

It was not within the scope of this work to make a detailed study of the range of sizes over which either the single main rotor or tandem configuration is optimal. However, it was felt that the single rotor configuration was superior for the 20 seat size and the tandem superior for the 80 and 110 seat sizes. Both configurations were considered for the 50 seat size. Within the accuracy of the design procedures used here, the tandem was very slightly, but not significantly, superior.

DOC vs. liftoff noise and DOC vs. cruise noise are plotted in Figures 12 and 13, respectively, for the various vehicle sizes. As expected, the curves have the same shape as the basic variation,

Table 5 : Parameters for Size and Time Frame Variations

	C	M	Q	S
DL, lb/ft^2	6.0	5.0	5.0	4.0
σ	0.087	0.110	0.193	0.251
V_{cr}, mph	237	219	196	168
μ	0.50	0.55	0.60	0.60
M_{at}	0.95	0.825	0.70	0.575
$(C_T/\sigma)_{cr}$	0.070	0.065	0.055	0.050
V_{tcr}, ft/sec	694	584	480	411
GBH	1.20	1.10	1.02	1.00
$(C_T/\sigma)_h$	0.063	0.100	0.100	0.100
V_{th}, ft/sec	680	437	330	270

Table 6 : Parameters Varied with Size

	Seats			
	20	50	80	110
Flight Crew	2	2	2	3
Stewardesses	1	2	2	3
Fuselage Length, ft.	37.6	59.8	73.0	86.2
Fuselage Diameter, ft.	7.8	9.4	11.0	12.6
Seats Abreast	3	4	5	6
Doors	1	2	3	4
Payload Weight, lb.	4200	10,400	16,400	22,600
Furnishings Weight	1800	3650	5240	6880
Avionics and Instruments Wt.	700	840	900	1000
Main Rotors	1	2	2	2
Number of Engines	2	3	3	3

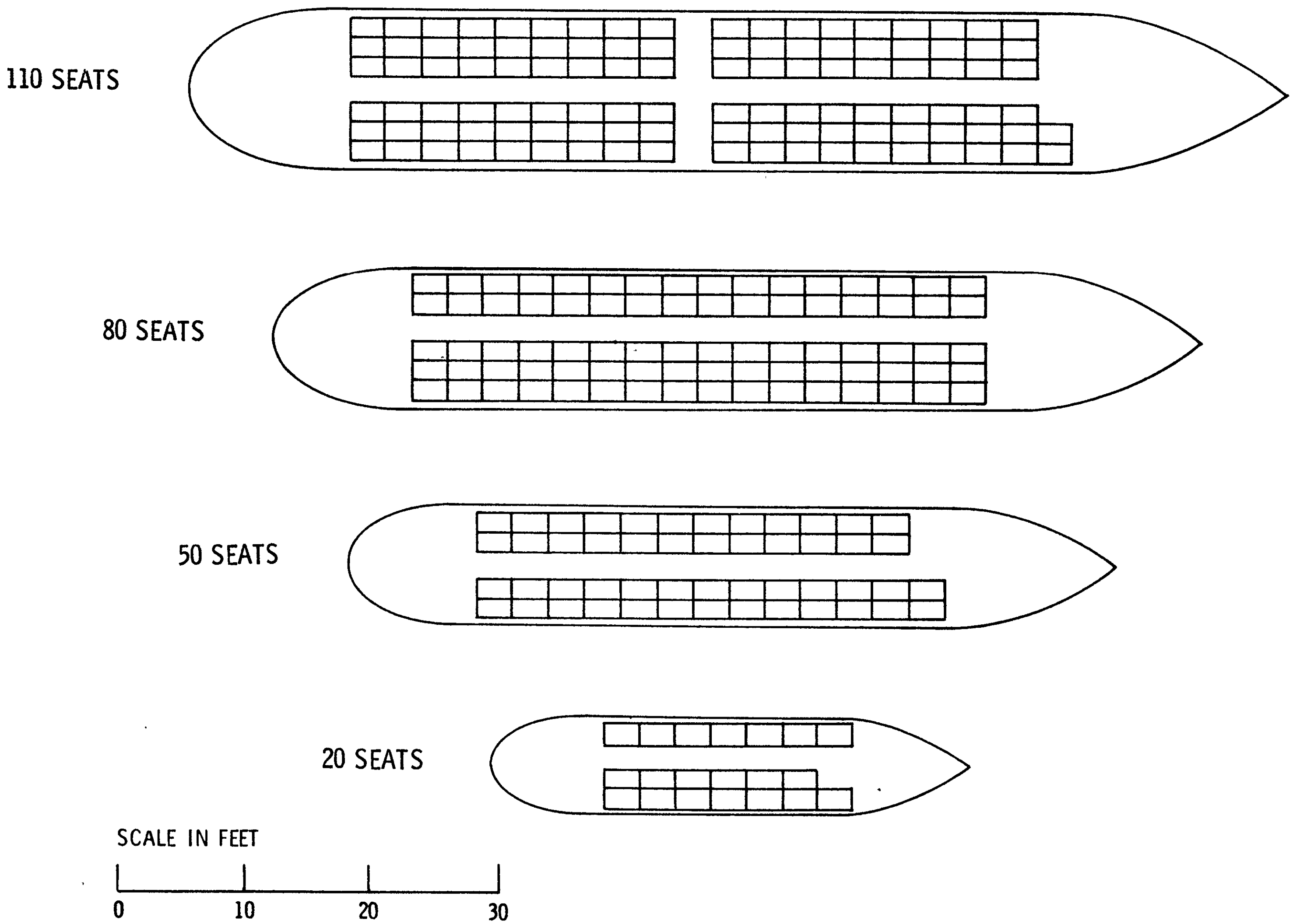

Fig. 11 Fuselage planform layouts.

