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COMPARATIVE PERFORMANCE OF SEVERAL SST CONFIGURATIONS POWERED BY NOISE-LIMITED TURBOJET ENGINES

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ABSTRACT

A simplified study was made in which the mission performance of three Mach 2.7 airplane configurations was compared. Both wing loading and size of the unaugmented turbojet engines were varied at different levels of suppressor technology. The lowest gross weight and the best return on investment were obtained with an advanced arrow wing configuration when a mission range of 4200 nautical miles was specified. This comparison was made for the takeoff noise levels specified in F.A.R. 36 using retractable jet noise suppressors assumed to be capable of 15 PNdB of suppression with only a 7.5-percent thrust loss. With less advanced suppressor technology, a modified delta configuration is a close competitor of the arrow wing. Despite its good takeoff characteristics, a swing-wing configuration was too structurally heavy to be competitive at F.A.R. 36 noise levels. Engine performance and weight commensurate with engine definition in 1975 were postulated. Lowering the takeoff noise requirements from those of F.A.R. 36 to F.A.R. 36 minus 10 EPNdB caused gross weight to increase by 65 percent when noise is suppressed 15 PNdB at a 7.5-percent thrust loss. To obtain a postulated acceptable level of return on investment, the load factor of a 200-seat airplane must increase from 58 percent to 88 percent at current diluted international fares.

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SUMMARY

A simplified study was made in which the mission performance of three Mach 2.7 airplane configurations was compared. Both wing loading and size of the unaugmented turbojet engines were varied at different levels of suppressor technology. Comparisons were made first without any noise constraint and then at the 108-EPNdB sideline takeoff levels specified by F.A.R. 36. For the best configuration from the latter comparison, a penalty was then assessed for meeting a noise requirement 10 EPNdB below that of F.A.R. 36. A turbojet cycle with a compressor pressure ratio of 10 and a turbine rotor-inlet temperature of 2400° F was used in these comparisons. Component efficiencies, stage loadings, and weights commensurate with a 1975 engine definition were postulated.

At F.A.R. 36 noise levels, the lowest gross weight and the best return on investment were obtained with an advanced arrow wing configuration. A range of 4200 nautical miles and a payload of 200 passengers were specified for this comparison. Advanced retractable jet noise suppressors capable of 15 PNdB suppression at a 7.5-percent thrust loss were assumed. A reduced throttle setting was required at takeoff to meet the noise requirement. With less advanced suppressor technology, a modified delta configuration is also a close competitor of the arrow wing. Despite its good takeoff characteristics, a swing-wing configuration was found to be non-competitive at the F.A.R. 36 noise level due to its heavy structure and poor supersonic cruise efficiency.

Lowering the takeoff sideline noise requirement by 10 EPNdB from the level of F.A.R. 36 causes the gross weight of the arrow wing airplane to

increase by 65 percent. The dry turbojet engine size increased by almost 150 percent. To obtain a postulated acceptable level of return on investment, the load factor of a 200-seat airplane must increase from 58 percent to 88 percent at current diluted international fares. Such a penalty is unacceptable. If F.A.R. 36 minus 10 EPNdB is to be the goal, a cycle providing some bypass air with a resultant lower exit velocity at takeoff will probably be more attractive. At the F.A.R. 36 noise level, however, the turbojet cycle seems rather attractive in view of the fact that its 58-percent load factor is close to the 50-percent average which now prevails on the North Atlantic.

INTRODUCTION

One of the difficulties that contributed to the cancellation of the U.S. SST program was the jet noise problem associated with afterburning turbojet engines - the propulsion system originally proposed. At the time of cancellation, other powerplants were being considered to achieve quieter operation - especially at lift-off. One of the most promising, yet relatively simple, of these alternative powerplants was the dry (i.e., unaugmented) turbojet with multitube or multichute jet noise suppressors that would be retractable after takeoff.

To achieve the full benefit of afterburning turbojets, afterburning would probably be desirable during takeoff if noise were no consideration. (The need for afterburning at takeoff is less apparent with certain variable-sweep wing configurations.) Otherwise, larger engines with some associated weight penalty would be required. If engine size is forced to increase enough, ultimately little benefit will be received by afterburning (even transonically) and the need for the extra weight of the afterburner becomes questionable.

Jet exit velocity is the primary jet noise parameter. Gas mass flow has some influence, but its effect is only secondary compared to velocity. Dry turbojets are quieter than afterburning turbojets without any suppression because of the lower jet velocity. Additionally, dry turbojet engine noise is more easily suppressed mechanically with multitube or multichute suppressors inserted in the gas stream because of the lower temperature. Building jet noise suppressors which can operate for any length of time in the hot environment of an afterburning exhaust stream is a serious materials problem. Even with mechanical suppression, it is likely that additional noise reduction will be required to meet the F. A. R. 36 sideline noise constraint of 108 EPNdB which occurs shortly after lift off. This can be accomplished by throttling the engines back from design thrust to some lower exit velocity. Engine size (i. e., airflow) must be increased as the takeoff throttle setting is reduced in order to obtain sufficient thrust. Here, the interaction with the airframe is important because the required takeoff thrust-to-gross-weight ratios can vary considerably from one airplane design to another. A high aspect ratio unswept wing, for example, may require considerably less thrust at takeoff than the highly swept low-aspect-ratio wings typical of supersonic designs. This explains some of the interest which occasionally recurs regarding variable sweep wings which can be unswept at takeoff and highly swept in supersonic cruise. With a given type configuration, takeoff performance can also usually be improved by a lower wing loading (i. e., larger wings) but at the expense of an increase in the structural weight fraction. The trades involved between aerodynamics, engine size, and structural weight are complex when noise is a design constraint.

In this study, dry turbojet engines are installed in three different airframe configurations to assess the effect of airframe choice on such propulsion parameters as engine size and takeoff throttle setting at different levels of jet noise suppressor technology. The penalties are assessed in terms of range/payload initially but ultimately in terms of economic results like direct operating cost and return on investment. Comparisons are made between engines sized without any noise constraint and those sized to meet the F. A. R. 36 noise constraint. A further effort is directed toward assessing the penalty for reducing the noise level to 10 EPNdB below that of F. A. R. 36.

Considerable material has already been published about studies comparing various configurations, usually powered by afterburning turbojets (e. g., ref. 1). Unfortunately, in most of the studies the airplanes were not designed for takeoff sideline noise limits as low as F. A. R. 36 (i. e., 108 EPNdB on a 0.35-n. mi. sideline). The F.A.R. 36 noise regulations exclude supersonic aircraft, but there is little likelihood that such an exclusion can be justified for a new design.

F.A.R. 36 had not then come into being when the GE4 afterburning turbojet was originally selected as the engine for the U.S. SST. Study results published since the issuance of F.A.R. 36 indicate that SST's with afterburning turbojet engines generally do not meet that sideline noise limit although the community (takeoff) noise limit at 3.5 miles from brake release could be net in most cases. Hence, the community noise is probably not as severe a constraint as the sideline noise limit if sufficient thrust is allowed for initial climb-out. In the present study a maximum limit to F.A.R. takeoff field length and a 3.5-mile altitude are specified in addition to a sideline noise limit. The climb rate that produces the noise limit was calculated for the 3.5-mile flyover station for some of the better cases. These rates were compared with minimum acceptable values defined in F.A.R. 36. Hopefully, an even greater climb rate can be maintained in order to decrease the noise footprint area.

ANALYSIS

Types of Aircraft Studied

Three types of aircraft were chosen for analysis in these mission studies because of the impact the airplane selection might have on the engine sizing requirements. The relationship between the engine size needed to meet a sideline noise constraint at takeoff and the minimum size needed for cruise, for instance, is likely to be different for each configuration. These particular requirements are dependent on the variation in the airplane's lift-drag ratio between lift-off and cruise. If the engines are sized by the sideline noise constraint, they will all have virtually the same part-power throttle setting at takeoff, but a different size (i.e., design sea-level-static airflow) will be required for each configuration if takeoff gross weight is fixed. For supersonic cruise at a given Mach number and altitude, each configuration will require a different part-power throttle setting to overcome drag since the engine sizes are different. Differences in cruise lift-drag ratios from one configuration to another may further change the throttle settings. The three configurations considered in this study are sketched in figure 1. All three types are considered in reference 1, but the particular arrow wing version shown here is an improved type discussed in reference 2.

The modified delta with tail airplane is a fixed wing concept that incorporates some compromises in supersonic aerodynamics to improve its subsonic characteristics. It features a supersonic leading edge wing with less sweepback and a higher aspect ratio than is typical for planes designed for cruise at Mach 2.7. The delta wing planform is the most efficient structurally and for a given takeoff gross weight would allow a greater payload and/or fuel load to be carried. This was the concept embodied in the Boeing 2707-300 SST proposal (ref. 1).

The advanced arrow wing has a highly-swept subsonic-leading-edge wing which gives excellent high-speed aerodynamic efficiency. Subsonically it is generally less efficient than the modified delta with tail; however, improvements have been incorporated in the advanced version which overcome some of the shortcomings of earlier versions (ref. 2). For instance, the wing leading edge radius has been increased to improve the subsonic stability in low-speed flight. Also, inboard trailing-edge flaps have been enlarged over earlier versions in order to decrease the lift-off angle of attack required to achieve the desired lift coefficient. Structural weight reductions resulting from a shorter landing gear and a less complicated variable geometry nose are some of the benefits.

The variable-sweep wing concept was chosen for analysis because its low-speed aerodynamic efficiency is generally superior to that of the other two types when its wings are unswept. Good low-speed characteristics are especially important when noise is considered because the thrust requirement and, hence, engine size and weight are reduced. Supersonic aerodynamic efficiency, however, generally suffers in comparison to the other two configurations shown. Also, there is a severe structural weight penalty due to the variable sweep feature, thus limiting either the amount of fuel or payload that can be carried for a fixed takeoff gross weight.

For all three airplane types considered, takeoff wing loading was varied to maximize a mission figure of merit. Structural weight fraction and aerodynamic efficiency variations with wing loading were accounted for.

General Procedure

Initially, the takeoff gross weight was fixed at 750 000 pounds for all three airplane types. The fuselage and furnishings were designed for a 200-passenger load. Throughout the study the fuselage dimensions and weights remained fixed, as did the 200-passenger payload (which corresponds to a weight of 40 000 lb, including baggage). The total range was allowed to vary as a figure of merit as the wing area (i. e., wing loading) and design airflow of the dry turbojet engines were varied. The analysis was made both with and without jet noise suppressors.

Airplane economic indices such as direct operating cost and return on investment are only meaningful when they are based on a constant-range mission. The best of the noise-constrained airplanes were then resized for a constant range of 4200 nautical miles (sufficient to include Seattle-Tokyo, nonstop). The takeoff gross weight was allowed to vary as required to obtain this range with a 200-passenger payload. Wing, landing gear, and tail weights were assumed to vary as a function of gross weight. The wing loading of each aircraft type was fixed at the value found to be optimum in the earlier range optimization. DOC and ROI were then calculated. Aircraft purchase price in these studies was based on a fixed (but different) cost per pound for airframe and engines.

Mission

The climb and acceleration flight path, in altitude and Mach number coordinates, was fixed for all airplanes in this study as shown in figure 2. Sonic boom constraints were ignored in the selection of this path. Weight, time, and range were computed at frequent intervals along the path by solving the equations of motion by iteration with a high-speed digital computer. An optimum initial Breguet cruise altitude at Mach 2.7 was chosen to maximize range. If necessary, horizontal cruise was maintained until the slowly rising optimum Breguet cruise path was intercepted.

A limit that was imposed during climb and acceleration was that thrust-drag ratio must not decrease below 1.3. Engine sizes which could not produce this thrust margin up to Mach 2.7 were discarded as being too small. When the optimum Breguet cruise altitude is sought at Mach 2.7, it is sometimes found that an engine which produced sufficient thrust margin in the climb and acceleration up to Mach 2.7 has insufficient thrust margin in climbing above 59 500 feet. In this case, a Breguet cruise path would begin at an altitude where the thrust-drag ratio is 1.3 - that is, at an altitude less than optimum for maximum cruise range.

Descent time and range were not computed but were assumed to be constant at a value consistent with other airplane studies. Descent fuel was based on flight idle fuel consumption over most of the assumed descent time span. All mission performance was calculated for a U.S. Standard Atmosphere (1962) with the exception of F.A.R. takeoff field length which was estimated empirically for a Standard +15° C Day.

A reserve fuel allowance used in all cases included the following: (1) extra fuel equal to 7 percent of the mission fuel; (2) fuel for a 261-nautical-mile cruise to an alternate airport at Mach 2.7 at the best altitude, and (3) fuel for a 30-minute hold at Mach 0.5 at an altitude of 15 000 feet.

Engines

Selection. - Single-spool unaugmented turbojet engines with a sea-level-static compressor pressure ratio of 10 and a turbine rotor inlet temperature of 2400° F were selected for this study. When the engine components are rematched for Mach 2.7 at full power with constant shaft speed, the compressor pressure ratio becomes 4.23. The compressor discharge temperature which results from combining the ram and mechanical compression is 1015° F at Mach 2.7 for this engine. This air is the source of the air used for turbine cooling. According to reference 3, this compressor discharge temperature is the approximate upper limit which can be used before stress-temperature problems arise for the best high-temperature alloy materials for compressor and turbine disks. A turbine rotor-inlet temperature of 2400° F is about the maximum allowable when we limit consideration to reasonable amounts of cooling airflow per stage and the turbine blades are cooled down to 1650° F with coolant at 1015° F by means of film-impingement cooling. (This corresponds to a cooling effectiveness of 0.54 and 4 percent bleed flow for the rotor blades by the method of ref. 4 when shroud and wall cooling are neglected.)

Turbine cooling. - Compressor discharge air was bled around the combustor to cool the single-stage turbine. Bleed air requirements for a metal temperature of 1650° F were calculated along the entire flight path at design shaft speed and turbine rotor-inlet temperature by using the combination film-impingement cooling curve of reference 4. The flow computed by this method was multiplied by 1.33 to account for wall and shroud cooling in addition to blade cooling. It was found that the Mach 2.7 full power condition had the most severe cooling requirement. A chargeable bleed of 5.2 percent of the compressor airflow was required for this condition. Bleed was fixed at this percentage for the entire mission except in the calculation of the hold reserve fuel and at some of the reduced thrust settings required for takeoff noise abatement. At these low-thrust conditions, no bleed was extracted for cooling. During cruise at Mach 2.7, over-cooling of the turbine components will result from the 5.2 percent bleed since operation is usually at less than maximum thrust and temperature. Even over most of the climb path some amount of over-cooling results. This provides a degree of conservatism to the design.

Component matching and mode of operation. - Engine performance was computed by means of the GENENG computer program described in reference 5. Compressor and turbine maps which were representative of the dry turbojet engine cycle used in this study were stored in the program to permit the calculation of off-design performance. Basically, the computer program matches the compressor with its driving turbine to satisfy the relations involving flow continuity and power balance.

The shaft speed N and the turbine rotor-inlet temperature were fixed at their design sea-level-static values during the full-power climb and acceleration up to the Mach 2.7 cruise condition. This was accomplished by varying the nozzle throat area as required. During cruise at Mach 2.7, reduced thrust operation was obtained by a reduction in turbine inlet temperature with an increase in the nozzle throat area to maintain constant shaft speed operation. The noise abatement takeoff was also accomplished by keeping the shaft speed constant while the nozzle throat area was increased and the turbine-inlet temperature was reduced. This procedure tends to maximize the noise reduction for a given amount of thrust reduction. The decrease in jet velocity, the main noise variable, accounts for all the thrust decrease since the gas mass flow remains practically constant. During hold at Mach 0.5 at a 15 000-foot altitude, the operating mode was somewhat different at reduced thrust. Shaft speed was allowed to vary as turbine inlet temperature was reduced from design in a manner that tended to minimize specific fuel consumption. The nozzle throat area was varied as required to obtain this operation.

Installation. - An inlet pressure recovery schedule similar to that of the Boeing 2707-300 (ref. 6) was used in these engine performance calculations. (Sea-level-static and Mach 2.7 full thrust values of this parameter as well as other component pressure ratios and efficiencies are shown in table I.) Variable inlet geometry was assumed to provide external compression at speeds up to Mach 1.6 with the centerbody or ramp fully extended. Beyond Mach 1.6 the centerbody or ramp was fully retracted for external-internal compression. No secondary airflow from the inlet was required for the translating shroud plug nozzles used in this study. Nozzle thrust coefficients used in these performance calculations (typical values of which are shown in table I for two conditions) included both internal and boattail effects, based on NASA-Lewis Research Center data obtained in flight testing an F106 with podded J85 turbojet engines mounted beneath the wings (ref. 7). Inlet drag was also accounted for in the engine performance calculations. It included spillage, bypass, and bleed drags. At Mach 2.7, there was no spillage or bypass; inlet drag consisted entirely of that due to dumping boundary layer bleed air. During part-power cruise at Mach 2.7, engine airflow was essentially unchanged from its full power value. The inlet drag during cruise, therefore, was essentially independent of power setting. During full-power supersonic climb and acceleration, spillage was the dominant component of inlet drag up to about Mach 2.2.

Installed performance. - The calculated full-power installed engine performance is shown in terms of specific fuel consumption and net thrust in figure 3. This performance includes a degradation for both inlet drag and nozzle boattail effects. It does not, however, include the inlet and nacelle wave and friction drags, which are included in the airplane aerodynamics. The net thrust shown is for a reference 900-pound-per-second engine size (i.e., 900 lb/sec corrected airflow at the compressor face, sea-level-static, standard day). As engine size was changed, it was necessary to scale this thrust with airflow. The engine performance shown in figure 3 was calculated at each Mach number for the corresponding altitude shown in figure 2.

Figure 4 shows part-power cruise sfc plotted against net thrust for Mach 2.7 cruise in part (a) and for hold at Mach 0.5 in part (b). Since the optimum supersonic cruise altitude is not really known until the engine is "flown" in an airplane on a simulated mission, performance is plotted for a range of altitudes in figure 4(a). The circled points at the right of each curve represent the full thrust condition. Sfc improves as power is reduced from full thrust down to about 60 percent of full thrust. Because of thrust margin requirements, takeoff sideline noise constraints, and the optimization of the Breguet cruise altitude, the engines of this study typically cruise at a thrust setting just to the right of the bucket of these curves. The Reynolds number variation with altitude, which would have only a small effect over this range of altitudes, has been ignored.