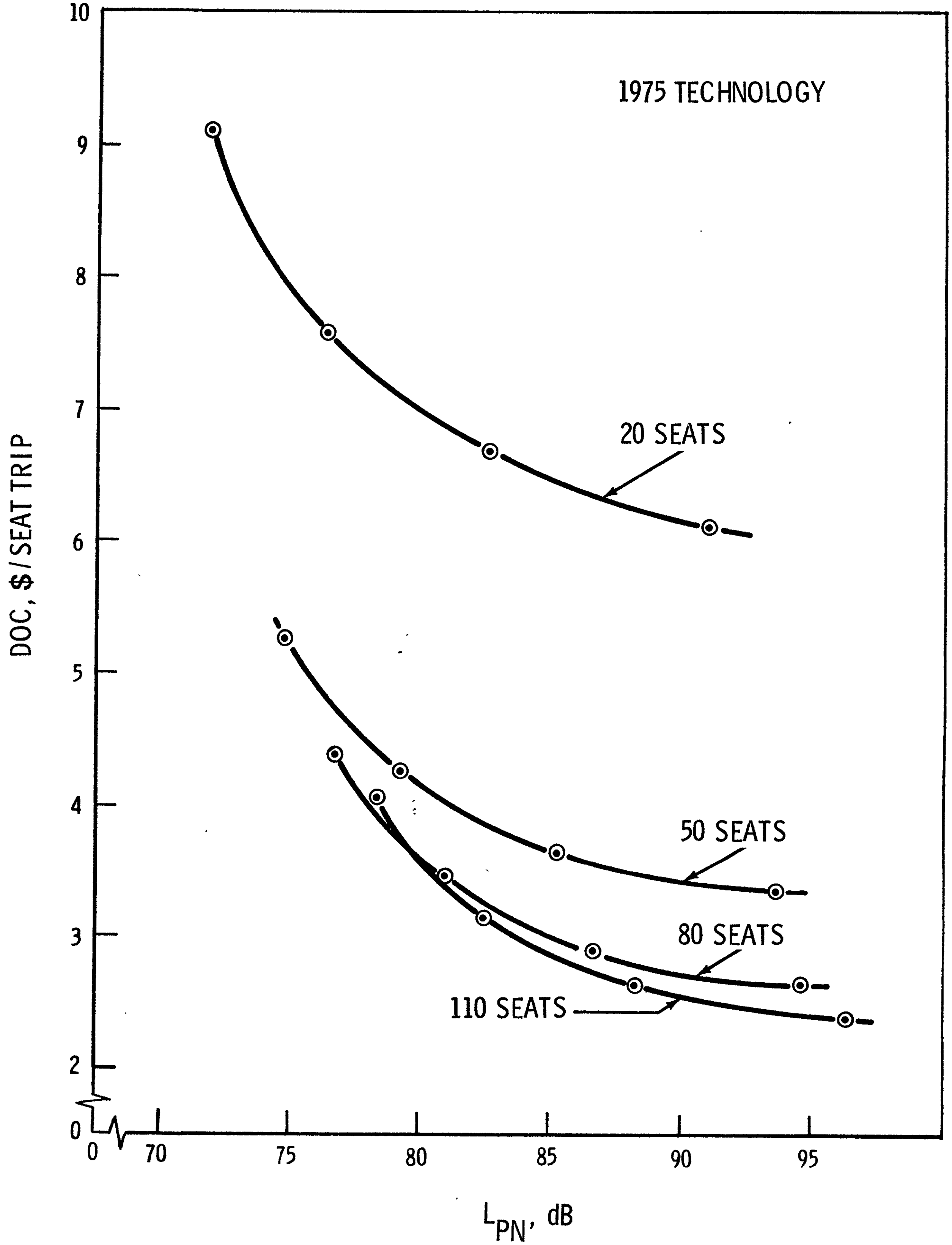

Fig. 12 DOC @ 100 mi. vs liftoff noise @ 500 ft for varying size.

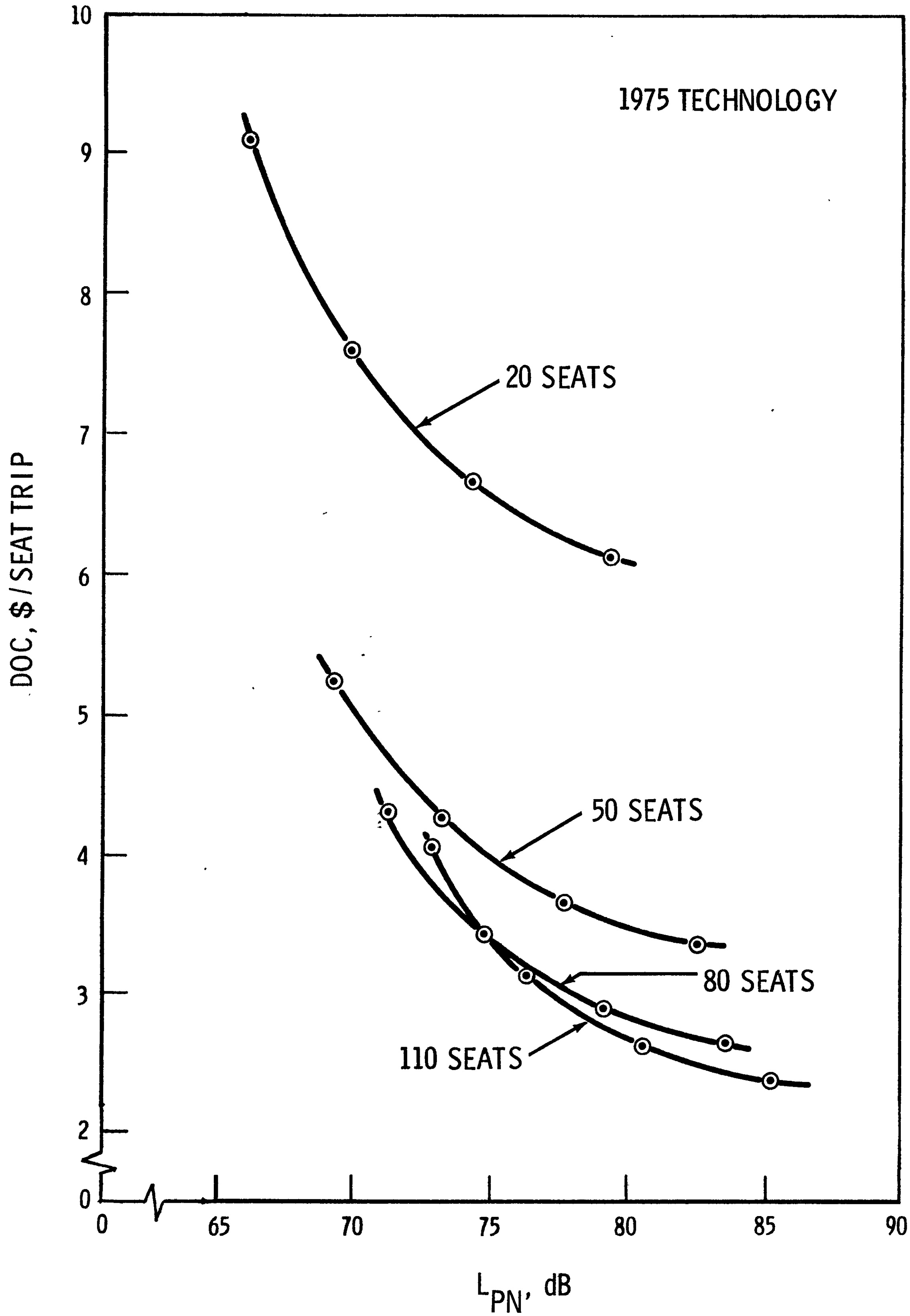

Fig. 13 DOC @ 100 mi. vs cruise noise @ 5000 ft altitude for varying size.