In figure 4(b) the part-power performance curve is shown only over the thrust range of interest. Maximum thrust would be about 55 000 pounds with design cooling bleed. For the range of thrusts shown in this figure, no cooling bleed was used. For the optimized airplanes of this study, the hold sfc generally varied between 1.45 and 1.15 pounds of fuel per hour per pound of thrust. This variation was a function of the airplane configuration and the effectiveness of the jet noise suppressors during takeoff. The thrusts shown in figure 4 represent levels obtained at any given sfc with the reference 900-pound-per-second engine size. For any other design corrected airflow, the thrusts shown would have to be scaled with airflow.

Some of the pertinent characteristics of the propulsion system are shown in table I for sea-level-static and Mach 2.7 full thrust conditions. The sea-level-static condition was chosen because it represents design point operation. Takeoff, however, is not generally accomplished at the full-thrust design condition because of the excessive jet noise that would be generated. At Mach 2.7, full thrust conditions are only used for a short time in climb to seek the best cruise altitude. During cruise, thrust is reduced from the maximum setting. The two conditions in table I were chosen merely to illustrate the maximum excursion that would occur from the design point at full thrust. The compressor is seen to move considerably away from its design point by the time the Mach 2.7 condition is reached. Nevertheless, because of the shape of the efficiency contours on the compressor map, the efficiency has improved. The turbine, on the other hand, does not move very far away from design point operation at full-thrust conditions. Component design efficiencies may seem somewhat optimistic in some cases but generally they are those felt to be attainable for an engine definition in 1975.

Weight. - Calculations made with the Gerend engine weight estimating procedure (ref. 8) indicate that the GE4/J5P represents technology available in the year 1972. (Estimates of afterburner, nozzle, and thrust reverser weights were made and then subtracted from a quoted GE4 package weight since the Gerend calculations do not include these propulsion system elements.) The Gerend procedure further indicates that for the near-term future engine weight reductions of about 5 percent per year can be expected. For this study it was decided that a two-year advance over GE4 weight technology would be appropriate. Hence, if the GE4/J5P cycle design parameters were retained for this study, the engine weight would be about 10 percent lighter than before. For the same design airflow, an even greater weight reduction is obtained in our study engines with the Gerend weight estimating procedure because the compressor pressure ratio is lower than in the GE4. Weight is also scaled with engine design airflow raised to the 1.2 power, as recommended by Gerend. The bare engine weight (i.e., without nozzle and reverser) of a 900-pound-per-second dry turbojet was calculated by this procedure to be 11 868 pounds. It was assumed that the weight of the nozzle, inlet, nacelle, and added installation equipment would also be 10 percent lighter than scaled GE4/J5P weights for these components. A nozzle weight of 3362 pounds, an inlet-plus-nacelle weight of 4825 pounds, and an added installation weight (which includes installation subsystem, controls, starting and fuel systems) of 1455 pounds was used for the reference 900-pound-per-second engine size. The total propulsion system weight for a four-engine airplane with reference-size engines, then, would be 86 050 pounds without noise suppressors.

There is some uncertainty as to the weight of jet noise suppressors which retract into the plug nozzle centerbody when they are not needed. There is also some uncertainty as to how much suppression they will provide and the thrust penalty involved. In this study, "baseline" suppressors were assumed to weigh 3660 pounds per engine and provide 10 PNdB of suppression with a 10 percent loss in net thrust. After being retracted, it was assumed there would be no thrust loss due to the presence of the suppressor mechanism. "Advanced" technology suppressors were also considered and were assumed to provide 15 PNdB suppression with 7.5 percent thrust loss at half the weight penalty of the "baseline" suppressors. With baseline suppressors, then, the total airplane propulsion system weight was 100 690 pounds for the reference-size engines. With advanced suppressors, the corresponding weight was 93 370 pounds. The baseline suppressor characteristics are those which are thought to be currently attainable. Advanced suppressor characteristics are those which could be attainable with considerable effort in a 1975 engine definition.

Airframes

Aerodynamics. - Appropriate drag polars based on unpublished industry data were scheduled against flight Mach number for each of the three airplane types shown in figure 1. Each set of polars was for a particular reference airplane design. In this study, however, we desire to change wing loading and, later, gross weight while payload and fuselage dimensions are fixed. This implies that wing area and perhaps tail area will change with respect to fuselage area. The reference polars for each of the three airplane types, then, must be allowed to change to account for the dimensional nonsimilarity within a given airplane type. Drag build-up curves showing the amount of drag accounted for by each component are needed before the effect of relative area changes can be determined. Unfortunately, these were not readily available.

The airframe component areas and dimensions for a reference airplane of each of the three types considered were then estimated as a first step in the synthesis of a drag build-up procedure. The airframe was broken down into a set of components: wing, body, vertical tail, and horizontal tail. Total minimum drag from the reference polars was assumed to be composed of the sum of the friction and pressure, or wave, drags of these components. The nacelles were not considered as such, but an area representative of their surface area for a reference size was included with the body area.

The component skin friction coefficients were calculated by means of the Prandtl-Schlichting equation

$$C_{fic_{component}} = \frac{0.455}{(\log_{10} Re_{component})^{2.58}} \tag{1}$$

This equation gives the skin friction coefficient for incompressible turbulent flow over one surface of a flat plate. These coefficients were then corrected for compressibility effects by multiplying by a correction factor which was a strong function of Mach number and a weak function of altitude. These correction factors are based on empirical turbulent flow flat plate data such as that shown in reference 9 plotted against Reynolds number at various Mach numbers. The total airplane friction drag coefficient, based on a reference wing planform area, was obtained by correcting the component skin friction coefficients to the common reference area and adding, as follows:

$$C_{D_{f_{total}}} = 2\left(C_{f_{wing}} + C_{f_{vt}} \times \frac{S_{vt}}{S_{wing}} + C_{f_{ht}} \times \frac{S_{ht}}{S_{wing}}\right) + C_{f_{body}} \times \frac{S_{body}}{S_{wing}} \tag{2}$$

The wing, vertical tail, and horizontal tail skin friction coefficients in this equation are doubled to account for both surfaces of these components.

The pressure drag coefficients of each of the various components based on a representative component area are corrected to the wing planform area and added, as follows:

$$C_{D_{p_{total}}} = C_{D_{p_{wing}}} + C_{D_{p_{vt}}} \times \frac{S_{vt}}{S_{wing}} + C_{D_{p_{ht}}} \times \frac{S_{ht}}{S_{wing}} + C_{D_{p_{body}}} \times \frac{A_{body}}{S_{wing}} \tag{3}$$

The body and tail pressure drag coefficients based on their representative component areas are assumed to be scheduled with Mach number in the same manner as representative empirical data for these types of components in other airplanes. The wing pressure drag coefficient was varied in an iterative calculation at each Mach number until the total minimum drag coefficient obtained by adding equations (2) and (3) agreed with the $C_{D_{min}}$ obtained from the reference drag polars for each of the three types of airplanes considered. The drag polars were assumed to be parabolic so that the total drag coefficient could be expressed as

$$C_{D_{total}} = C_{D_{min}} + \left[\frac{C_{D_i}}{(C_L - C_{L_o})^2}\right]\left(C_L - C_{L_o}\right)^2 \qquad (4)$$

The induced drag term within brackets and C_{L_o} were both assumed to have a schedule against Mach number that did not change with variations in takeoff wing loading or gross weight for a given airplane type. The $C_{D_{min}}$ term in equation (4) will, however, vary with takeoff wing loading and gross weight because the body to wing area ratios in equations (2) and (3) will change. (The area ratios between the tail components and the wing are assumed to remain constant within a given airplane type.) In addition, the skin friction coefficients of the wing and tail components change because the characteristic length in the Reynolds number term (eq. (1)) changes with takeoff wing loading or gross weight. The component pressure drag coefficients based on their own representative areas (eq. (3)) are assumed to have a fixed schedule with Mach number for each airplane within a given type.

Typical aerodynamic characteristics resulting from this analysis are shown in table II for the three airplane types considered. Takeoff wing loading was optimized for maximum range for an F.A.R. 36 sideline noise constraint of 108 EPNdB. A jet noise suppression of 10 PNdB was assumed to be available. The table shows that cruise L/D at Mach 2.7 is best for the arrow wing and worst for the variable sweep configuration. The L/D during the initial climb with gear retracted, however, is worst for the arrow wing and best for the variable sweep airplane at an altitude of 767 feet at the takeoff power setting. (The altitude of 767 ft was selected because it is about midway between the 35-ft obstacle at takeoff and the 1500-ft altitude at the 3.5-n. mi. point where thrust is reduced to meet the community noise constraint.) Good L/D in initial climb up to an altitude of 1500 feet at a distance of 3.5 miles from brake release is important because of its effect on the thrust-to-gross-weight ratio required. The lift-off C_L is lowest (worst) for the arrow wing and highest (best) for the variable sweep.

High values are desirable because they tend to shorten the takeoff distance. The numbers appearing in table II are merely typical of results obtained in this study. Changes in takeoff wing loading, gross weight, or amount of jet noise suppression available would cause the L/D ratios to change somewhat although the relative ranking would remain the same for airplanes within the range of interest.

Weights. - Airframe weight varied as takeoff wing loading and gross weight varied in this study. The weight of the body, furnishings, etc., was fixed for each airplane of a given type, but the wing, tail, and landing gear weight varied. The wing weight for each airplane type was assumed to vary according to the relationship (ref. 10)

$$W_{wing} = K(W_g S_{wing})^{0.64} \tag{5}$$

where K is a function of airplane type. This type of scaling equation shows good agreement with empirical data for many types of wings and airplanes. The value of K that was used for each airplane type in the study can be determined by correlating the weight data in table III with equation (5).

Table III shows the weight of the various airframe components for each of the three airplane types at a particular reference wing loading with takeoff gross weight equal to 750 000 pounds. The landing gear weight was maintained at a fixed fraction of the takeoff gross weight. The tail weight was scaled directly with wing area for each airplane of a given type. The reference wing loadings used in table III do not represent range-optimized values from this study with F. A. R. 36 noise as a constraint, but instead represent estimates of the loadings used by others in their weight analyses for similar airframes (e.g., ref. 1). The airframe weight components shown are based on unpublished industry data. The body weight, however, has been adjusted in some cases to provide a 200-passenger seating arrangement for each study airplane. Some changes have been made also to gross-weight sensitive components (e.g., wing and landing gear) so that component weights could be tabulated on a common gross-weight basis. In the case of the arrow wing concept, the landing gear weight at all except very high wing loadings has been reduced to the same value as the other configurations. This reflects the addition of more inboard trailing edge flap than in earlier configurations of this type, thus permitting an appropriate lift-off C_L to be obtained at about half the previously required wing angle of attack. No composite structure was assumed for any of the study airplanes.

To obtain the takeoff gross weight, the propulsion package, payload, and fuel load must be added to the airframe weight. The propulsion weight in this instance includes not only the normal propulsion items but also the nacelles and inlets.

Takeoff

The calculations made for takeoff are important because they affect the thrust-to-weight ratios obtained. For a given noise constraint and a given amount of jet noise suppression, the calculated thrust-to-gross-weight ratio determines the specific airflow and engine weight.

An F. A. R. takeoff field length maximum limit of 12 400 feet on a $+15^{\circ}$ C day was used in this study. The F. A. R. takeoff distance assumes clearance of a 35-foot obstacle with an engine failure at a point where the distance to climb over the obstacle would be the same as the distance to stop with maximum braking. An equation was developed from SST study empirical data for several different configurations to relate F. A. R. distance to airplane aerodynamic and engine thrust parameters, as follows:

$$L_{F.A.R.} = 26.6 \times \frac{(W_g/S_{wing})}{(C_{L_{L.O.}})(F_n/W_g)} - 390 \tag{6}$$

This grouping of variables was suggested by reference 11. The coefficients, however, were obtained from more recent SST study data. Equation (6) gives the F. A. R. distance for a $+15^{\circ}$ C day, but F_n/W_g is based on standard day calculations. W_g/S in equation (6) is calculated at maximum gross weight but F_n/W_g is calculated at the point of lift-off, which for most airplanes in the study is near Mach 0.3. W_g at the point of lift-off is about 1.1 percent less than maximum for most of the airplanes considered.

F.A.R. 36 specifies that flyover noise must not exceed 108 EPNdB (ignoring possible noise trades) for airplanes in this weight category at a point beneath the flight path 3.5 nautical miles from the point of brake release. Thrust may be reduced from the takeoff setting to a level which will provide a 4-percent climb gradient or level flight with an engine out, whichever is greater. To help insure that this noise goal would be met, an altitude of 1500 feet was specified at the 3.5-mile measuring point. It was anticipated at the beginning of the study that with an altitude this high the sideline noise after lift-off would be the most difficult noise goal to meet. Once the 1500-foot altitude is specified at 3.5-miles, the F_n/W_g requirement for takeoff and initial climb can be approximated by

$$F_n/W_g = \frac{\dfrac{V_{3.5} - (V_{L.O.} + 10)}{g} \times 1.69 + \dfrac{1500 \ln\left[V_{3.5}/(V_{L.O.} + 10)\right]}{V_{3.5} - \left[(V_{L.O.} + 10)\right] \times 1.69}}{\dfrac{3.5 \times 6076 - 0.875\, L_{F.A.R.}}{0.5(V_{3.5} + V_{L.O.} + 10) \times 1.69}}$$

$$+ \frac{\bar{q}\, C_{D_{min}} + \dfrac{\left[C_{D_i}/\left(C_L - C_{L_o}\right)^2\right]}{\bar{q}} \left(0.989\, W_g/S_{wing} - C_{L_o}\, q\right)^2}{0.989\, W_g/S_{wing}} \qquad (7)$$

This equation is essentially a rearrangement of the terms of equation (10) from reference 11. The velocity terms are in knots in equation (7), but the other terms are in the usual units. Changes in gross weight from lift-off to the 3.5-mile point are ignored and lift is assumed to equal weight.

Equation (7) calculates the F_n/W_g required to go from the 35-foot obstacle to a point 3.5 miles from brake release while gaining 1500 feet in altitude. It is assumed that the initial velocity is 10 knots greater than the lift-off velocity and that the 3.5-mile velocity can be determined later in an iteration which minimizes F_n/W_g. The lift-off velocity is calculated from the equation

$$V_{L.O.} = \sqrt{\frac{296 \times 0.989\ W_g/S_{wing}\ (1 - 0.15\ F_n/W_g)}{C_{L_{L.O.}}}} \qquad (8)$$

The lift-off C_L was given previously in table II for each of the three airplane types. F_n/W_g is included in equation (8) because at lift-off the thrust and free-stream velocity vectors are assumed to be displaced by an angle of about 9 degrees.

The F.A.R. field length from equation (6) must be substituted into equation (7) to obtain the distance from a 35-foot obstacle to the 3.5-mile range. In equation (7) the field length is multiplied by a factor of 0.875 to remove the engine-out effect from the distance. An iterative technique is involved in solving the system of equations (6) through (8) since F_n/W_g appears in each one.

The coefficients describing a parabolic drag polar are also required in equation (7). It is assumed that these aerodynamic coefficients are constant after clearing the obstacle until the 3.5-mile range is reached. This is merely an approximation, however, since during the initial part of climb the landing gear is being retracted, while in the final part the flap setting is being reduced for a slower climb rate beginning at the 3.5-mile range. These changes in actuality would cause the aerodynamic coefficients to change.

The interative solution of the system of equations just described was accomplished with a high-speed computer over a range of takeoff wing loadings for each of the three airplane types. The results are shown in figure 5. The lift-off Mach number, F.A.R. field length, and F_n/W_g required to climb to an altitude of 1500 feet at 3.5 nautical miles from the start of takeoff roll are shown in this figure. All of these variables improve as wing loading is reduced. The velocity at 3.5 miles has been varied in each case to minimize the F_n/W_g requirement or to keep the field length below the

12 400-foot, +15° C, limit when it was approached. Although the takeoff characteristics improve as wing loading is reduced, the structural weight fraction probably increases. Improving the takeoff characteristics will very likely reduce the engine size required, especially when noise constraints are imposed. The variable sweep airplane has the best takeoff characteristics, but it is the heaviest structurally - even though its wing loadings of interest are higher than those of the other two types. The only way to assess the tradeoffs involved is to perform a mission study.

A discontinuity appears in the Mach number and distance curves of figure 5 for the arrow wing airplanes. It occurs because the landing gear was lengthened to the value it originally had in the older arrow-wing designs without inboard flaps as wing loading was increased beyond 84 pounds per square foot. At the higher wing loadings, the lift-off angle of attack was increased to 9.5 degrees to permit a C_L of 0.75 to be obtained with an acceptable F.A.R. takeoff field length (i.e., below the 12 400-ft limit). A structural weight penalty was incorporated in the mission calculations to account for the longer landing gear. Below a wing leading of 84 pounds per square foot, the lift-off angle of attack was maintained at 4.5 degrees with a C_L of 0.51.

Despite the seemingly large number of approximations and assumptions in equations (6) through (8), the results obtained with them agree closely with those obtained by more exact calculations in other SST studies. The scarcity of refined low-speed aerodynamic data for these configurations does not warrant more elaborate calculations at this time. The trends shown in figure 5 are believed to be generally representative of the takeoff characteristics of the types of aircraft shown in figure 1.

Noise

In the noise calculations for the turbojet-powered airplanes of this study it was assumed that jet noise was the predominant noise source. Peak sideline noise (as per F.A.R. 36) was calculated after lift-off on a 0.35-nautical-mile sideline with the airplane at an 800-foot altitude.

The peak sideline noise occurs at this altitude because it is the lowest at which there is no significant extra ground attentuation. (When the angle of elevation from the observer is 20^{o} or more, extra ground attentuation is insignificant, according to ref. 13).