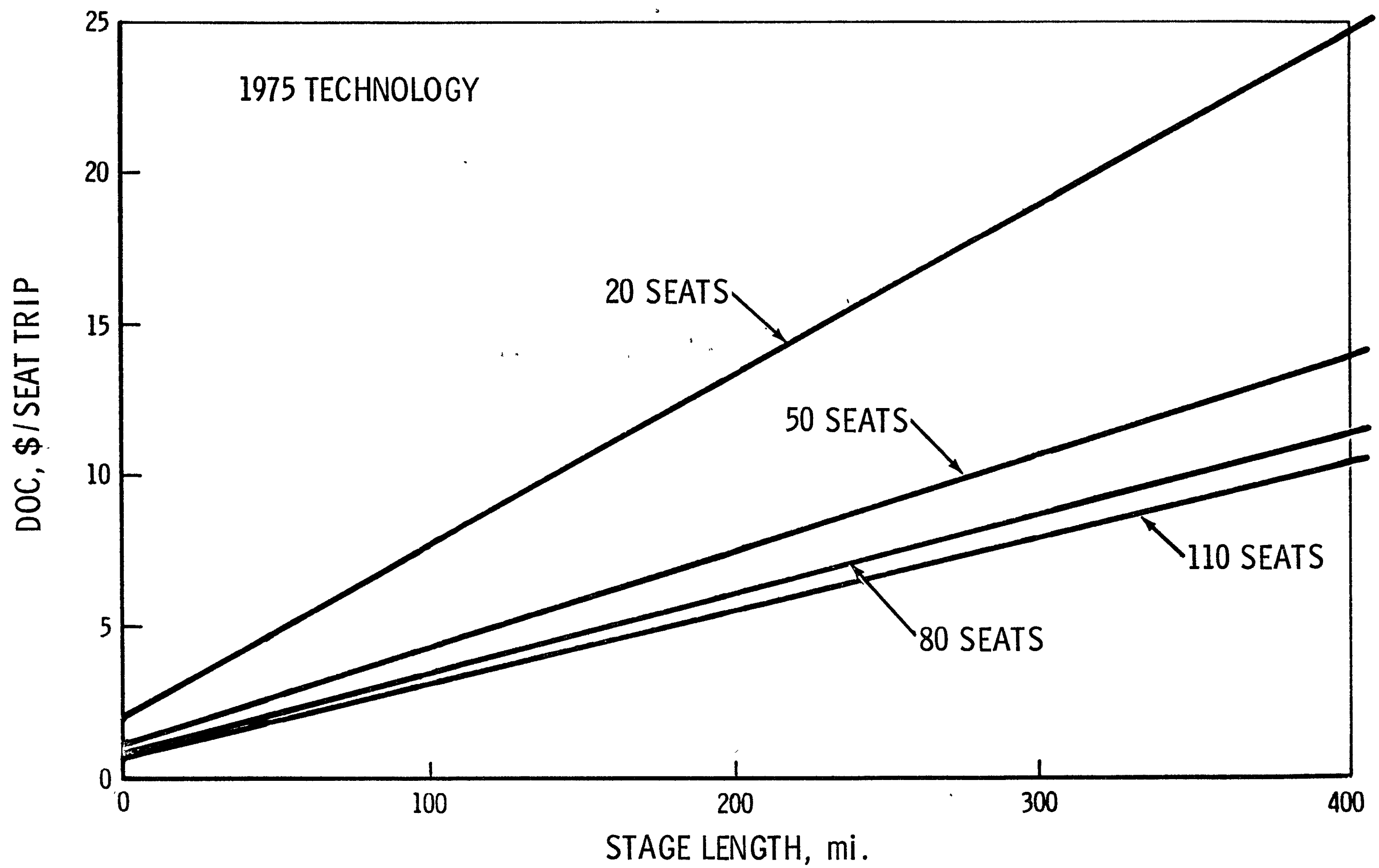

Fig. 14 DOC vs stage length for Q series helicopters and varying size.

with the curves for larger vehicles being lower. However, a very significant result is indicated by the crossing of the 80 seat curve and the 110 seat curve in both figures. It is a generally accepted rule in transport aircraft economics that a larger vehicle will have a lower direct operating cost per seat. These curves show that, if certain noise objectives are to be met, this is no longer true; there is an optimum aircraft size based on DOC alone.

Again DOC at 100 miles stage length was **chosen as representative.** The DOC is plotted vs. stage length in Figure 14 for the Q series vehicles. These vehicles are represented by the second point from the left on each of the DOC vs. noise curves. DOC vs. stage length curves for C, M, and S series vehicles are very similar.

The terminal noise vs. time history, for a given vehicle series, moves upward slightly for larger size, but does not change shape.

3.4 Time Frame Variation

Holding the size fixed at 50 seats, the basic variation was extended along another dimension, the technology timeframe. Variations equivalent to the basic variation, of 1975 time frame, were developed for time frames of 1970, 1980 and 1985. As mentioned earlier, the time frame is the year in which a production prototype could be flying, using the latest technology both in design and manufacturing.

The parameters in Table 3, except time frame, were kept the same as the basic variation. The **parameters of Table 5 were used** again for the variation of noise for a given time frame. As with the size variation, it is assumed that these parameters remain optimal, in this case for different time frames.

The parameters that are varied with time frame are shown in

Table 7 : Parameters Varied With Time Frame

	Time Frame				
	E70-50	1970	1975	1980	1985
Fuselage Drag Factor	3.4	3.0	2.5	2.0	1.8
Hub and Pylon Drag Factor	0.0310	0.0250	0.0225	0.0200	0.0190
Engine Power/Weight	5.0	-	-	-	-
C series	-	5.0	7.0	9.0	10.0
M series	-	4.5	6.5	8.5	9.5
Q series	-	4.0	6.0	8.0	9.0
S series	-	3.5	5.5	7.5	8.5
Specific Fuel Consumption	0.52	0.52	0.43	0.40	0.37
Rotor Weight Factor	1.20	1.05	0.90	0.80	0.70
Drive System Weight Factor	0.82	0.80	0.70	0.60	0.50
Fuselage Weight Factor	1.10	1.00	0.95	0.90	0.85

Note : The drag and weight factors multiply the appropriate formulae.

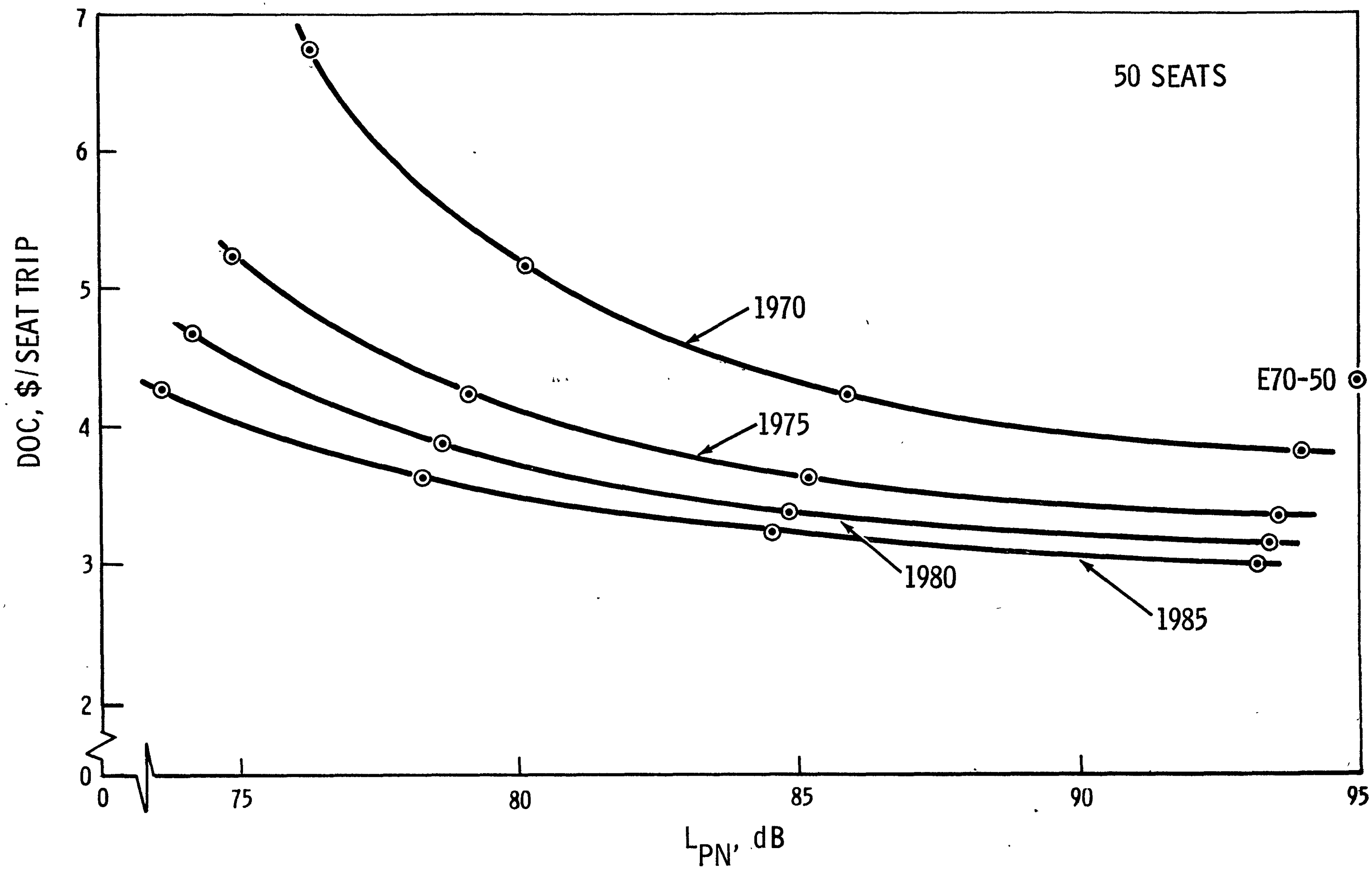

Fig. 15 DOC @ 100 mi. vs liftoff noise @ 500 ft for varying time frame.