As mentioned previously, sideline noise was the major noise constraint in this study because an airplane altitude of 1500 feet was specified for the 3.5-nautical-mile takeoff (community) flyover noise measuring station. At this station, F.A.R. 36 specifies that power can be reduced to that required for a 4-percent climb gradient or level flight with an engine out, whichever is greater. In these 1500-foot altitude flyover calculations, SAE jet noise calculated in PNdB (refs. 14, 15) was converted to effective PNdB (EPNdB) by subtracting 1.6 dB, an approximate conversion factor which is a function of altitude and Mach number and was developed for use as an approximation in the Advanced Transport Technology program. EPNdB and PNdB were assumed to be equal at the sideline condition. The EPNdB scale attempts to correct the PNdB scale for (1) subjective response to maximum pure tone and (2) the duration of the noise heard by the observer (ref. 16).

Economics

Both direct operating cost (DOC) and return on investment (ROI) economic analyses were made for the best airplanes from this study. These results were computed for a fixed range of 4200 nautical miles with a variable takeoff gross weight for the two most promising configurations at two levels of jet noise suppressor technology. The 1967 ATA standard method of computing DOC (ref. 17) was used in this study. A discounted cash flow method was used to compute the return on initially invested capital (i.e., the down payment in the ROI studies. The crew cost, fuel cost, spares factor, economic life, residual value, and utilization specified in reference 17 for an international flight SST were used in both the DOC and ROI computations. It was necessary to add an indirect operating cost

(IOC) to the DOC less depreciation to obtain an annual operating cost in calculating ROI. There are several methods available for calculating IOC, but data for three types of large four-engine transports (ref. 18) including an SST, indicate that a value of 1.1 cents per seat-statute-mile would be approximately correct for all three. Thus, an IOC of 1.1 cents was used in all ROI calculations of this study.

To calculate both DOC and ROI, it is necessary to make assumptions about airframe and engine purchase prices. In these studies, these prices were each assumed to have a fixed value per pound. Airframe price, including development, was assumed to be \$158.70 per pound in current 1972 dollars. The airframe in these calculations is assumed to include the inlets and nacelles. The engine price, including development costs, was assumed to be \$179.30 in current collars. These are the prices without spares. A spares factor of 1.1 for the airframe and 1.4 for the engines (from ref. 17) was applied in calculating ROI. Based on the above assumptions, a Boeing 2707-300 SST with afterburning turbojet engines would cost \$52.1 million without spares. This agrees with an estimate of unit price by the late 1970's by Government witnesses before the House Appropriations transportation subcommittee in 1969 (ref. 19).

In the ROI calculations, a reference load factor of 50 percent was generally used. This is the approximate average now prevailing on North Atlantic routes. Other load factors were later examined in a sensitivity study. The prevailing average yield per revenue-passenger-statute-mile is just under 5 cents on a fully diluted basis for U.S. scheduled airlines on international flights. The 5-cent yield was used as a reference in another sensitivity study with load factor varying.

The return on initially invested capital can vary appreciably with the initial investment (i.e., down payment) when part of the purchase price is to be financed. A sensitivity study will show the importance of the initial debt-equity ratio when calculating ROI. In many of the other ROI calculations, results were obtained both with and without financing. In the cases with financing, a down payment of 40 percent of the full purchase price (including spares) was used as being fairly typical of past airline purchases. The balance was financed at a true annual rate of interest of 10 percent over the 15-year economic life of the airplane. The amount of each annual loan payment was constant at

$$M = P \frac{i(1+i)^N}{(1+i)^N - 1}$$

$$= 0.131\ P \tag{9}$$

The income tax rate was assumed to be 48 percent of pretaxed earnings. Residual value (as recommended in ref. 17) was zero at the end of 15 years. A double-declining balance method of depreciating the airplane was used for the first half of its economic life. Straight-line depreciation was used thereafter.

In the ROI calculations, no assumptions were made concerning the fleet size and date of introduction of each airplane into the SST fleet. The analysis was made only for a single airplane.

RESULTS OF MISSION STUDIES

Variable Range with Fixed Gross Weight

With takeoff gross weight fixed, range is expected to decrease as engine size is increased beyond that required to satisfy takeoff and supersonic thrust margin constraints. The decrease in range is expected because the engine weight goes up as the engine size increases. There is a direct tradeoff between engine weight and fuel load since takeoff gross weight and payload are fixed. With heavier engines, the available fuel load at takeoff must decrease. Range generally suffers as a result. There are some compensating effects, however, from the larger engines that tend to reduce this range decrement. A faster acceleration through the high-drag, high-specific fuel consumption part of the climb path is one benefit. Another is that thrust can be reduced more in supersonic cruise, thereby permitting operation at a lower sfc (see fig. 4(a)) with larger engines.

These expectations are verified in figure 6 (parts (a) through (f)). The various parts of figure 6 are plots of range against engine design corrected ariflow for (1) no suppressor weight penalty and (2) a baseline suppressor weight penalty for each of the three configurational types. In each part of figure 6 a curve is shown for each of several takeoff wing loadings. The circled points on each of these curves represent engine sizes required to meet the F.A.R. 36 sideline noise goal of 108 EPNdB. For the curves with baseline suppressor weight technology, the circled points represent engine size requirements when jet noise suppression is 10 PNdB (gross) with a 10 percent loss in net thrust during takeoff. (The suppressors are assumed to be retracted after takeoff with no further penalties.) Even with 10 PNdB of mechanical suppression, a considerable amount of throttling is required to meet the F.A.R. 36 noise goal. For the curves without any suppressor weight penalty, the circled points represent F.A.R. 36 engine size requirements with the noise goal obtained entirely by throttling.

A considerable reduction in engine size is possible when the 108 EPNdB sideline noise level is met with jet noise suppressors, as may be seen by comparing the circled points with suppression in figure 6(b) with those without suppression in figure 6(a) for the modified delta with tail configuration. A similar comparison can be made in succeeding parts of figure 6 for the (fixed) arrow wing and the variable-sweep (arrow) wing configurations. A further comparison shows that the 108-EPNdB engine size giving the best range is the smallest with the variable-sweep aircraft and the largest with the arrow wing.

A summary of the range results of figure 6 is replotted as figure 7 for cases meeting the 108 EPNdB sideline noise goal (a) with the 10 PNdB baseline suppressors and (b) without any suppression. It is apparent that, even with the large weight penalty and thrust loss accompanying the baseline suppressors, there is a range improvement relative to that attainable without any mechanical suppression. At the optimum wing loadings, the improvement amounts to about 200 miles for the fixed arrow wing and 100 miles or less for the modified delta and variable sweep configurations. With suppression, the arrow wing range is about 100 miles greater than that of the modified delta. The variable sweep has the poorest range - about 900 miles less than that of the arrow wing. Without any suppressors, the arrow and modified delta airplanes achieve about the same range at their best wing loadings. The range obtainable with a variable sweep airplane is about 800 miles poorer than that of the other two types.

An airplane weight breakdown (as a percentage of takeoff gross weight) is shown in figure 8 for the best airplane of each configurational type meeting the F. A. R. 36 noise goal with baseline (10 PNdB) suppressors. From left to right, they are listed from best to worst, based on the range comparison. The structural weight fraction (including inlets, nacelles, and furnishings) is about 3 percentage points higher for the arrow wing than for the modified delta configuration. The structural fraction of the variable-sweep configuration is over 7 points higher. The variable-sweep airplane receives some weight benefit from its smaller engines, but not enough to completely offset the effect of the heavier structure. Some benefit for the variable-sweep airplane also accrues from its smaller reserve fuel load - the result of better subsonic hold aerodynamics. But its supersonic aerodynamics are considerably worse than those of the arrow wing (see table II) which has just a slightly greater mission fuel fraction. Despite the closeness of the mission fuel fractions (about 7500 lb more fuel is available with the arrow wing), the variable-sweep configuration suffers in the range comparison by 900 nautical miles. If 7500 pounds of additional mission fuel were somehow available to the variable-sweep aircraft, its range would increase by only about 100 miles. Hence, a smaller mission fuel fraction is only a part of the range problem with the variable-sweep airplane. A significant part of the range insufficiency is related to aerodynamics.

Figure 7 showed that even the rather conservative assumptions that were made for the baseline suppressors were sufficient to provide range improvement over that obtained without mechanical suppression when the 108 EPNdB sideline noise goal is met. Figure 9 shows the range improvement over the baseline that can be obtained with either a greater amount of suppression or less specific thrust loss per unit of suppression. The comparison is made only for the modified delta airplane type. The suppressor weight assumptions made for this comparison are all at the baseline technology level. Figure 9 shows that about 200 miles of additional range can be obtained with 10 PNdB of suppression if the thrust loss due to suppression is reduced from 10 percent to 5 percent. If 15 PNdB of suppression is assumed instead of 10 PNdB, the range will increase about 300 miles for the baseline thrust loss of 1 percent per PNdB of suppression. A range improvement of about 500 miles could be obtained if the additional suppression and the reduced specific thrust loss could be combined at the baseline weight technology level. The engine size reductions resulting from these perturbations are also shown in figure 9. Both of these perturbations are thought to be reasonable goals to seek for an advanced engine defined in 1975 that is ready for first flight in about 1980 and commercial service prior to the mid-80's.

The next figure (fig. 10) shows the effect of suppressor weight on range with the modified delta aircraft. Both baseline weight technology and weightless suppressor results are shown to illustrate the potential range improvement due to suppressor weight reduction. The baseline suppression of 10 PNdB was used in these curves, but both baseline and half the baseline thrust loss due to suppression were considered. Weightless suppressors would offer a potential range improvement of about 300 miles for the engine sizes of interest.

In addition to having 15 PNdB of suppression with a 7.5 percent resultant loss in net thrust at takeoff, an advanced suppressor is defined as having half the weight penalty of a baseline suppressor for the same gas flow capability. Figure 11 shows the range comparison between modified delta airplanes using baseline and advanced suppressors to meet the 108 EPNdB sideline noise constraint. At a wing loading of 90 pounds per square foot, advanced suppressor technology would permit a range increase of 700 nautical miles, or 18.5 percent. Because of the size of this possibly achieveable improvement, it is concluded that much emphasis should be given to SST suppressor research and development. It could very well mean the difference between success or failure of an SST when the results are reduced to economic terms. A further gain of about 400 miles could be obtained if the wing loading were increased and no noise constraint were imposed. It is unrealistic to assume that no noise constraint will be imposed, but the upper curve of figure 11 representing this case also illustrates the potential of weightless suppressors capable of 19 to 20 PNdB of suppression without any thrust loss. The payoff is great from improvements in suppressor technology. Unfortunately, at this time there is more uncertainty regarding the installed performance characteristics of this component than for any of the other engine components. And the major SST design constraint is noise!

Figure 12 shows the same information as figure 11 except that the arrow wing and the variable sweep configurations are also included. A comparison between configurations shows that with advanced suppressor technology the arrow wing range is about 400 miles greater than that of the runner-up modified delta. A previous comparison had revealed that the arrow wing held a margin of about 100 miles over the delta when baseline suppressors were used. In the previous comparison, it was found that the ranges of the two aircraft were about equal when the noise goal was met entirely by throttling oversize engines. Hence, as suppressor technology improves, the superiority of the arrow wing concept becomes more evident. With advanced suppressor technology, the variable sweep configuration is in an even poorer relative position than in the previous comparison with baseline suppressors. The arrow wing range is 1350 nautical miles greater than that of the variable sweep with advanced suppressors when a sideline noise of 108 EPNdB is maintained.

DOC's and ROI's were computed for all three airplane types with baseline suppressors. DOC's were also computed for the cases where the 108 EPNdB noise goal was met without suppressors. The use of baseline suppressors caused the DOC in all cases to decrease less than 0.05 cent per seat-mile. The DOC for the best airplanes was about 1.75 cents per seat-statute-mile. DOC's for the variable sweep airplanes were calculated to be over 2 cents per seat-mile. The design ranges were used in these calculations. They were 4000 or more miles for the arrow and modified delta airplanes but only 3000 miles for the variable sweep airplane (see fig. 7). It is not entirely correct to compare the economic cost of the various possibilities when the ranges are significantly different.

ROI calculations were also made for all three configurations using a fare (yield) of 7 cents per revenue passenger-statute-mile with a 50 percent load factor. These results showed that ROI's of 20 percent or more were attainable with the two better configurations using baseline suppressors for cases where the down payment was 40 percent of the total purchase price. The variable sweep configuration, however, could obtain an ROI of only about 3 percent. Although these comparisons are not entirely valid because of the inequality of the ranges, they are sufficient to show that the variable sweep configuration is not a serious contender with the structural weight and aerodynamic assumptions of this study. (At noise goals lower than F.A.R. 36 it might receive a more favorable comparison.) To make a more valid comparison of the economics, it is necessary to scale the airplane weights as required for a fixed range and payload. Since it was desired to scale the airplanes for a range of 4200 nautical miles with a payload of 200 passengers, a tremendously heavy variable sweep airplane would result if the scaling rules used in this study are valid so far from the basepoint. Because of the poor economic results just discussed for this airplane on a fixed gross-weight basis, it was eliminated from further consideration.

Variable Gross Weight with Fixed Range

The best arrow wing and modified delta airplanes from the preceding analysis were scaled to provide a constant range of 4200 nautical miles. The airplanes were scaled for both the baseline and the advanced suppressor technology. The results are presented in table IV. Based on gross weight, the arrow wing is superior to the modified delta airplane at either level of suppressor technology. For both airplane types, the gross weights were greater than the base of 750 000 pounds with baseline suppressors and lower than the base for advanced suppressors. The use of baseline instead of advanced suppressors causes a gross weight increase of from 36 to 41 percent - again reinforcing the previously stated conclusion that suppressor technology is the single most important propulsion item contributing to the success or failure of a turbojet-powered SST.

The design sea-level-static engine corrected airflows needed to meet the 108 EPNdB noise constraint are very large (approximately 1300 lb/sec) for the scaled-up airplanes with baseline suppressors. The manufacture of such large turbojet engines may be a difficult problem. The placement of such large engines on the airplane may also present drag and wing structural problems that were ignored in this analysis. Area ruling considerations may dictate a greater number of smaller engines. In this study, only four-engine airplanes were considered. With baseline suppressors, however, a six-engine airplane might be a better choice if dry turbojet engines are selected. Engine design airflows would then be less than 900 pounds per second.

The engine size is not a problem with the advanced suppressors. For a four-engine airplane, the airflow would be 750 pounds per second or less. Suppressor technology is so important, then, that it may dictate the number of engines per airplane.

The cruise sfc is such that the engines are operating near the bucket in the sfc curve (fig. 4(a)). This means that turbine rotor inlet temperatures lie between 2000° and 2100° F for the cruise data presented in table IV. These temperatures are from 300° to 400° F below design, but the design turbine cooling flow has, nevertheless, been retained. This provides a certain element of conservatism to the engine design.

The lift-off Mach number does not appear to be a significant problem for any of the optimized airplanes presented in table IV. It is generally agreed that the speed at lift-off should not exceed approximately 205 knots. The highest tabulated Mach number corresponds to 190 knots. Part of the reason for this margin is that the optimum wing loading with a noise constraint is lower than without such a constraint. Lower wing loadings make possible lower lift-off speeds if the landing gear length is unchanged.

It was found that with the engines sized to meet a 108 EPNdB sideline noise goal, the 3.5-mile noise goal of 108 EPNdB could be easily met with the suppressors still deployed by a further thrust reduction while still exceeding a specified minimum acceptable climb rate. In fact, the actual noise-constrained climb rates were 2 to 3 times the minimum acceptable values.

The direct operating costs were exceedingly high for the two airplanes with baseline suppressors, with the arrow wing being the choice on this basis of selection. Advanced suppressors lower the DOC to more reasonable levels which are more than 30 percent lower than those obtained with baseline suppressors.

The calculated return on initial investment (i.e., an initial 40 percent down payment) is shown for each of the four airplanes in table IV for a fare of 7 cents per revenue-passenger-statute-mile and a load factor of 50 percent. With baseline suppressors, an ROI of zero was obtained with the modified delta airplane, whereas an ROI of 13 percent was obtained with the arrow wing. The ROI is vastly superior with either configuration if advanced technology suppressors are used. An ROI of 50 percent is obtained with these suppressors in the modified delta airplane; 55 percent is obtained with the arrow wing.

The amount of the airplane purchase price to be financed has a significant effect on the return on initial investment. This is illustrated by the curve of figure 13, which shows that as the down payment approaches zero the return on initial investment approaches infinity. A down payment equal to 40 percent of the purchase price was assumed for the ROI results of table IV since this has been fairly typical for aircraft purchases in recent years.

The rather high fare of 7 cents per revenue-passenger-mile used in the ROI comparison of table IV was chosen to ensure that the calculated ROI's would be positive. (Negative values cannot be handled by the computer program.) The effect of fare on ROI is shown in figure 14 for both airplane configurations and suppressor technologies used in table IV. These curves were computed for a load factor of 50 percent and an initial down payment equal to 40 percent of the purchase price. With baseline suppressors, fares of 7.5 to 7.9 cents per revenue-passenger-mile are required for a 20-percent return on initial investment. With advanced suppressors, the same ROI can be obtained with fares of only 5.9 to 6.0 cents. At this level of ROI, this means that a fare increase of about 30 percent will be required if advanced suppressor technology is not available. These are fare increases beyond those that might be required even with advanced suppressor technology.

Another way to maintain the ROI at an acceptable level without raising the fare is to improve the load factor as performance deteriorates. This is illustrated in figure 15, where the two levels of suppressor technology are shown for the arrow wing configuration. The solid curves represent the ROI computed on the basis of a 40-percent initial down payment whereas the broken curves are for a 100-percent down payment. The fare is fixed at 5 cents per revenue-passenger-mile in this figure - a value approximating the fully diluted yield obtained by U.S. airlines on scheduled international flights. If yield does not improve from today's level, figure 15 shows that a load factor of about 58 percent will be required for an ROI of 20 percent (based on a 40-percent down payment) with advanced suppressor technology. This load factor is not too much greater than the 50-percent average now prevailing on the North Atlantic. With the competitive advantage of a lower block time, an SST could possibly sustain a load factor of 58 percent. With baseline suppressors, however, a load factor of about 75 percent is needed for this same level of ROI at current yield. This would be much harder to sustain. The probability of an increase in the actual fare is much greater if baseline suppressor technology is the best that is available and the F.A.R. 36 noise goal is to be met. The curves of figure 15 also show that if total payment is made on delivery of the airplane, instead of the 40-percent down payment assumed in the above calculations, ROI would be 10 percent instead of 20 percent at these load factors.

The airplane weight breakdown resulting from the airplanes being scaled for a constant 4200-nautical-mile range is shown graphically in figure 15 for the four airplanes listed in table IV. The structure of the arrow wing is as much as 4.5 percentage points higher than the modified delta, but the payload-to-gross weight percentage is, nevertheless, about half a point higher. The big change in payload percentage with either configuration occurs when suppressor technology is improved. The shift from baseline to advanced suppressors causes the payload percentage to jump almost 1.7 points. This occurs because the use of better suppressors reduces the engine size and weight requirements.

It is possible that by the mid-1980's, which is about the soonest these airplanes could enter commercial service, the noise levels of F.A.R. 36 will no longer be satisfactory. Perhaps levels that are 10 EPNdB below those of F.A.R. 36 will then be required. If this is the case, much higher gross weights and poorer economics will result for a dry-turbojet-powered SST than has been indicated for an F.A.R. 36 goal. Further calculations based on the data of this study indicate that an arrow wing airplane using turbojets with advanced suppressor technology would need a gross weight of 1 050 000 pounds to meet such a noise goal. This is about 65 percent more than the 636 000-pound TOGW calculated for the F.A.R. 36 goal at the same range and payload. The engine size requirement increases by about 150 percent to 1780 pounds per second for this 10 EPNdB reduction in the goal when the level of suppressor technology is fixed. Such an engine airflow requirement may dictate the use of a six-engine instead of a four-engine configuration. To obtain a 20-percent ROI with a 40-percent down payment (or a 10-percent ROI with full payment at delivery), figure 17 shows that a load factor of 88 percent is needed for a fixed fare of 5 cents per revenue-passenger-mile. (Compare this with the load factor of just 58 percent needed to get the same ROI at a 10 EPNdB higher noise goal.)

The arrow wing configuration may not be the best choice for a noise goal 10 EPNdB below F.A.R. 36. Perhaps the variable sweep configuration with its better takeoff and low-speed characteristics should be restudied at this lower noise level. In addition, as the noise goal is reduced, the need becomes more apparent for a better engine cycle. A cycle providing some bypass air with a resultant lower exit velocity at takeoff will be attractive from the noise standpoint. Takeoff performance must be improved, however, without seriously compromising supersonic cruise performance. This can best be accomplished by some degree of augmentation in the supersonic speed regime. Alternatives to this approach include various schemes to accomplish high bypass operation at takeoff with low bypass ratio at supersonic speeds. Such systems are likely to suffer in weight and complexity when compared to a simple turbojet. The turbojet, however, is not so lightweight and simple after including retractable multitube jet noise suppressors and oversizing to meet noise goals such as F. A. R. 36 minus 10 EPNdB. At any rate, further study is needed in both the cycle and the airframe areas when such low noise goals are contemplated.

CONCLUDING REMARKS

In this study, mission performance was calculated for an advanced light-weight dry turbojet cycle in three types of Mach 2.7 airframes both with and without the noise constraints of F. A. R. 36. The three airframe types were (1) the advanced arrow wing, (2) the modified delta with tail, and (3) the variable-sweep (arrow) wing. Each airplane was sized to carry 200 passengers. Takeoff wing loading was varied for each type since it affects the takeoff thrust-to-gross weight ratio required. This, in turn, affects the engine sizing when sideline noise is a constraint. The dry turbojet cycle had a design compressor pressure ratio of 10 and turbine inlet temperature of 2400° F. Component efficiencies, stage loadings, and weights were at levels commensurate with a 1975 engine definition.

Two levels of jet noise suppressor technology were studied. The first, called baseline technology, assumed that a reduction of 10 PNdB can be obtained with a given engine during takeoff at a 10-percent thrust loss. Its weight for a reference 900-pound-per-second turbojet was set at 3660 pounds. This level of technology is thought to be currently attainable. The second level, called advanced technology, assumed that a 15-PNdB reduction can be obtained with only a 7.5-percent thrust loss at half the baseline weight penalty.

This technology is thought to be attainable in a 1975 engine definition with considerable effort.

Although there was a significant range penalty for meeting the F.A.R. 36 sideline noise limit of 108 EPNdB, the results of this study indicate that an economically viable supersonic transport can be built to satisfy this noise requirement with a dry turbojet cycle using advanced technology jet noise suppressors. The selection of the proper airframe is most important. The advanced arrow wing concept had the greatest economic potential for airplanes designed for a total range of 4200 nautical miles (sufficient for Seattle-Tokyo, nonstop). With advanced suppressor technology, a takeoff gross weight of 636 000 pounds was required. With baseline suppressor technology, the gross weight increased to 864 000 pounds. Even more dramatic, perhaps, is the fact that engine design airflow requirements increased from 722 to 1288 pounds per second for this decrement in suppressor technology.

It is interesting to speculate about the passenger load factor needed to produce a postulated acceptable return on investment at current diluted international fares. For the best airplane with advanced technology suppressors, a load factor of only 58 percent would be required. With the baseline suppressors, a load factor of 75 percent will be required. The average prevailing on North Atlantic routes is now approximately 50 percent. A Mach 2.7 airplane, by virtue of its reduced block time compared to subsonic jets, could possibly generate a greater load factor and thus sustain profitability without an increase in fare if advanced suppressor technology is available. The 75-percent load factor required for baseline suppressors will be harder to sustain and may require a fare increase.

F.A.R. 36 noise levels may no longer be acceptable in the time period of the mid-1980's when these airplanes would be introduced into commercial service. There is speculation that the maximum permitted noise level will be 10 EPNdB below that of F.A.R. 36. If this is the case, much higher gross weights and poorer economics will result for a dry-turbojet-powered airplane than has been indicated for an F.A.R. 36 goal. Even with advanced suppressor technology it is necessary to raise the gross weight by 65 percent to

1 050 000 pounds to meet such a noise goal with the arrow wing configuration that was optimum for F. A. R. 36. The engine design airflow requirement will increase by almost 150 percent. The load factor must increase from 58 percent to 88 percent if a constant ROI is to be maintained without a fare increase. The arrow wing configuration that was optimum at F. A. R. 36 may no longer be optimum at the lower noise level, however. A concept with better takeoff characteristics than the arrow wing may yield better mission results. Better noise suppressors and/or a better engine cycle are also needed if fare increases are to be avoided. If F. A. R. 36 minus 10 EPNdB is to be the goal, a cycle providing some bypass air with a resultant lower exit velocity at takeoff will probably be more attractive. Further study in these areas is needed.

When takeoff wing loading was varied in the F. A. R. 36-noise-constrained mission studies, the best performance was obtained at a lower loading than the one which was best for no noise constraint. Although no wing loading optimization was made at F. A. R. 36 minus 10 EPNdB, it is likely that a still lower value would have resulted. The variable-sweep concept had the best takeoff characteristics at the highest wing loading and could thus meet the noise constraint with the least penalty. It was a poor third, though, compared with the other two configurations at F. A. R. 36 in terms of range/payload and economics. Its heavier structure and poorer supersonic aerodynamic efficiency detracted from its attractive low-speed characteristics. The modified delta configuration was competitive with the arrow wing at F. A. R. 36 when baseline suppressors were used. As improvements were assumed for suppression, however, the superiority of the arrow wing become more dominant.

APPENDIX - SYMBOLS

A	cross-sectional area, ft^2
C_D	drag coefficient
C_f	friction coefficient
C_L	lift coefficient
C_{L_o}	lift coefficient where $C_{D_{min}}$ occurs
D	drag, lb
DOC	direct operating cost, cents/seat-statute mi
F_n	net thrust, lb
i	true annual interest rate on loan
K	constant
L	lift, lb
$L_{F.A.R.}$	F.A.R. takeoff field length, ft
M	amount of each annual loan payment, dollars
N	total number of annual loan payments
N_c	compressor shaft speed, rpm
P	total amount borrowed, dollars
P_3/P_2	compressor pressure ratio
q	dynamic pressure, lb/ft^2
Re	Reynolds number at free-stream conditions
ROI	return on investment, percent
S	projected wing or tail planform area or body surface area, ft^2
sfc	specific fuel consumption, lb/hr of fuel per lb of net thrust
TOGW	takeoff gross weight, lb

T_4	turbine rotor inlet temperature, ^{o}F
V	airplane velocity, knots
W	weight, lb

Subscripts:

des	design
f	friction
g	gross, takeoff
ht	horizontal tail
i	induced, due to lift
ic	incompressible flow
L.O.	lift-off
min	minimum
p	pressure or wave
sls	sea-level, static
vt	vertical tail
3.5	at a range of 3.5 n. mi. from start of takeoff roll

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TABLE I. - CHARACTERISTICS OF UNAUGMENTED TURBOJET PROPULSION SYSTEM AT TWO FULL-THRUST CONDITIONS

Characteristic	Mach No./Altitude, ft	
	0/0	2.7/≧59 500
Inlet pressure recovery	0.960	0.916
Corrected shaft speed, percent design	100.0	73.7
Actual shaft speed, percent design	100.0	100.0
Corrected airflow at compressor face, percent design	100.0	56.9
Compressor pressure ratio	10.00	4.23
Compressor adiabatic efficiency	0.850	0.872
Combustor efficiency	0.985	0.985
Pressure ratio across combustor	0.944	0.935
Turbine rotor-inlet temperature, ^{o}F	2400	2400
Turbine adiabatic efficiency	0.916	0.912
Pressure ratio across nozzle (complete expansion)	3.867	36.6
Nozzle gross thrust (velocity) coefficient	0.992	0.990
Nozzle total temperature, ^{o}F	1882	1916
Nozzle throat area, percent design	100.0	98.4

TABLE II. - TYPICAL AERODYNAMIC RESULTS FOR RANGE-OPTIMIZED AIRPLANES SIZED FOR F. A. R. 36 SIDELINE NOISE WITH 10 PNdB JET SUPPRESSION

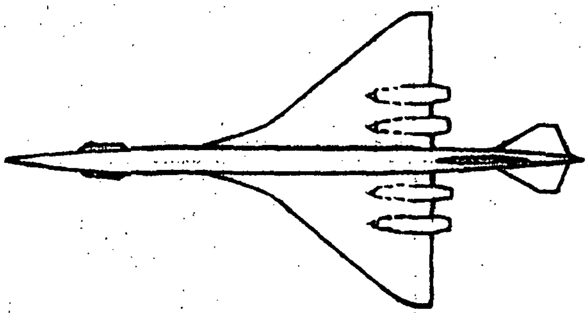
Aerodynamic characteristic	Arrow wing	Modified delta with tail	Variable sweep
Cruise L/D at Mach 2.7	9.7	7.8	7.3
Optimum initial cruise altitude, ft	66 000	64 500	59 500
C_L at lift-off	0.51	0.70	1.10
Initial climb L/D after takeoff, altitude = 767 ft	6.5	7.5	9.0

TABLE III. - AIRFRAME WEIGHT COMPARISON

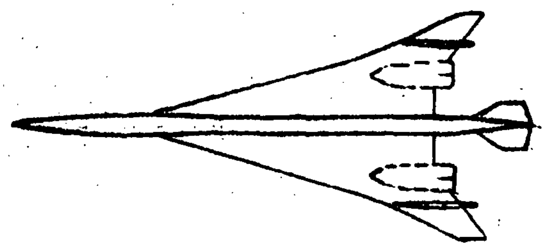
Component	Arrow wing, lb	Modified delta with tail, lb	Variable sweep, lb
Wing	85 000	81 000	124 000
Body	52 000	41 000	41 000
Tail - horizontal	2 100	4 800	4 800
Tail - vertical	4 900	4 800	4 800
Landing gear	28 000	28 000	28 000
Fixed equipment	54 600	54 600	54 600
Tolerance, standard and operational items and airline options	15 300	15 300	15 300
Airframe weight (total)	241 900	229 500	272 500
Takeoff wing loading, lb/ft^2	80.9	97.5	151
Takeoff gross weight, lb	750 000	750 000	750 000

TABLE IV. - CHARACTERISTICS OF BEST AST'S SCALED FOR CONSTANT RANGE WITH TWO LEVELS OF JET NOISE SUPPRESSOR TECHNOLOGY

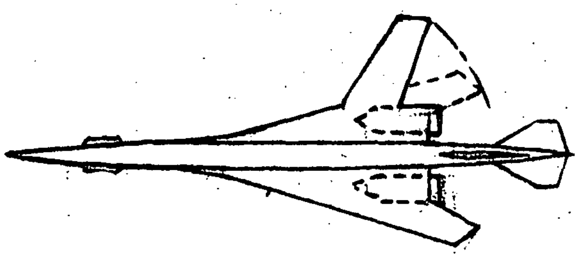
	Modified delta with tail		Arrow wing	
Jet noise suppressors ΔPNdB/ΔF_n/Weight	10/10%/ baseline	15/7.5%/ advanced	10/10%/ baseline	15/7.5%/ advanced
TOGW, lb	970 000	688 000	864 000	636 000
Takeoff Wg/S, lb	80	90	60	60
sls airflow per engine, lb/sec	1338	750	1288	722
No. of passengers	200	200	200	200
Total design range, n. mi.	4200	4200	4200	4200
L/D at Mach 2.7 cruise	8.09	7.51	9.95	9.32
Initial cruise alt., ft	64 000	60 000	67 000	66 000
sfc at Mach 2.7 cruise, hr^{-1}	1.385	1.390	1.385	1.393
F.A.R. takeoff field length on +15° C day, ft	10 747	11 640	10 212	10 212
Max. sideline noise, EPNdB	108	108	108	108
Lift-off Mach no./Fn/Wg	0.270/0.271	0.287/0.283	0.274/0.295	0.274/0.295
3.5 n. mi. alt., ft/Mach no.	1500/0.299	1500/0.315	1500/0.306	1500/0.306
Climb rate at 3.5 n. mi. for 108 EPNdB community noise (std. day), ft/min	2680	2570	2750	2250
Min. acceptable climb rate by F.A.R. 36 to meet 108 EPNdB at 3.5 mi., ft/min	880	935	1050	1050
DOC, cent/seat-s. mi.	2.137	1.425	1.937	1.327
ROI (at 40% down) with fare = 7¢/rpm and 50% L.F., %	0	50	13	55
Load factor required for 20% ROI (at 40% down) fare = 5¢/rpm, %	N/A	N/A	75	58



MODIFIED DELTA WITH TAIL



ADVANCED ARROW WING



VARIABLE SWEEP WING

FIGURE 1. – SST TYPES CONSIDERED IN THIS STUDY.

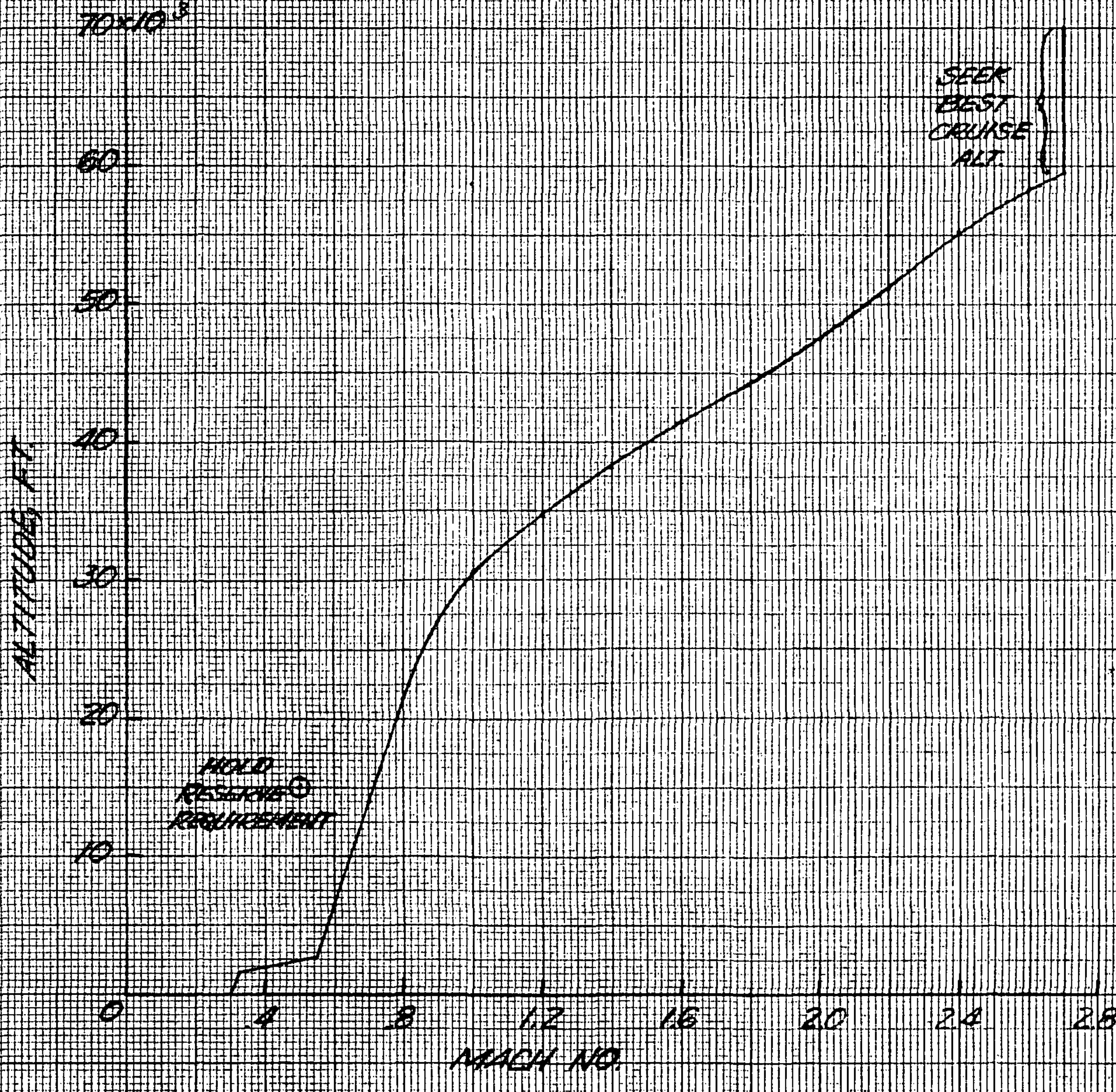


FIGURE 2.-MACH NO.-ALTITUDE SCHEDULE USED FOR STANDARD DAY PERFORMANCE CALCULATIONS IN THIS STUDY.