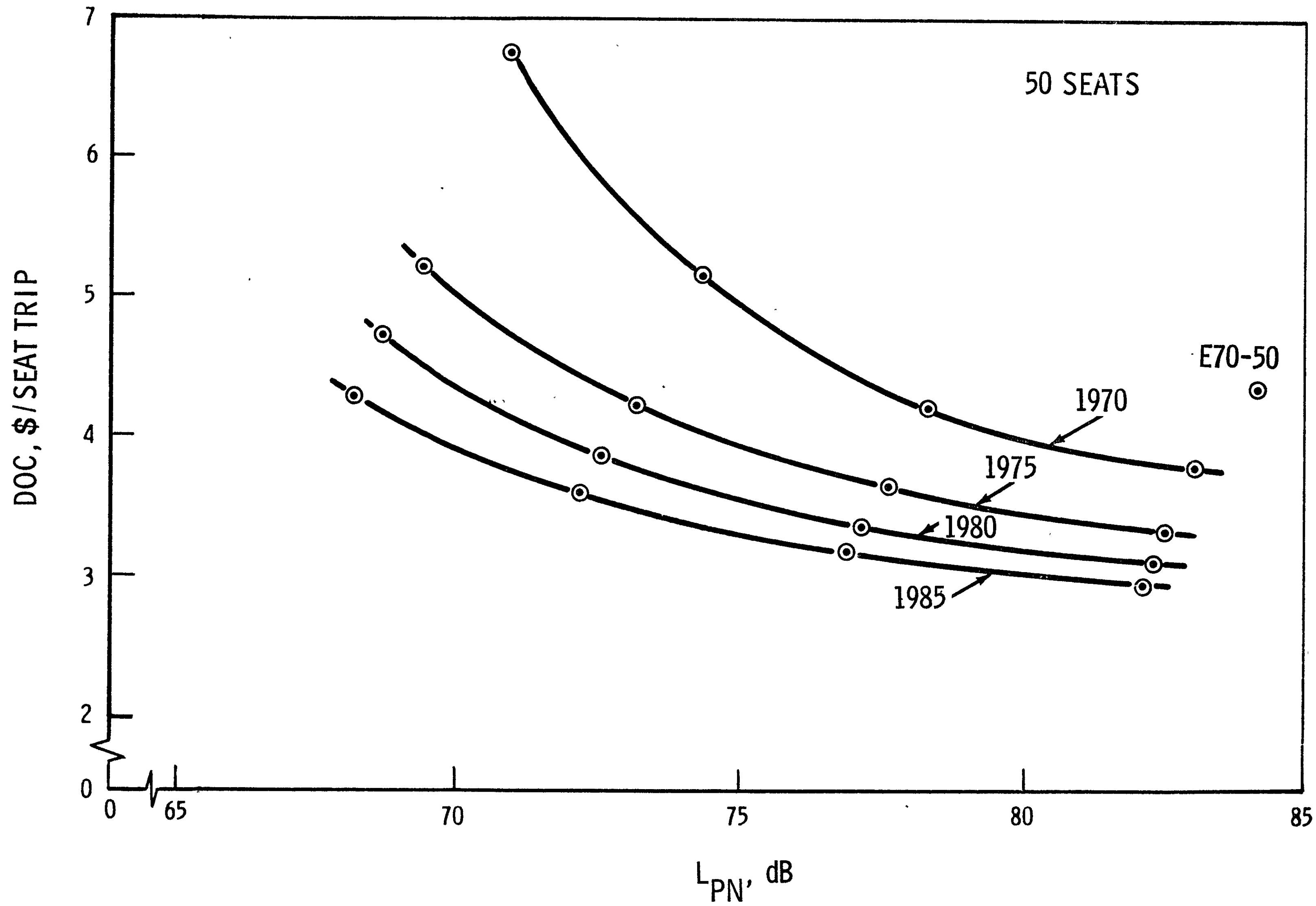

Fig. 16 DOC @ 100 mi. vs cruise noise @ 5000 ft altitude for varying time frame.

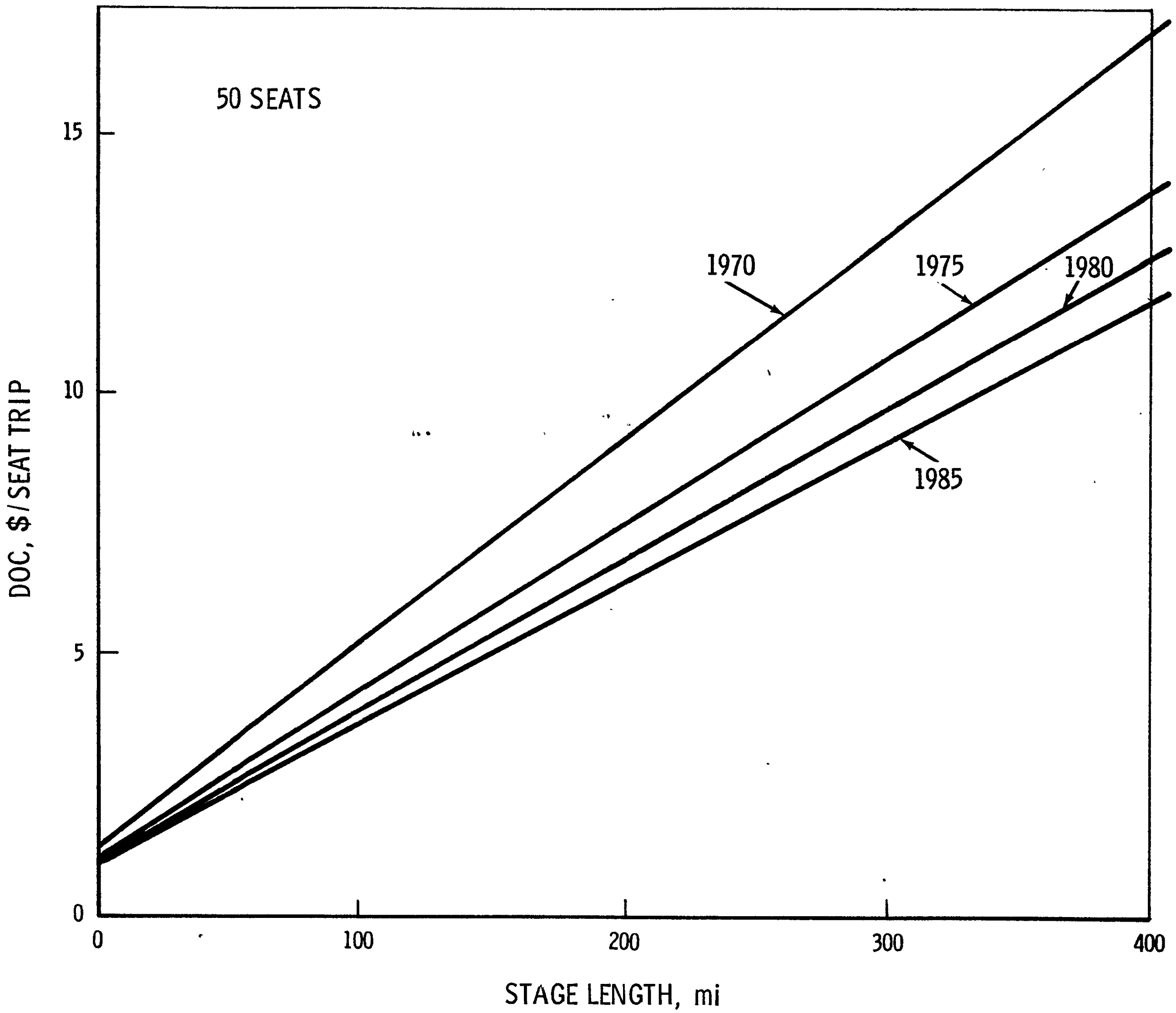

Fig. 17 DOC vs stage length for Q series helicopters and varying time frame.