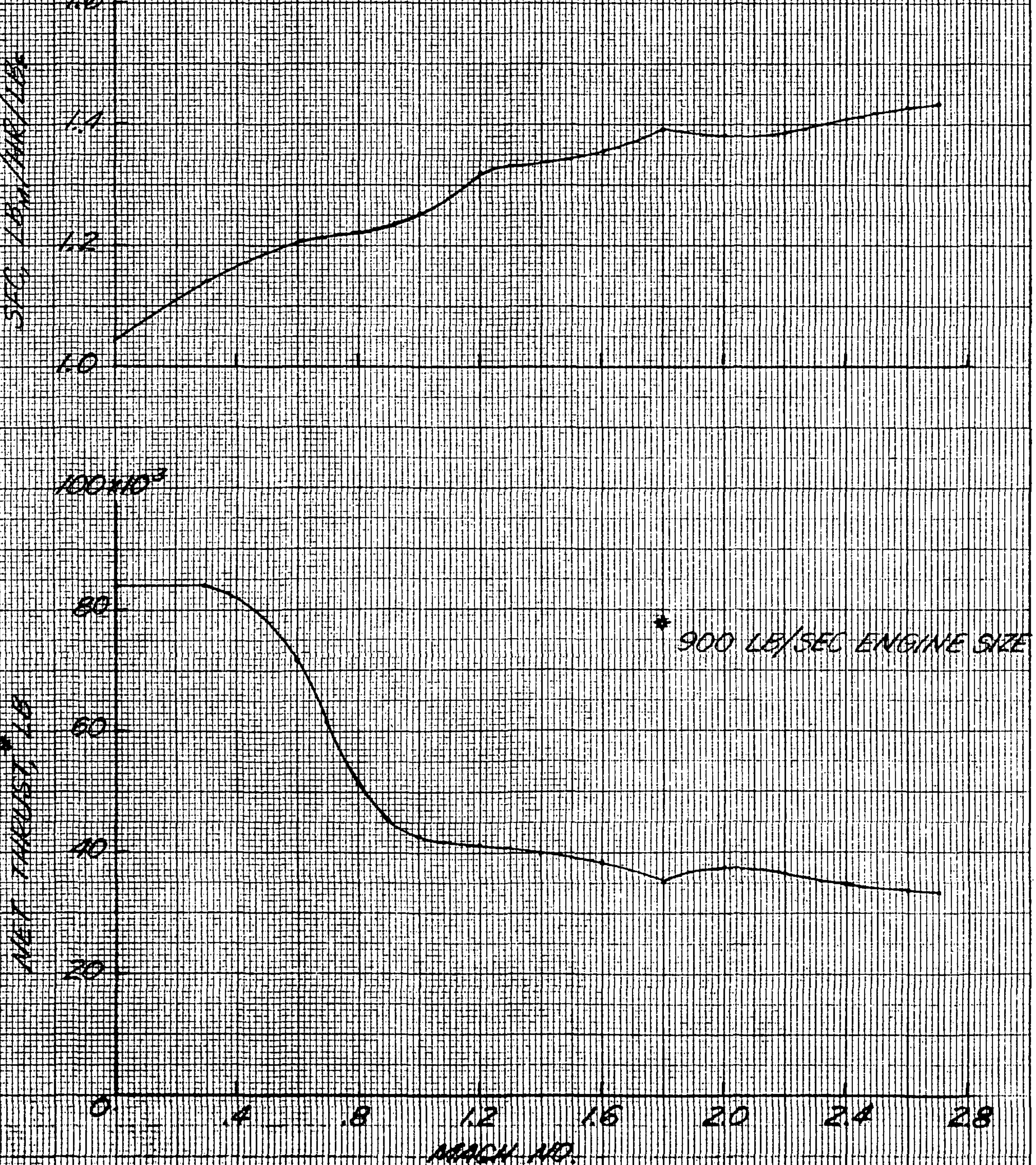
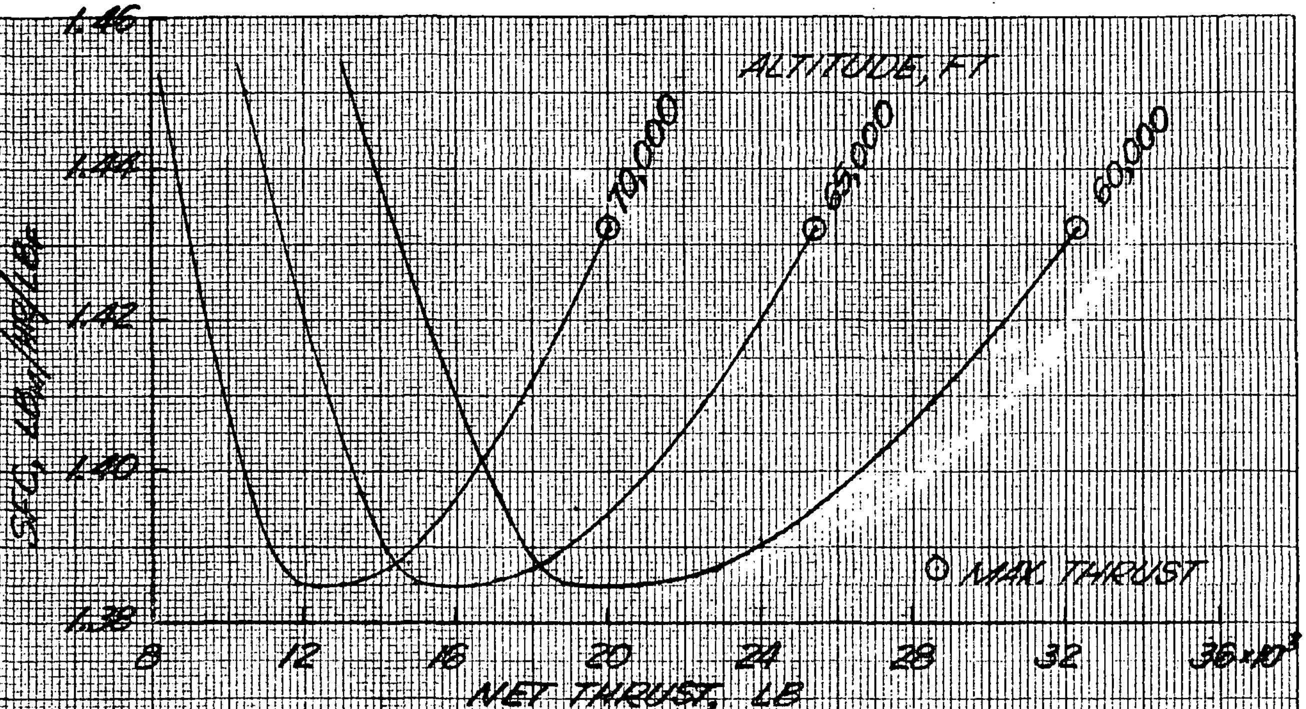
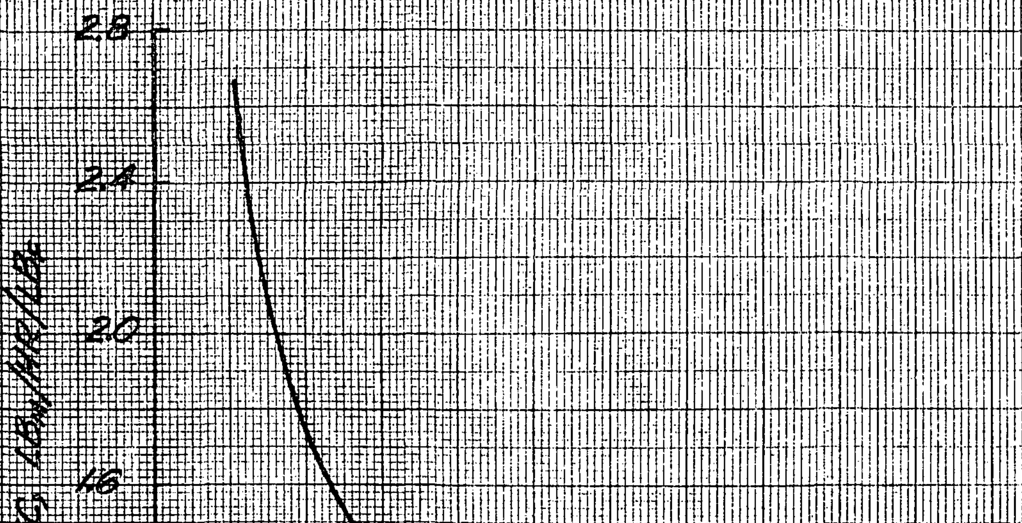


FIGURE 3 - INSTALLED, FULL POWER PERFORMANCE ON STANDARD DAY OVER REFERENCE FLIGHT PATH. DRY TURBOJET WITH $T_4 = 2400°F$, $(P_3/P_2)_{SLS} = 10$.



(A) MACH 2.7 CRUISE, $N_c = N_{c_{DESIGN}}$



(B) HOLD AT MACH 0.5, ALTITUDE = 15,000 FT, $N_c < N_{c_{DESIGN}}$

FIGURE 4-INSTALLED CRUISE PERFORMANCE ON STANDARD DAY. DRY TURBOJET WITH DESIGN AIRFLOW = 900 LB/SEC. $T_{4_{MAX}} = 2400°F$, $(P_3/P_2)_{SLS} = 10$.

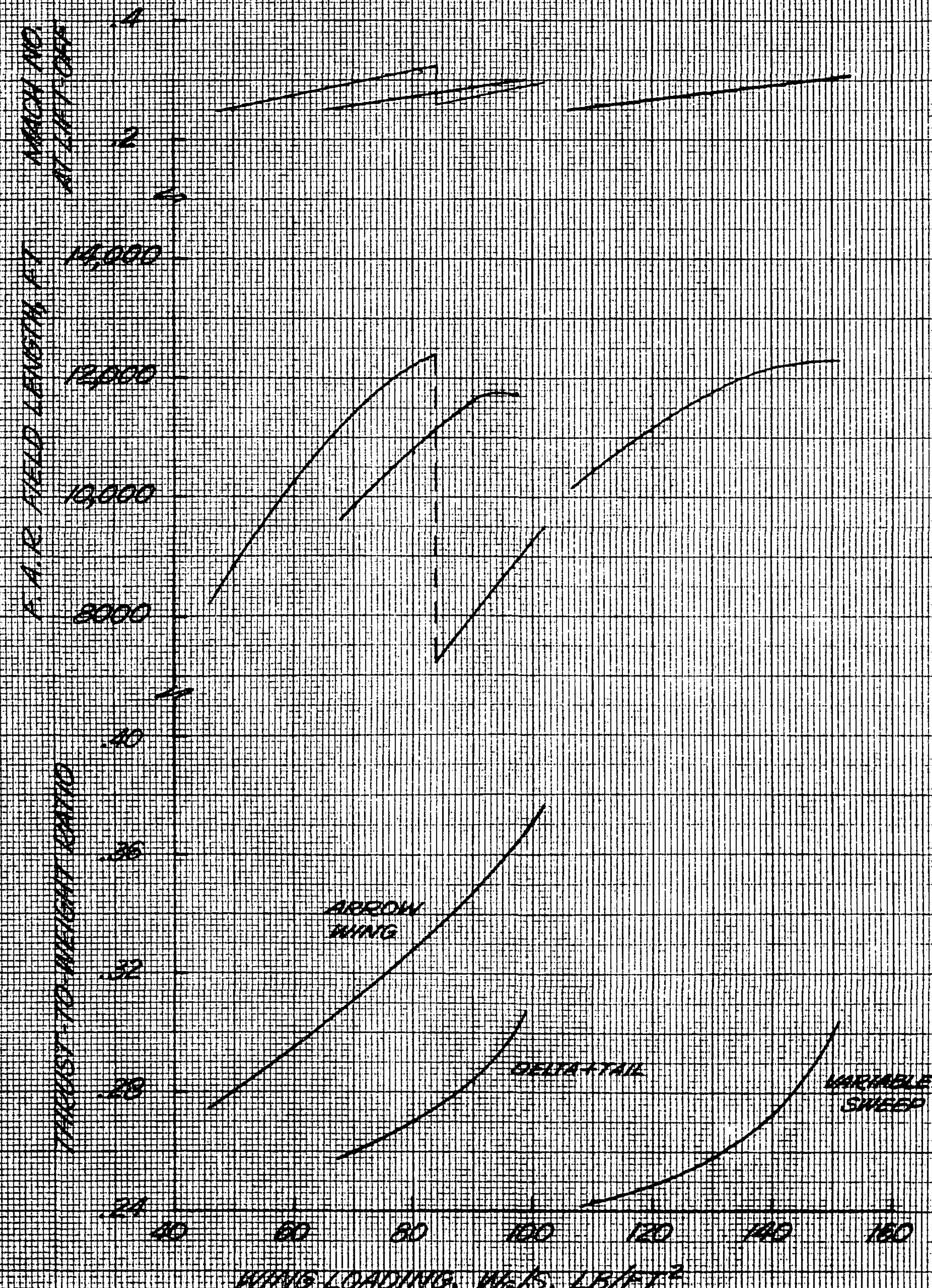
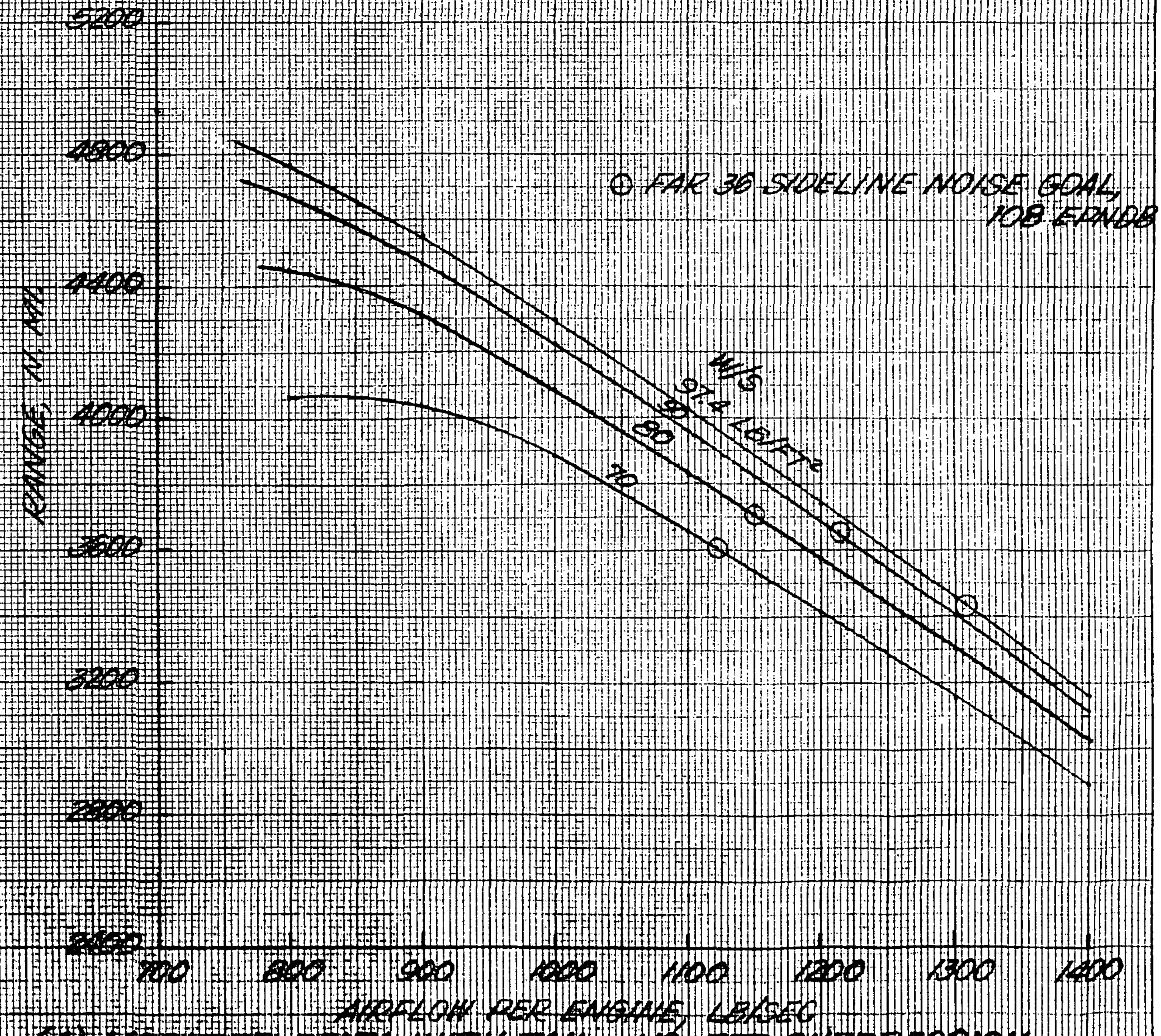
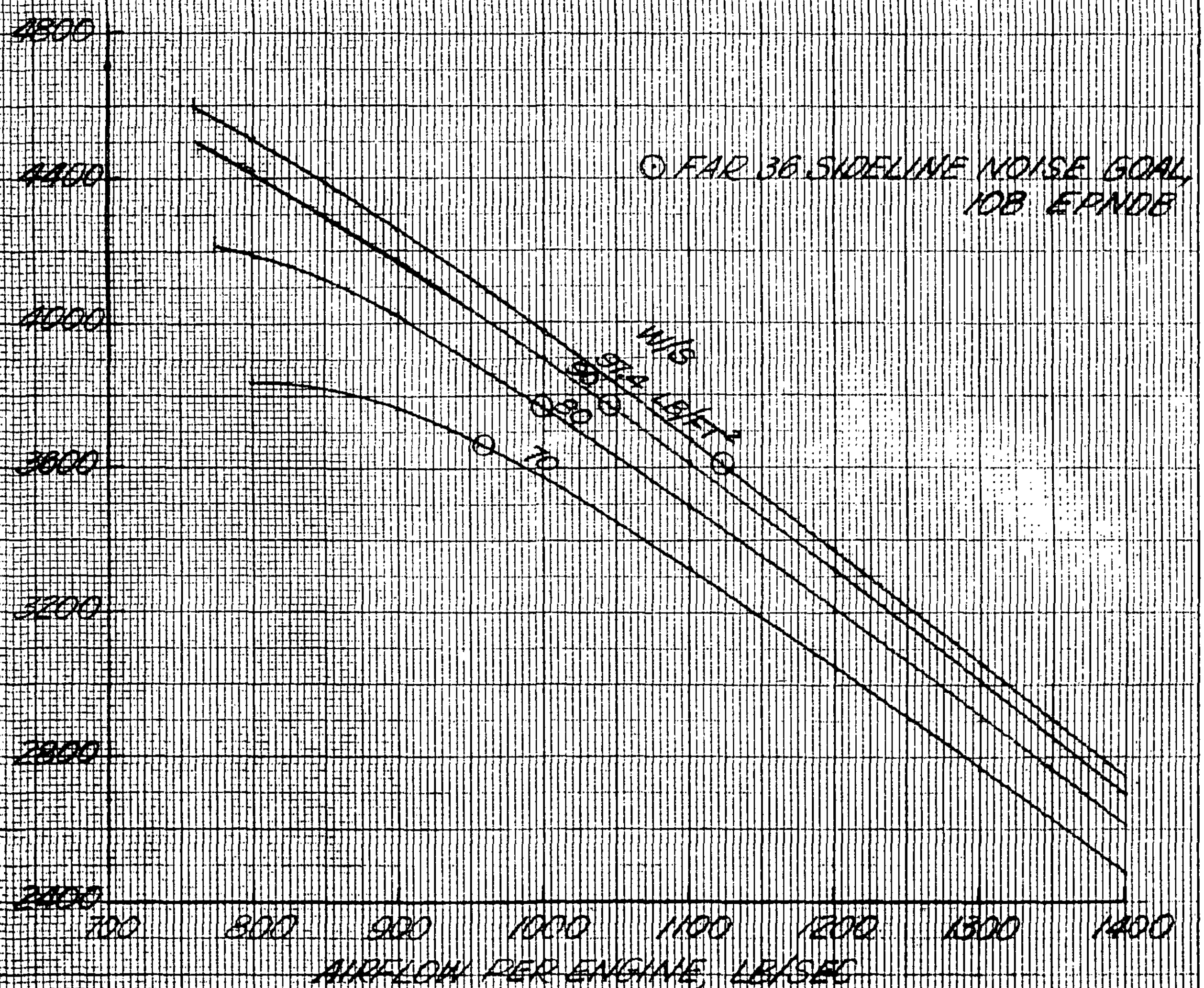


FIGURE 5 – AIRPLANE TAKEOFF CHARACTERISTICS WITH THRUST CAPABILITY TO CLIMB TO 1600 FT AT A POINT 3.5 N.MI FROM START OF TAKEOFF ROLL.



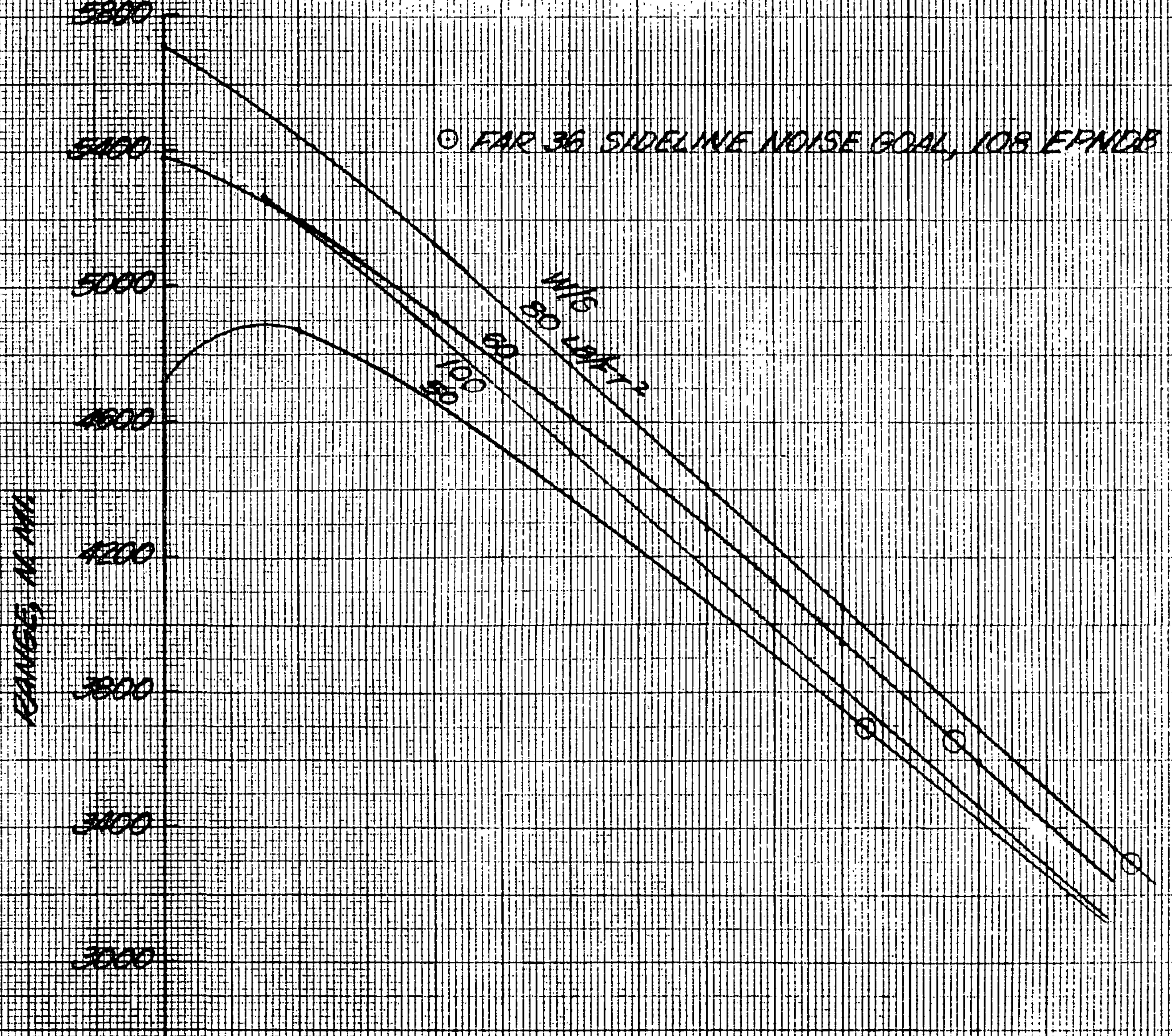
(A) MODIFIED DELTA WITH TAIL. NO JET SUPPRESSION.

FIGURE 5. - EFFECT OF ENGINE SIZE ON SST RANGE.
TOGW, 750 000 LB; PAYLOAD, 40 000 LB; DRY TURBOJET CYCLE.



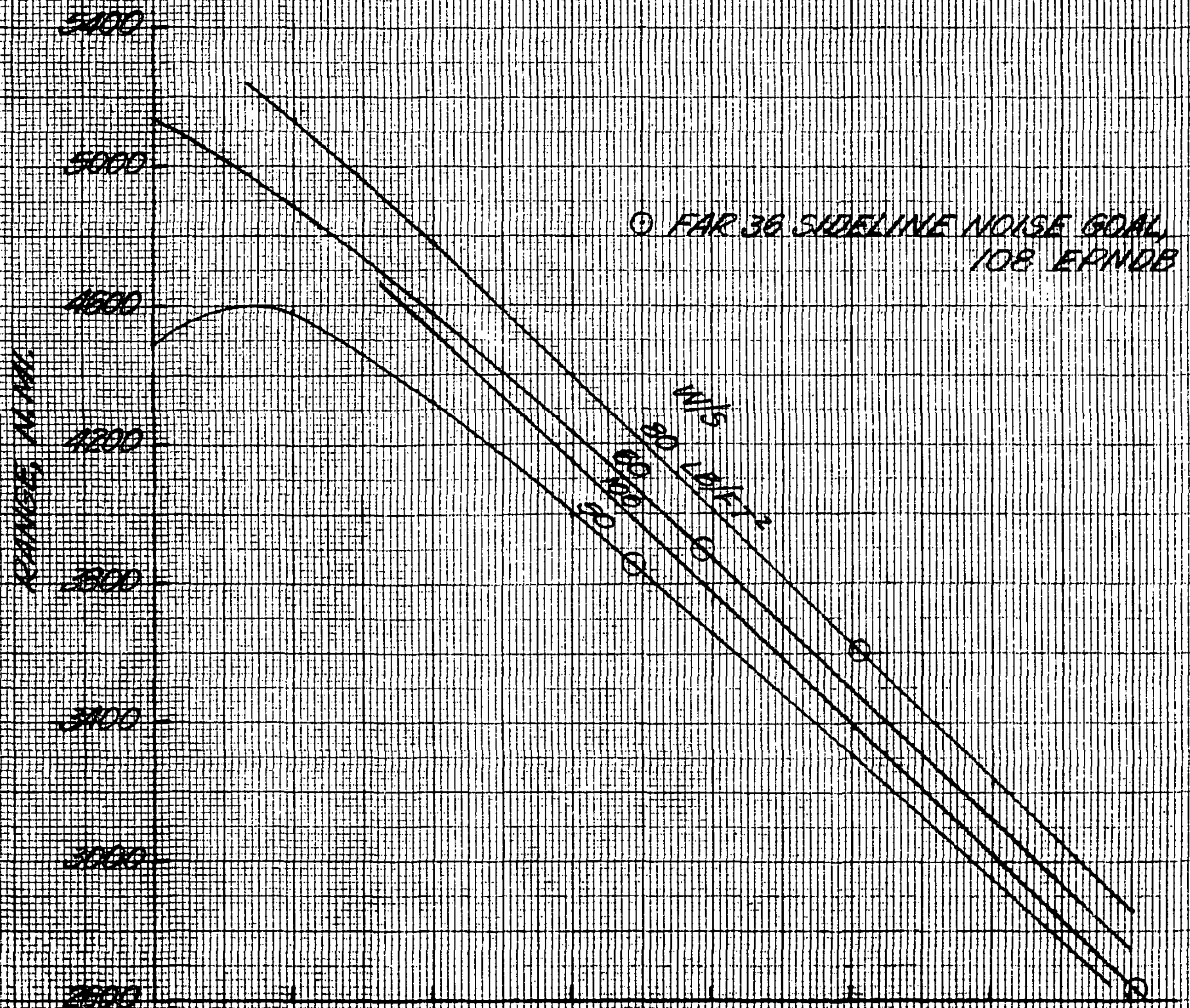
(B) MODIFIED DELTA WITH TAIL. 10 PNDB BASELINE SUPPRESSORS.

FIGURE 6.—CONTINUED



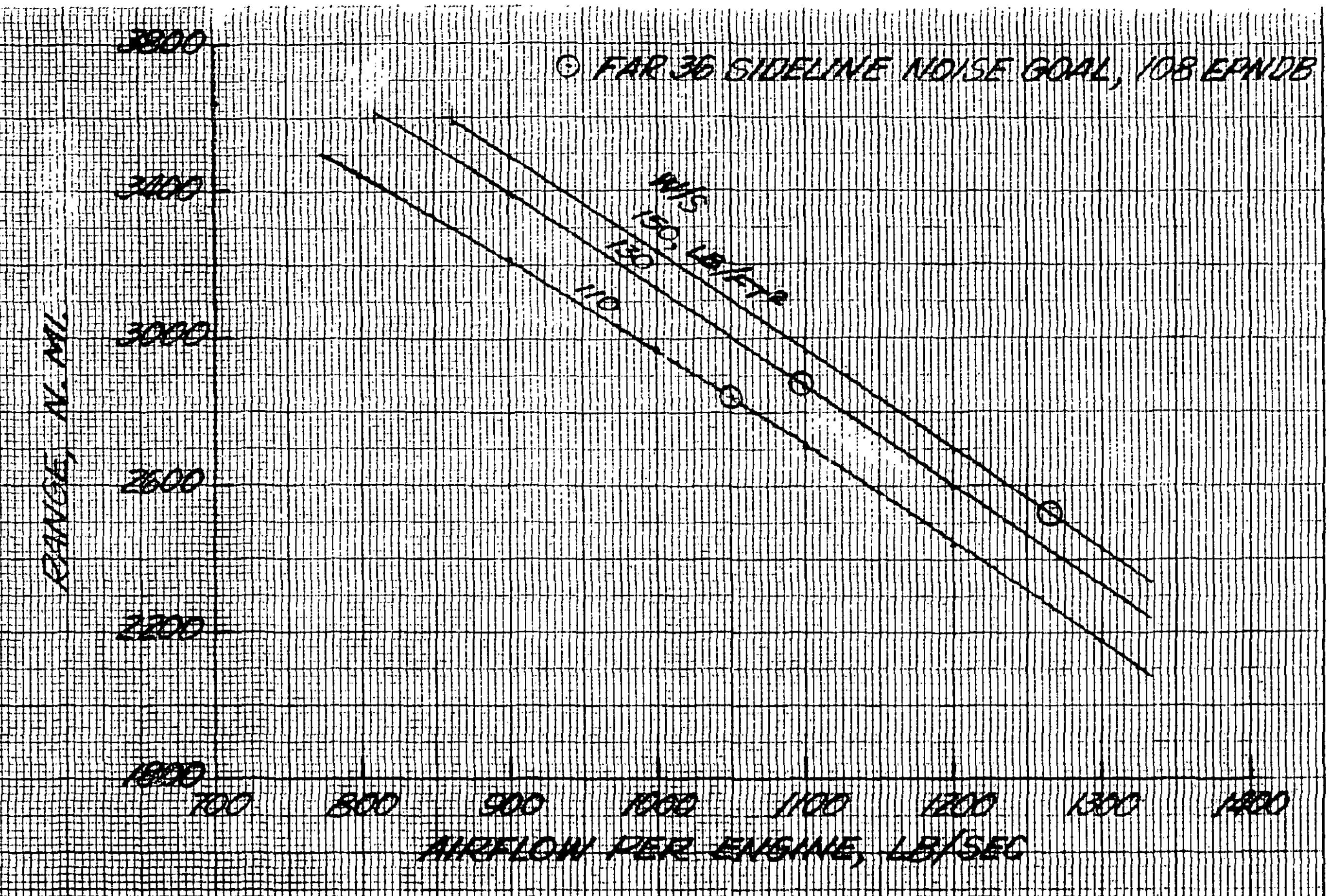
(C) ARROW WING AIRPLANES. NO JET SUPPRESSION.

FIGURE 6. - CONTINUED.

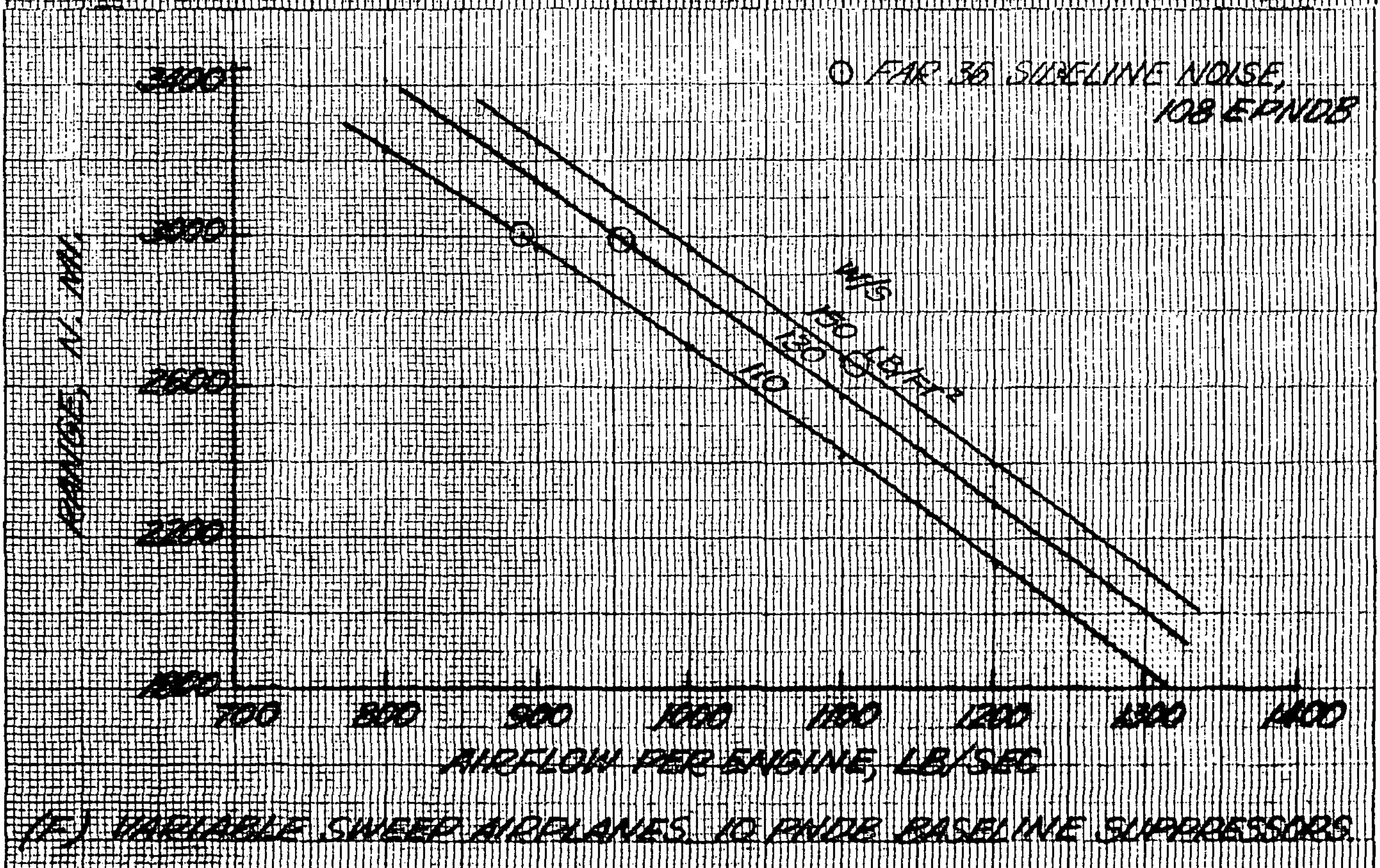


(D) ARROW WING AIRPLANES. 10 PNDB BASELINE SUPPRESSORS.