Table 7. The drag and weight factors used in E70 - 50, to simulate the Vertol 347, are somewhat higher than for 1970 time frame. While the 347 prototype did make its first **flight in** 1970, it does not represent the degree of advance that could have been achieved by a complete design and development effort. The parameters in Table 7 were derived by using engineering judgement and knowledge of specific projected technological developments to extrapolate historical trends.

DOC vs. liftoff noise and DOC vs. cruise noise are plotted in Figures 15 and 16 respectively. E70 - 50 is shown for added perspective. Again the curves have the same shape as the basic variation with the curves for later vehicles falling lower. It is interesting to note that the quietest 1985 vehicle costs very little more than the noisiest 1970 vehicle. In other words, the technology improvements can offset the penalties of a moderate pace of noise reduction.

Again DOC at 100 miles stage length was chosen as representative. The DOC is plotted vs. stage length in Figure 17 for the Q series vehicles. These vehicles are represented by the second point from the left on each of the DOC vs. noise curves. DOC vs. stage length curves for C, M, and S series vehicles are very similar.

The terminal noise vs. time history, for a given vehicle series, moves downward slightly for later time frames, but does not change shape.

3.5 Path Variation

This section considers variations on the design mission assumed for all the previous variations (See Table 3). It should be remembered that here each change in a path parameter represents a new vehicle designed for that mission, not the same vehicle flying a different path. The path variations here are based on the Q75 - 50 vehicle and the parameters under Q in Table 5 are used. Thus Q75 - 50 designates a family of different vehicles in this section.

Consider terminal path variations first. We can vary height of vertical climb and descent, which here are always equal. Noise vs. time histories for various heights of vertical climb are plotted in Figure 18. For a 1500 foot climb, the peak noise is still definitely overhead, but it is significantly reduced. The longer period of lower intensity noise prior to the peak may contribute to annoyance, however. DOC′ is plotted vs. stage length for the same vertical climb variation in Figure 19. The higher vertical climbs cost very little for stage lengths of greater than 50 miles.

The maximum allowable acceleration is another terminal path parameter which may be changed. As discussed in section 2.2 (c), reducing the maximum acceleration has the effect of tilting the flight path upward in the acceleration phase. This can be seen in Figure 20 where takeoff profiles are plotted for two extremes of allowble acceleration. Noise vs. time histories for these two values and an intermediate one are shown in Figure 21. Reducing allowable acceleration reduces the peak noise and causes the peak to occur slightly later. The change in DOC over this range of allowable accelerations is negligible.

Another terminal path parameter is the hover gearbox factor, GBH, which determines drive system limited power at hover rpm (low speeds). Increasing GBH has the effect of speeding up the takeoff procedure. The noise vs. time history is shifted to the left with very little change in shape or height of the peak.

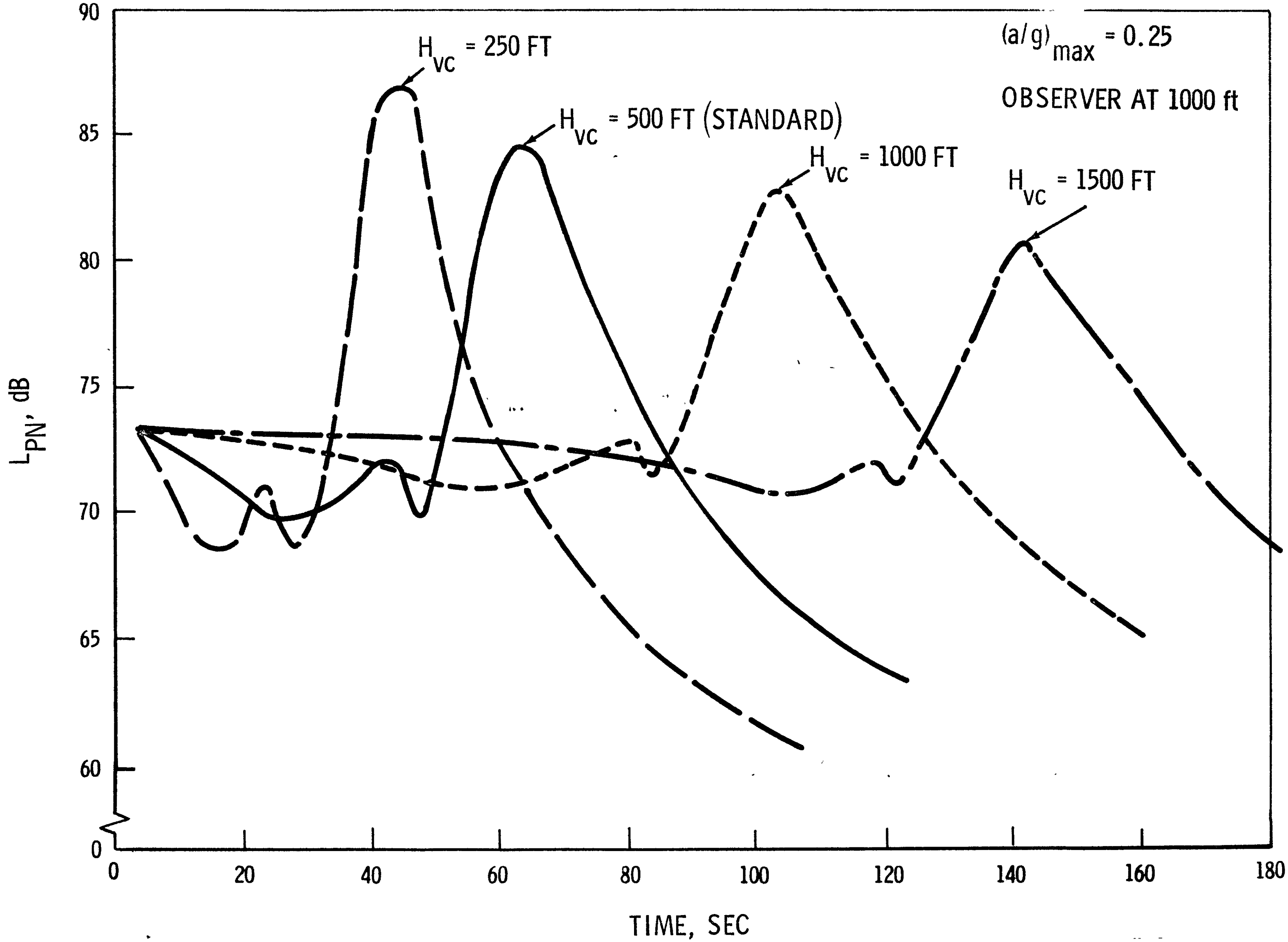

Fig. 18 Noise vs time from liftoff for Q75-50 helicopter and varying H_{vc}.