FIGURE 6. – CONTINUED.



(E) VARIABLE SWEEP AIRPLANES, NO JET SUPPRESSION.



(F) VARIABLE SWEEP AIRPLANES, 10 PNDB BASELINE SUPPRESSORS.

FIGURE 6.- CONCLUDED.

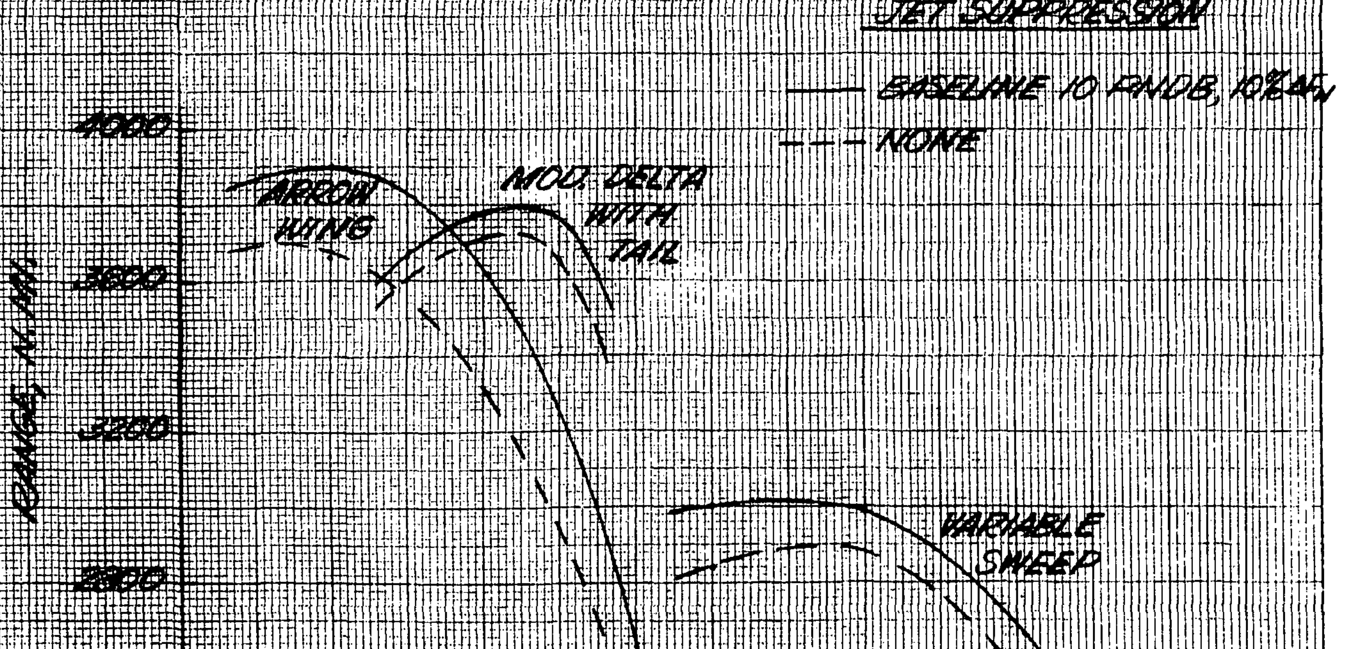


FIGURE 7. – EFFECT OF BASELINE JET NOISE SUPPRESSORS IN MEETING 108 EPNDB SIDELINE NOISE LEVEL. TOGW = 750,000 LB.

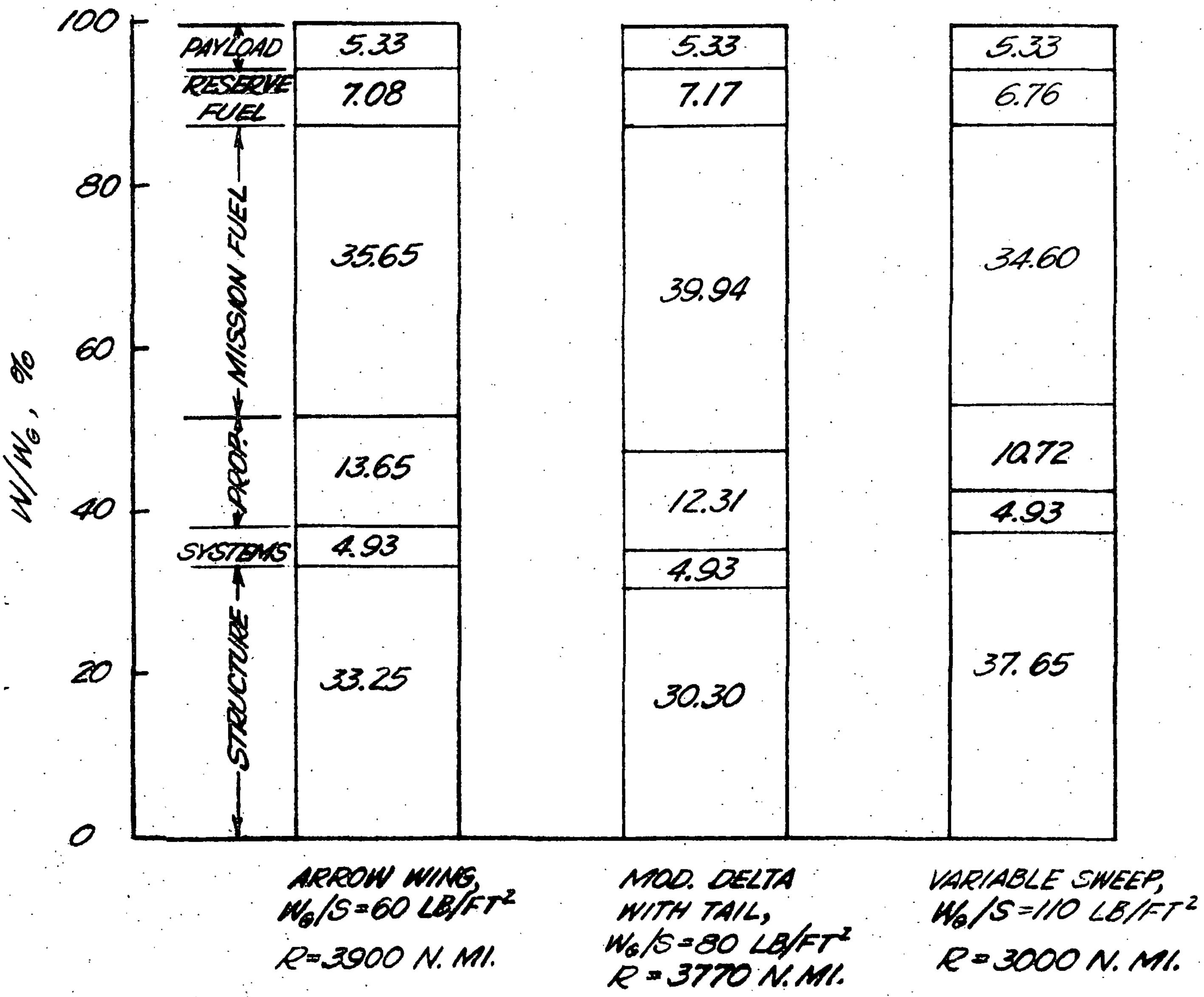


FIGURE 8.– WEIGHT BREAKDOWN FOR SST'S MEETING 108 EPNDB SIDELINE NOISE GOAL. DRY TURBOJET CYCLE WITH BASELINE SUPPRESSORS (10 PNDB SUPPRESSION, 10% THRUST LOSS). TOGW, 750 000 LB.

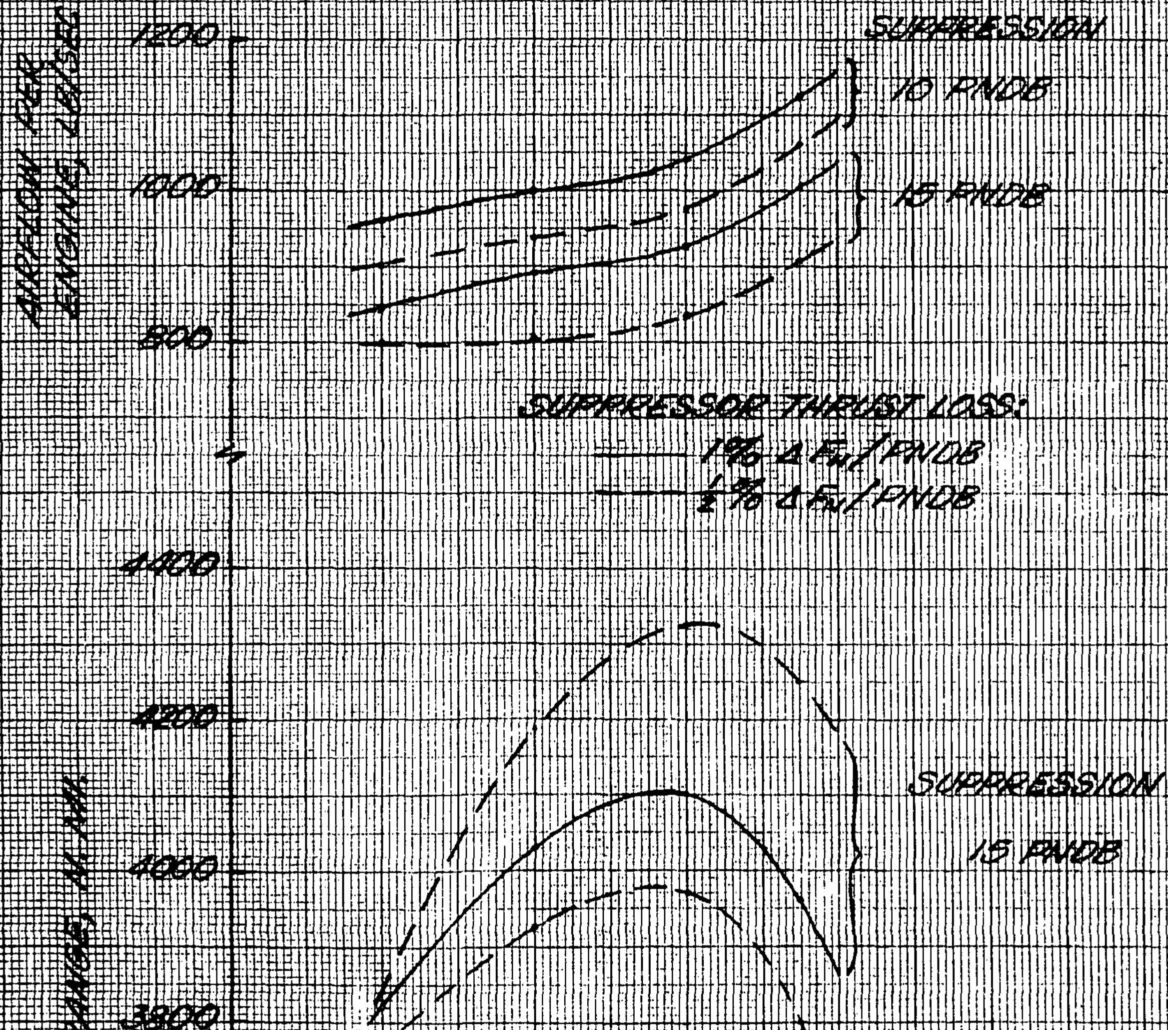


FIGURE 9. - EFFECT OF SUPPRESSOR THRUST LOSS ON RANGE AND ENGINE SIZE. SIDELINE NOISE, 108 EPNDB. DRY TURBOJET CYCLE. MODIFIED DELTA WITH TAIL. TOGW, 750 000 LB. BASELINE SUPPRESSOR WEIGHT SCHEDULE.

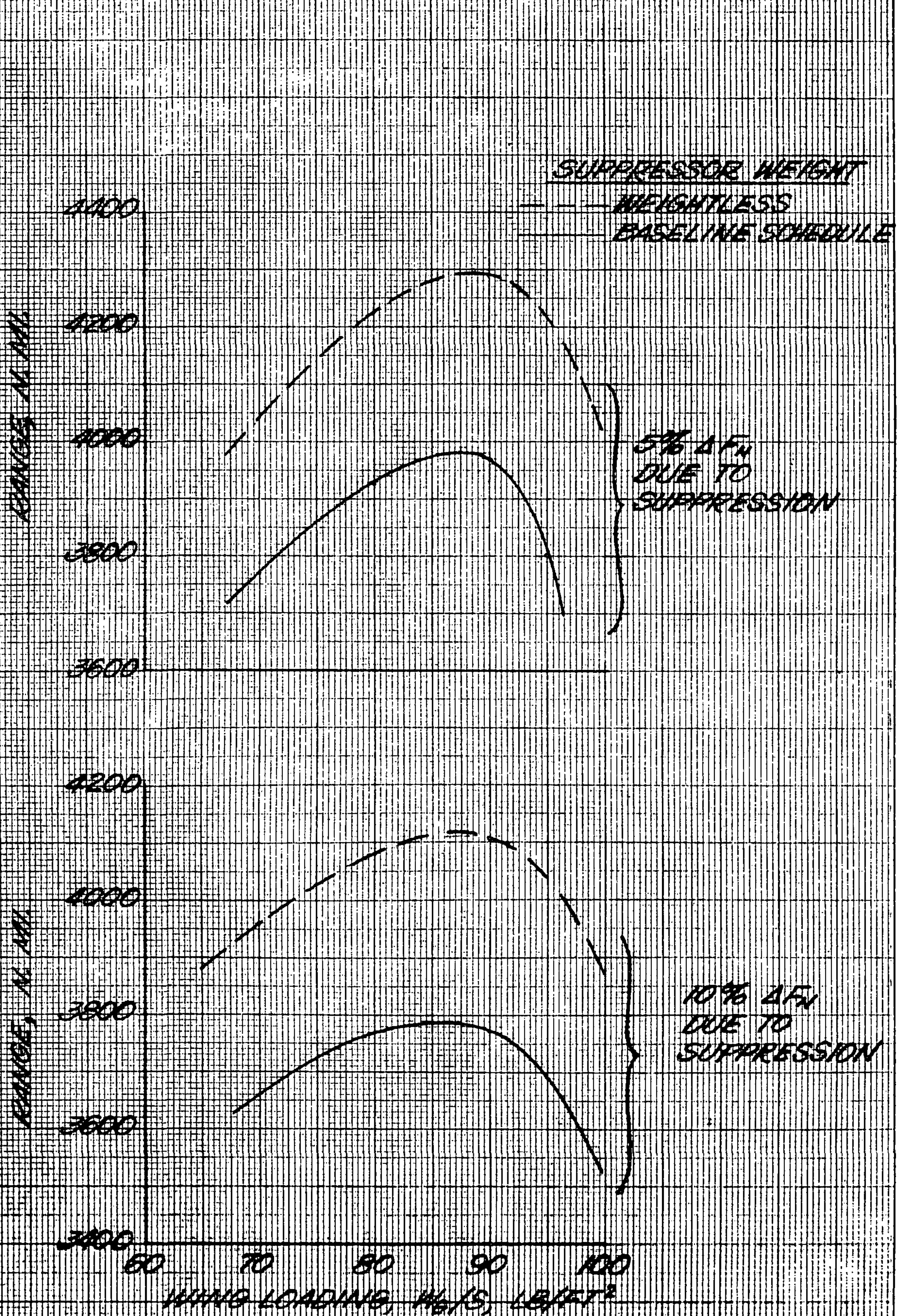


FIGURE 10 - EFFECT OF SUPPRESSOR WEIGHT ON RANGE. SIDELINE NOISE, 108 EPNDB. 10 PNDB SUPPRESSION. DRY TURBOJET CYCLE. MODIFIED DELTA WITH TAIL. TOGW, 750 000 LB.

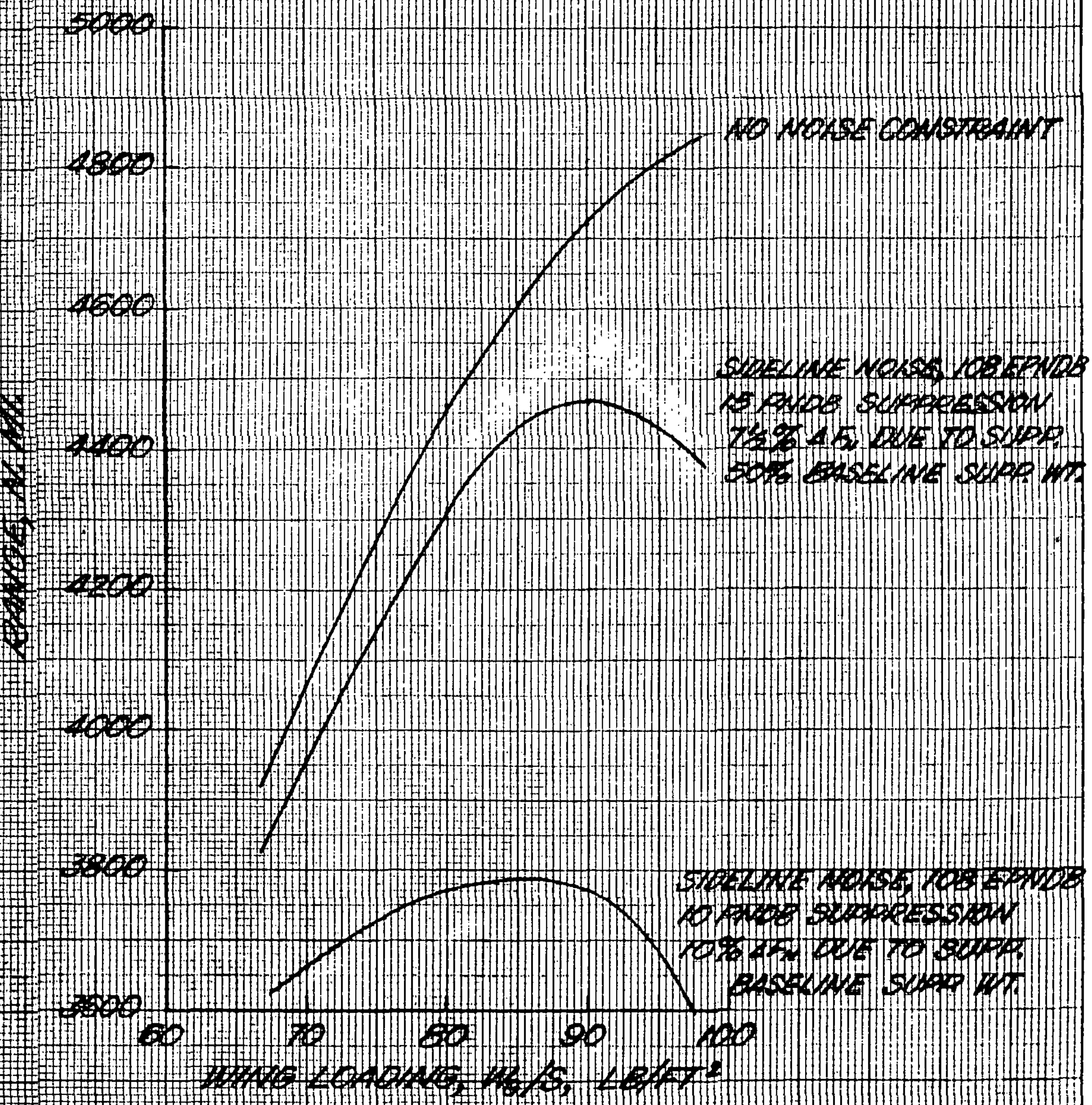


FIGURE 11.- EFFECT OF SUPPRESSOR TECHNOLOGY ON RANGE OF MODIFIED DELTA WITH TAIL AIRPLANES. TOGW, 750 000 LB. DRY TURBOJET CYCLE.