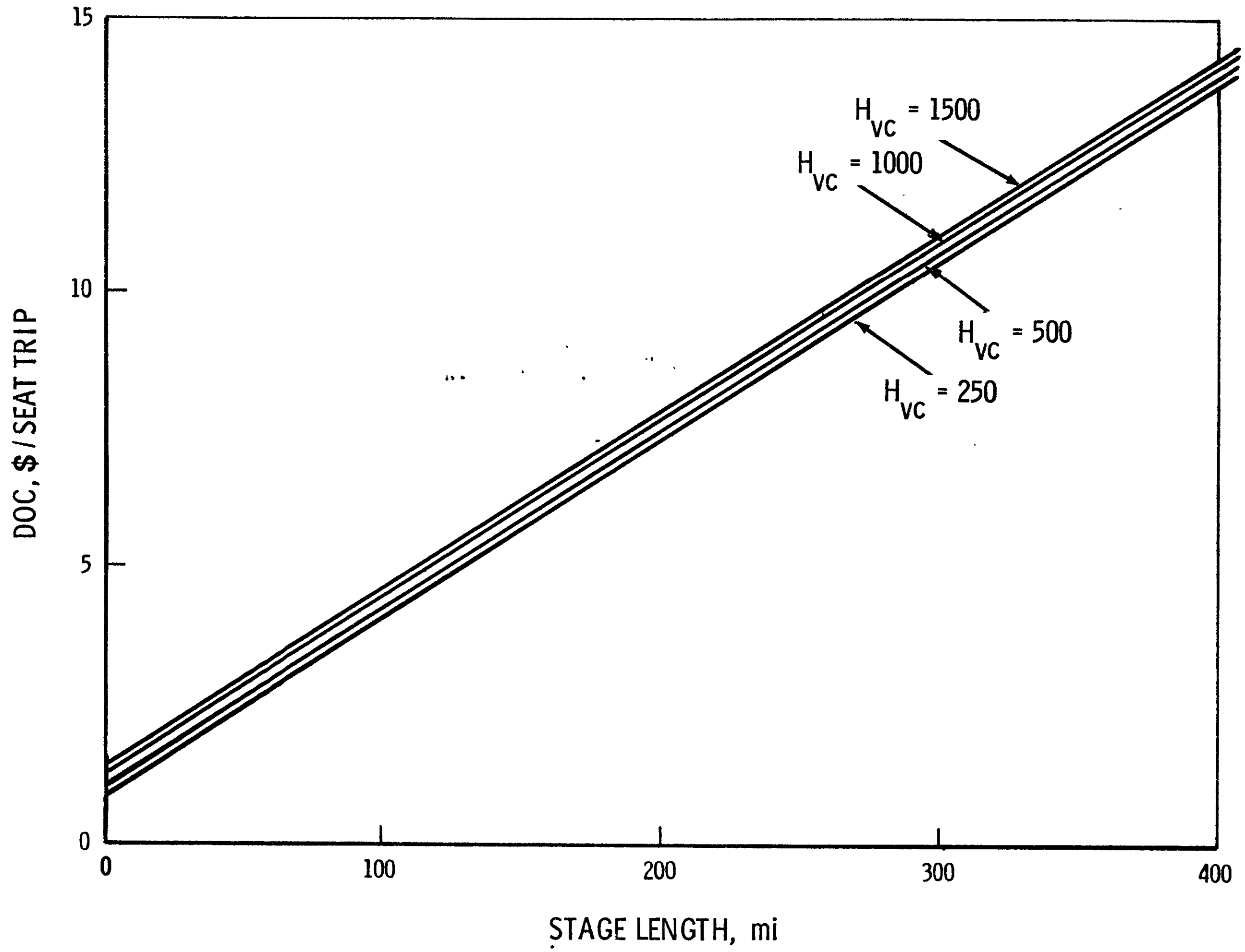

Fig. 19 DOC vs stage length for varying H_{vc} for Q75-50 helicopter.

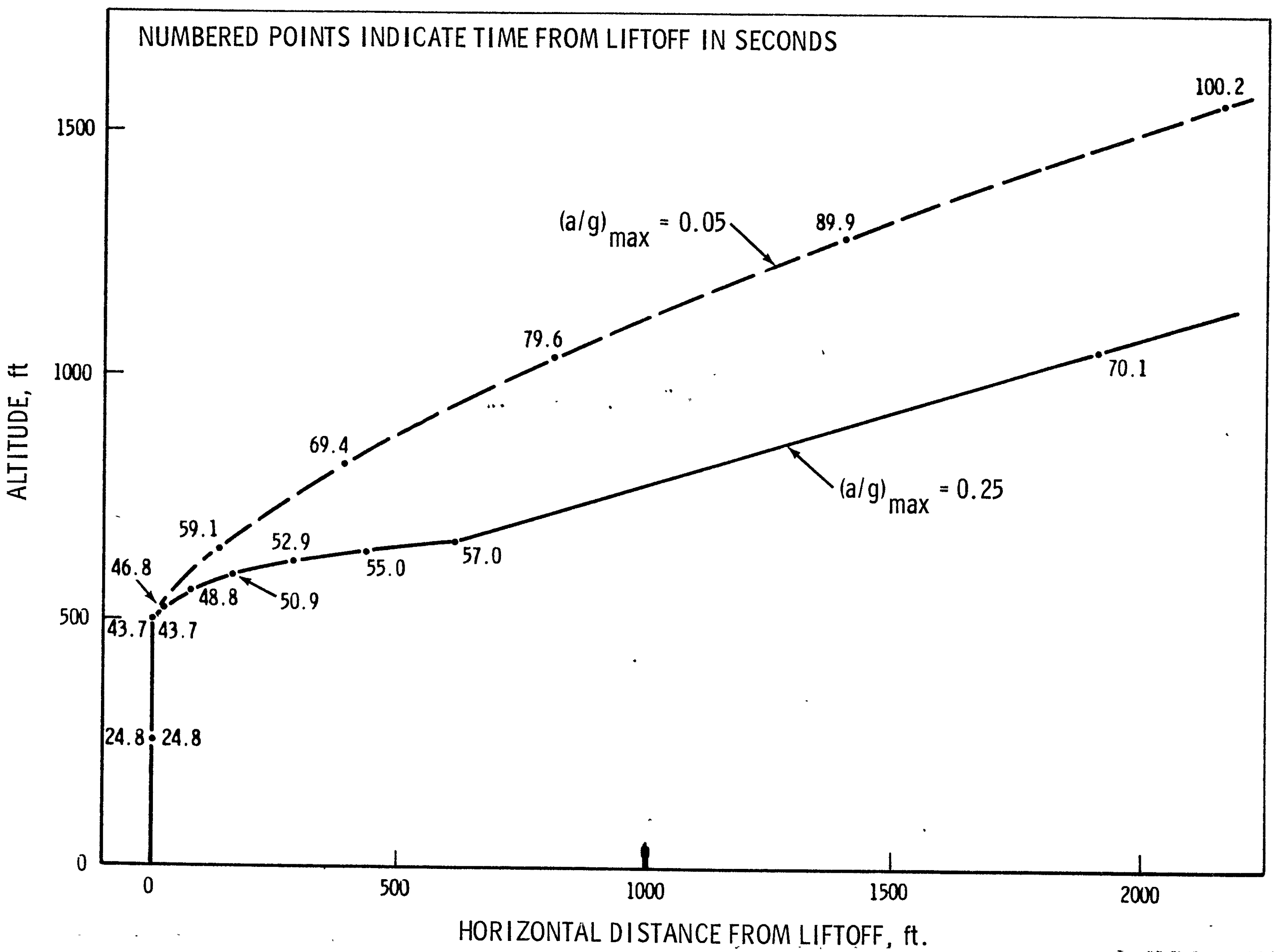

Fig. 20 Altitude vs horizontal distance from liftoff for Q75-50 helicopter and two values of $(a/g)_{max}$.

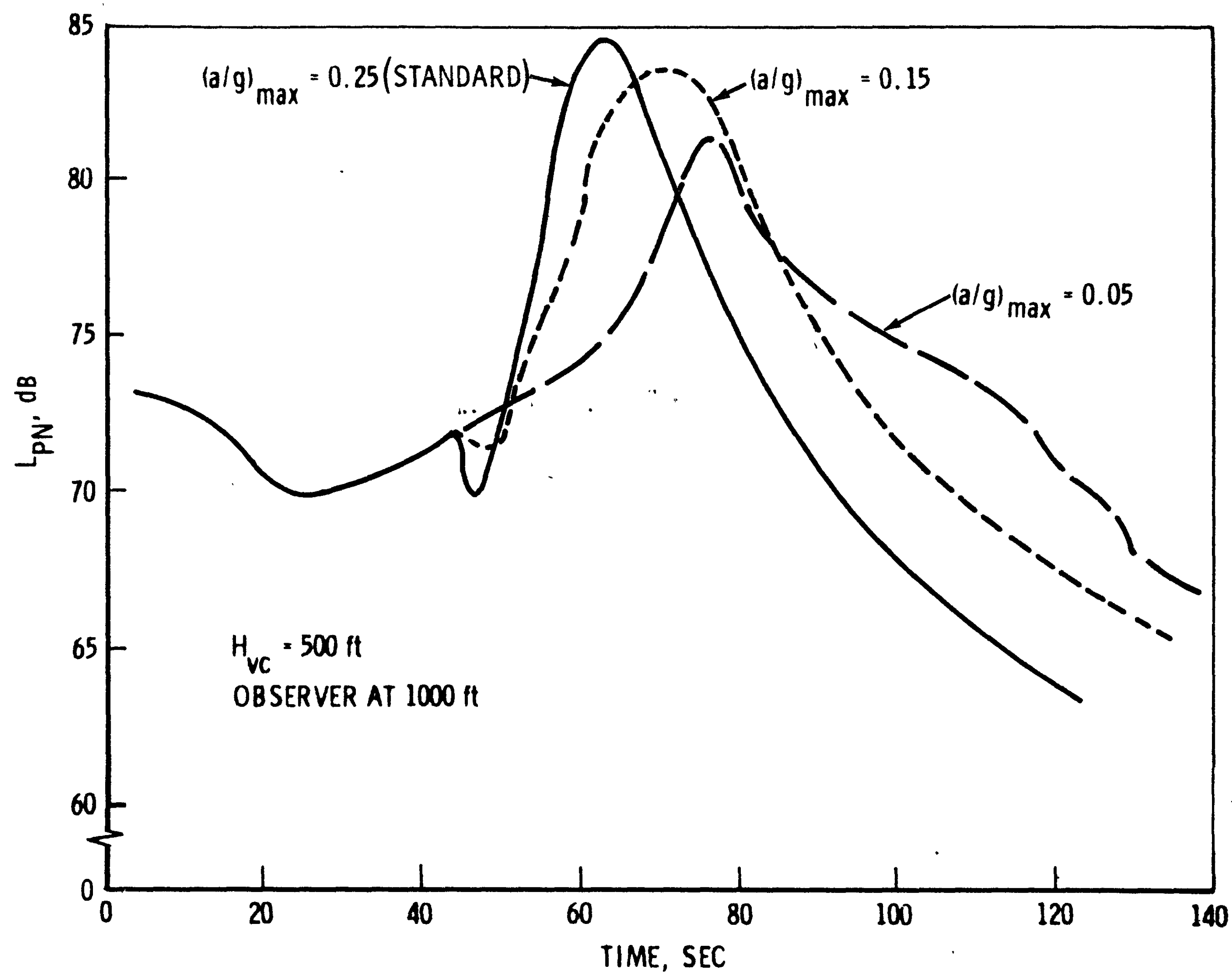

Fig. 21 Noise vs time from liftoff for Q75-50 helicopter and varying $(a/g)_{max}$.

DOC increases very slightly with increasing GBH. Note that the optimum GBH is somewhat higher for noisier vehicles (See Table 4).

It can be seen that the terminal noise vs. time history can be shifted around to a considerable extent. The problem of optimizing the terminal path cannot be pursued further at present because of the lack of a generally applicable method of condensing the noise vs. time history into a single measure of annoyance.

Now consider cruise path variations. A variation of design range was not considered necessary since it would yield results similar to a small size variation. This leaves cruise altitude, and peak flyover noise (cruise noise) is plotted vs. cruise altitude in Figure 22. DOC vs. stage length for various cruise altitudes is shown in Figure 23. Cruising higher than 5000 feet reduces flyover noise appreciably while increasing DOC only slightly. However, structural penalties for pressurization have not been taken into account here. Hence the DOC for higher cruise altitudes is optimistic.

It is well known (References 6 and 7) that the noise directivity pattern sweeps forward as speed increases, so that the maximum noise occurs in front of the helicopter rather than below. However, there is not sufficient data upon which to base a reliable empirical formula. Further study of helicopter cruise noise cannot be accomplished until new experimental data are obtained.

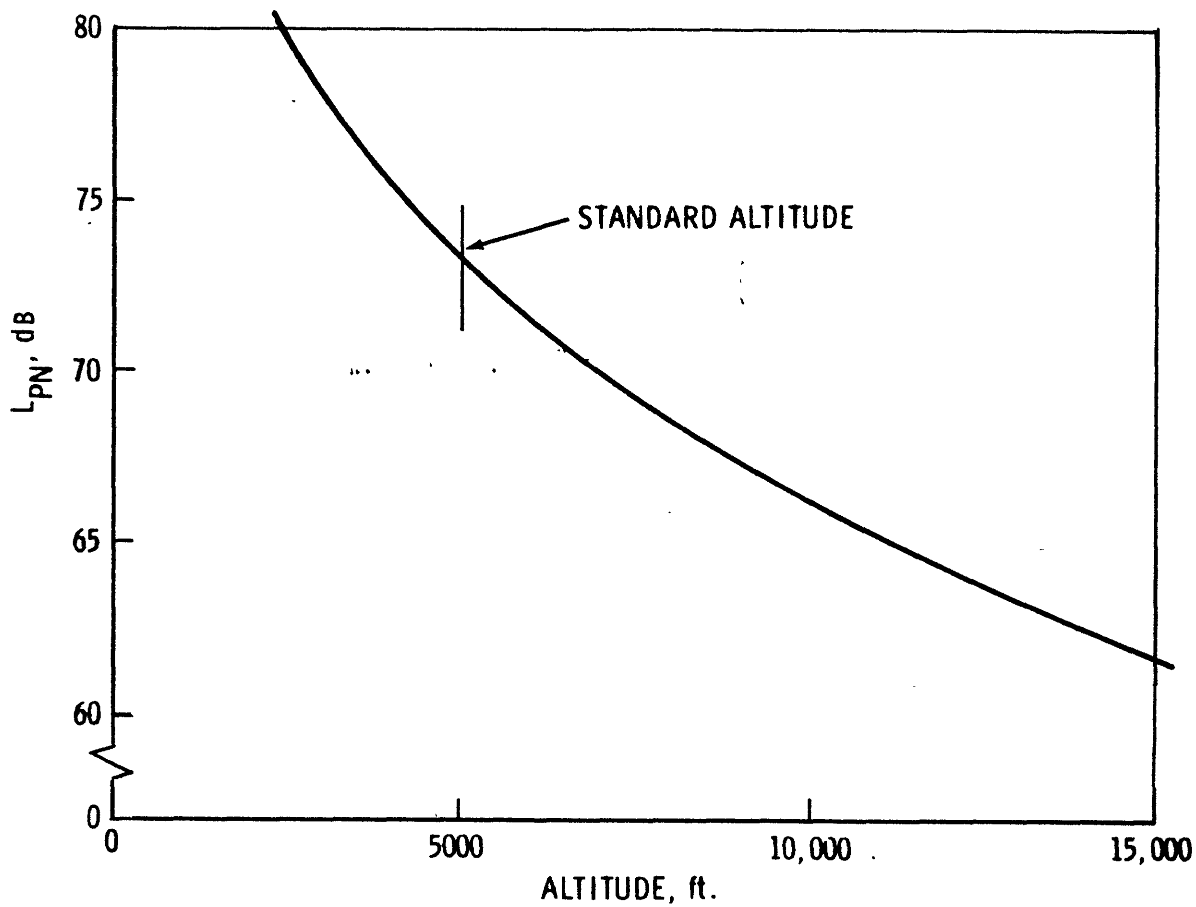

Fig. 22 Peak flyover noise vs cruise altitude for Q75-50 helicopter.

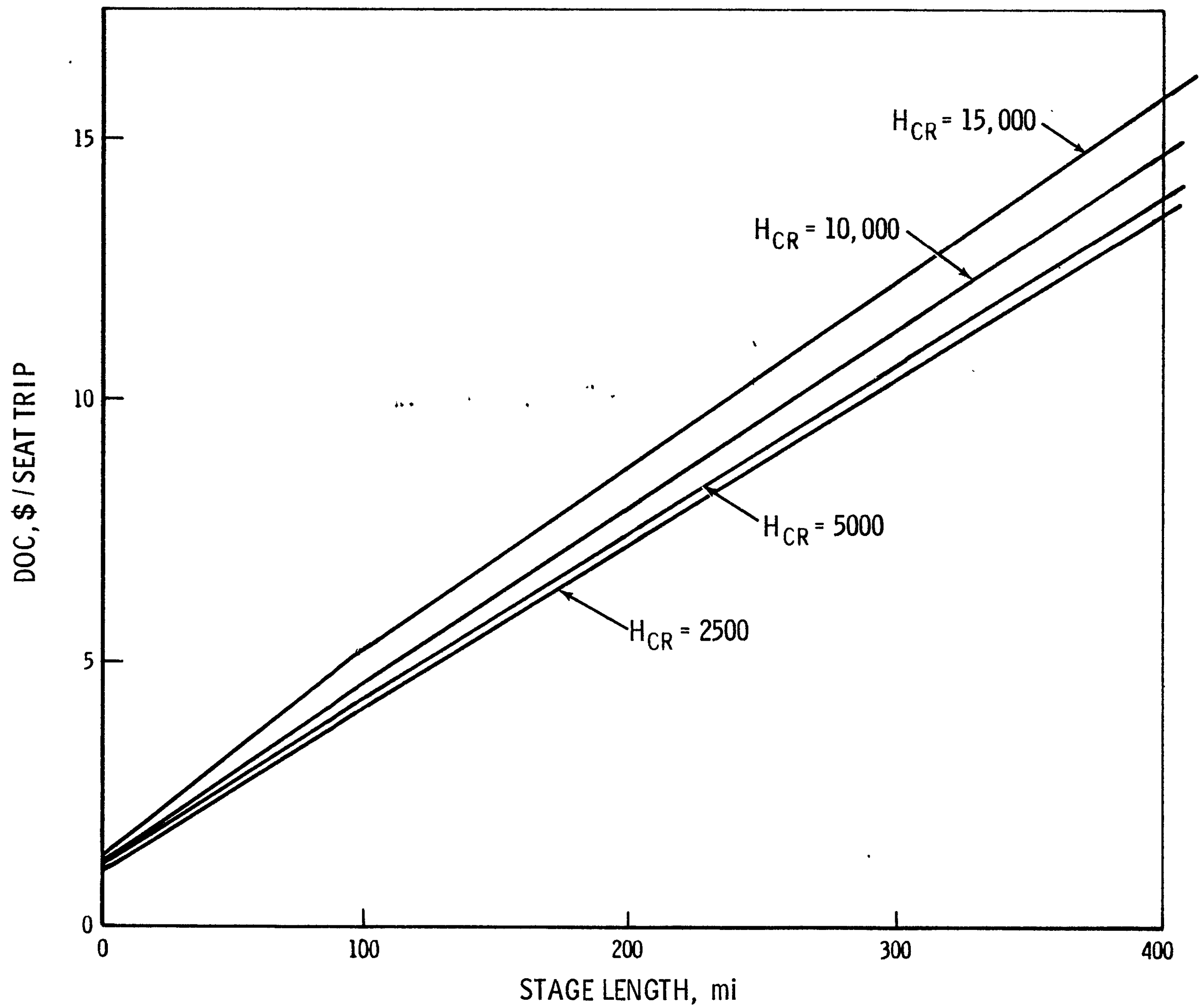

Fig. 23 DOC vs stage length for varying cruise altitude for Q75-50 helicopter.

4.0 Conclusions

The central conclusion of this work is that good economic performance can be expected from future helicopters which have low noise generation. By using high solidity rotors at lower disc loading, perceived noise levels at 500 feet for a 1975, 50 passenger, 400 miles design range vehicle can be kept below 85 dB. These levels are expected to be compatible with future operations from selected city center sites.

Experimental data on the noise generation and aerodynamic performance of low disc loading, high solidity rotors is urgently needed to improve the accuracy of the noise and performance prediction techniques used here.

Larger helicopters have a more severe economic penalty for low noise generation. Therefore the optimum economic size in a given operation will be smaller for a quiet vehicle than for a vehicle without noise constraints.

The expected improvements in helicopter technology over the next fifteen years can offset the economic penalties due to noise reduction. Thus the direct operating cost of a very quiet 1985 vehicle can equal that of a present day vehicle designed without regard to noise.

The path studies indicate that large terminal vertical climb and descent, and relatively high cruise altitude, are inexpensive ways to reduce noise. A method incorporating the noise vs. time history into a single measure of annoyance is required to pursue terminal path optimization studies further.

5.0 References

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6.0 Appendix : Definition of Helicopter Design Variables

Rotor geometry:

$$\sigma = \frac{ncR}{\pi R^2}$$

where σ = rotor solidity = blade area/disc area

n = number of rotor blades

c = mean chord of rotor blade

R = rotor radius

It is assumed that there are only two rotor angular velocities: one for hover and low vehicle speed, Ω_h, and one for high vehicle speed, Ω_{cr}. The corresponding rotational tip speeds (relative to a non-rotating observer at the hub), V_{th} and V_{tcr}, are then:

$$V_{th} = \Omega_h R$$

$$V_{tcr} = \Omega_{cr} R$$

The advance ratio is then:

$$\mu = \frac{V}{V_{th}} \quad \text{for low speed}$$

$$= \frac{V}{V_{tcr}} \quad \text{for high speed}$$

where V = forward speed of the vehicle.

The advancing blade tip speed (relative to still air), for high vehicle speed, is:

$$V_{at} = a\,M_{at} = V + V_{tcr} = V_{tcr}\,(1 + \mu)$$

where a = speed of sound

M_{at} = advancing tip Mach number

The corresponding non-dimensional thrust coefficients for low and high speed conditions are:

$$C_{T_h} = \frac{T}{\rho \pi R^2 V_{th}^2}$$

$$C_{T_{cr}} = \frac{T}{\rho \pi R^2 V_{tcr}^2}$$

where T = thrust of rotor

ρ = air density

The thrust coefficient to solidity ratios are:

$$(C_T/\sigma)_h = C_{T_h}/\sigma$$

$$(C_T/\sigma)_{cr} = C_{T_{cr}}/\sigma$$

For hover, the rotor thrust is given by:

$$T = C_L \int_0^R \tfrac{1}{2} \rho (\Omega_h r)^2 \, ncdr$$

where C_L = average blade lift coefficient

Hence, integrating and equating to the thrust as given above,

$$C_L \frac{1}{6} \rho \Omega_h^2 R^3 \, nc = C_{T_h} \rho \pi R^2 (\Omega_h R)^2$$

and $C_L = 6(C_{T_h}/\sigma)$