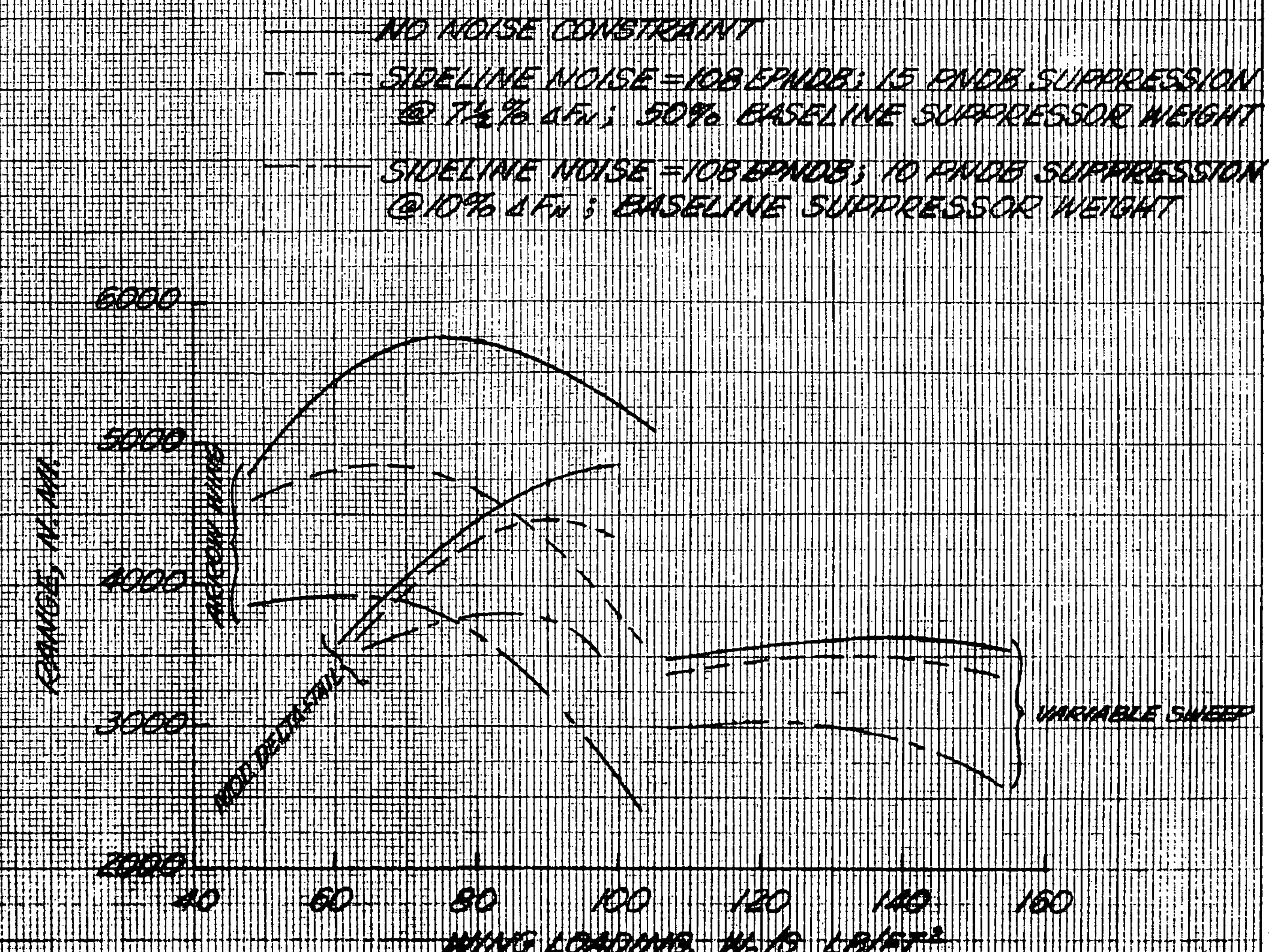


FIGURE 12 – EFFECT OF SUPPRESSOR TECHNOLOGY ON RANGE FOR THREE TYPES OF SST. TOGW, 750,000 LB. DRY TURBOJET CYCLE.

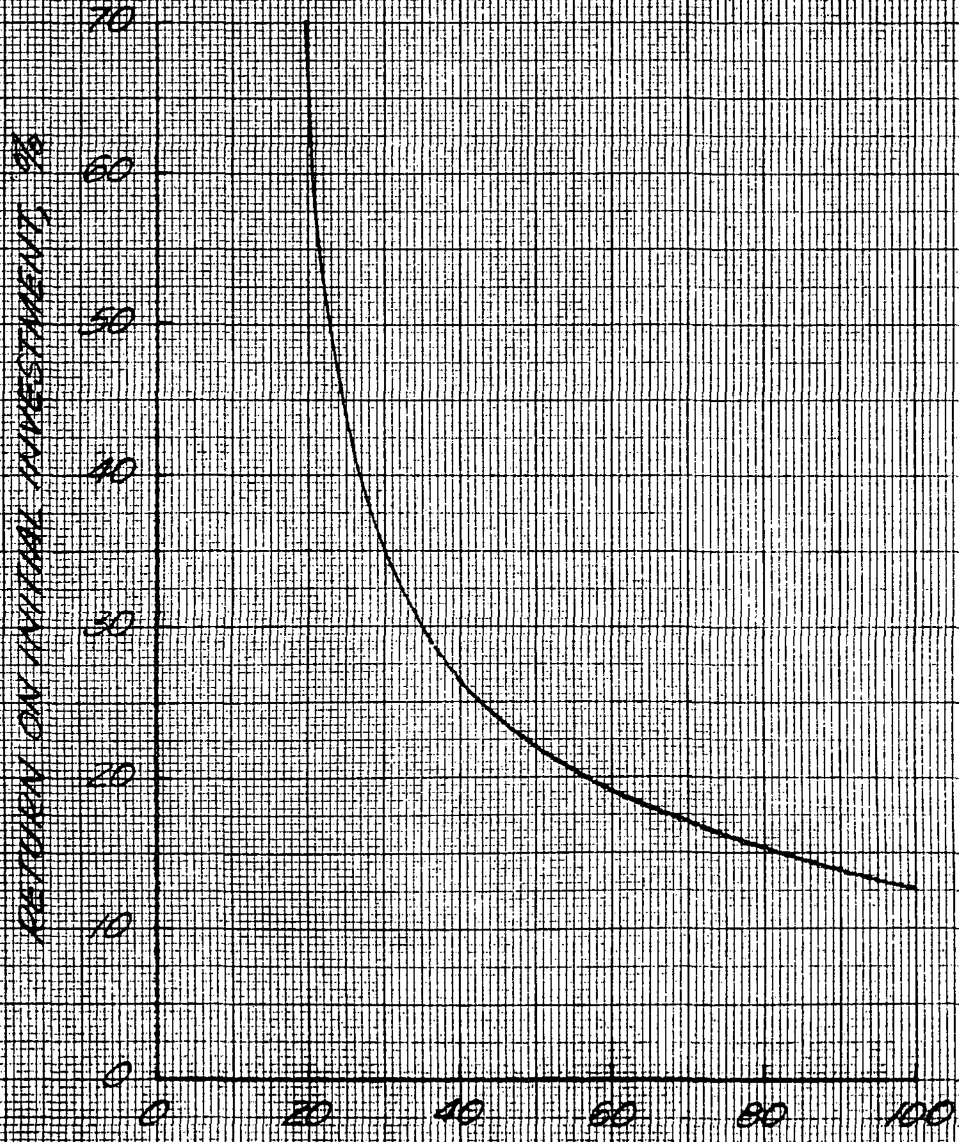


FIGURE 13. – EFFECT OF INITIAL DOWN PAYMENT ON R.O.I.; PURCHASE PRICE, $58.2×10^6; INTEREST ON LOAN, 10%; FARE, 5 CENTS / EQV.-PASS-ST.-MI.; LOAD FACTOR, 60%; ARROW WING AIRPLANE; WING LOADING, 60 LB/FT²; DRY TJ ENGINES WITH ADVANCED 15 PNdB SUPPRESSORS; RANGE, 4200 N. MI.; 200 SEATS; SIDELINE NOISE, 108 EPNdB.

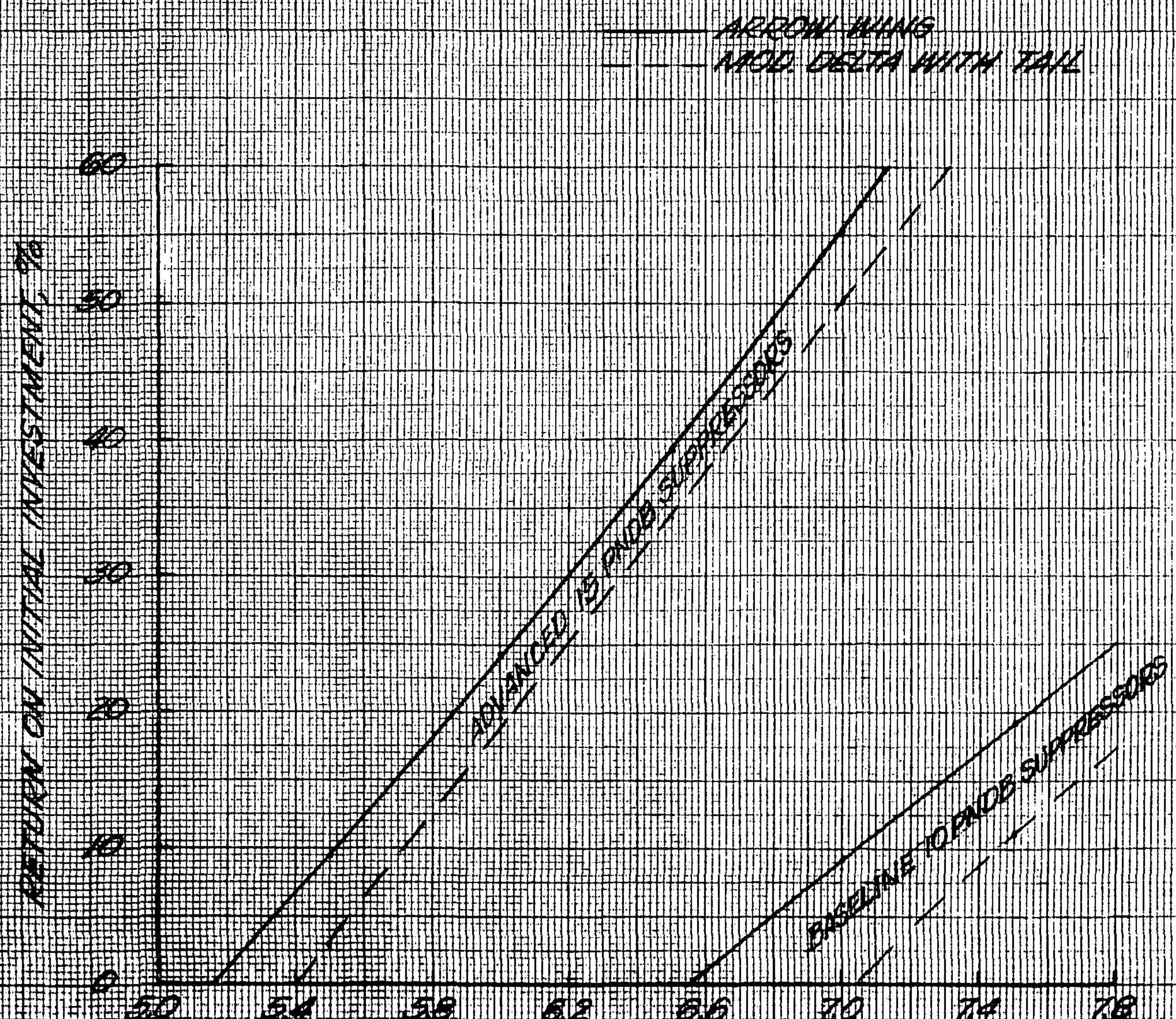


FIGURE 18. - EFFECT OF SUPPRESSOR TECHNOLOGY AND FARE ON ROI.
SIDELINE NOISE, 108 EPNDB. RANGE, 4200 N. MI. 200 SEATS.
LOAD FACTOR, 50%. DOWN PAYMENT, 40%. DRY TURBOJET CYCLE.

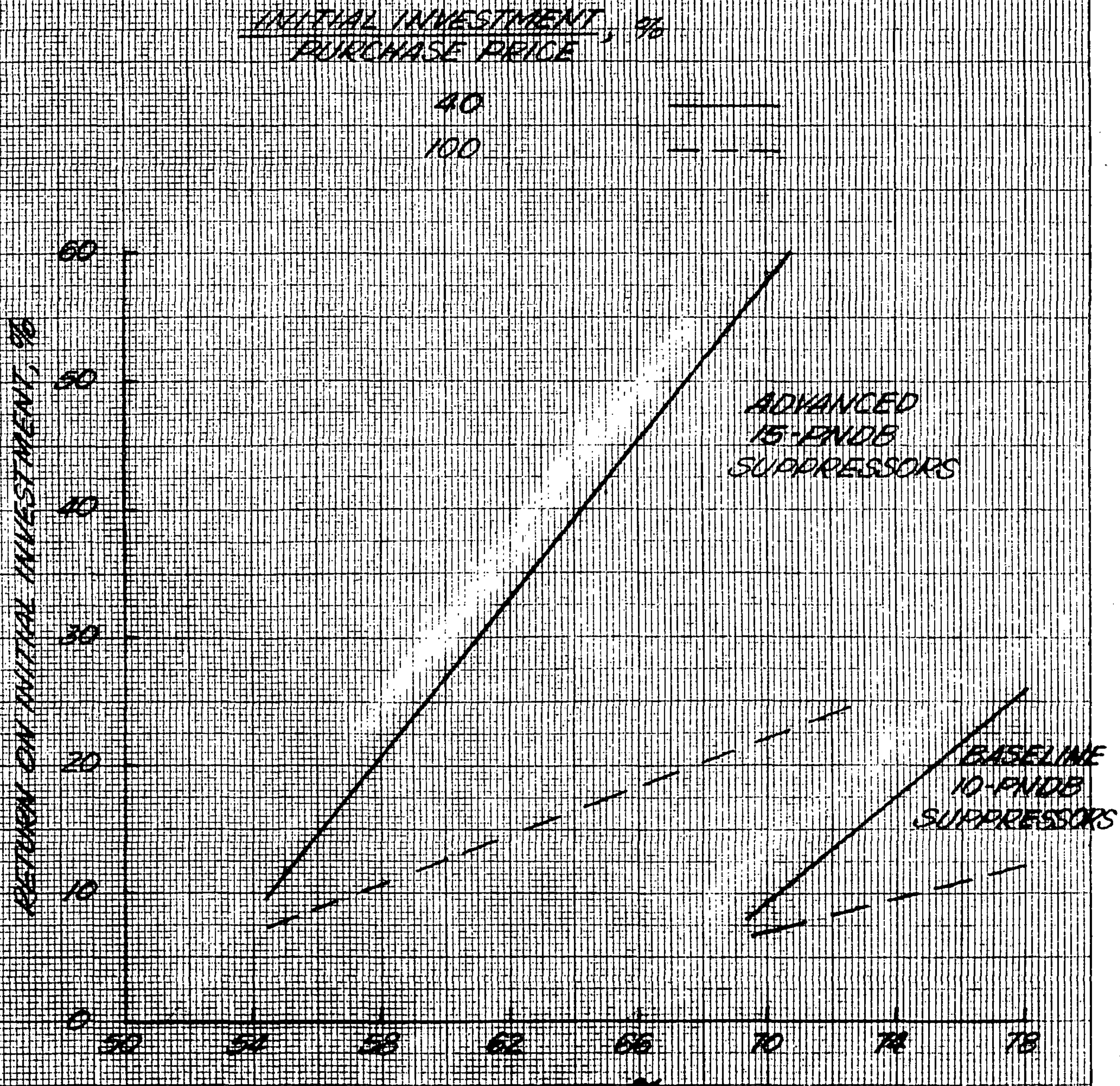


FIGURE 15. - EFFECT OF LOAD FACTOR ON ROI OF AN ARROW WING SST. SIDELINE NOISE, 108 EPNDB. RANGE, 4200 N.MI. 200 SEATS. FARE, 5 CENTS/REV-PASS.-S-MI. WING LOADING, 60 LB/FT^2. DRY TURBOJET CYCLE.

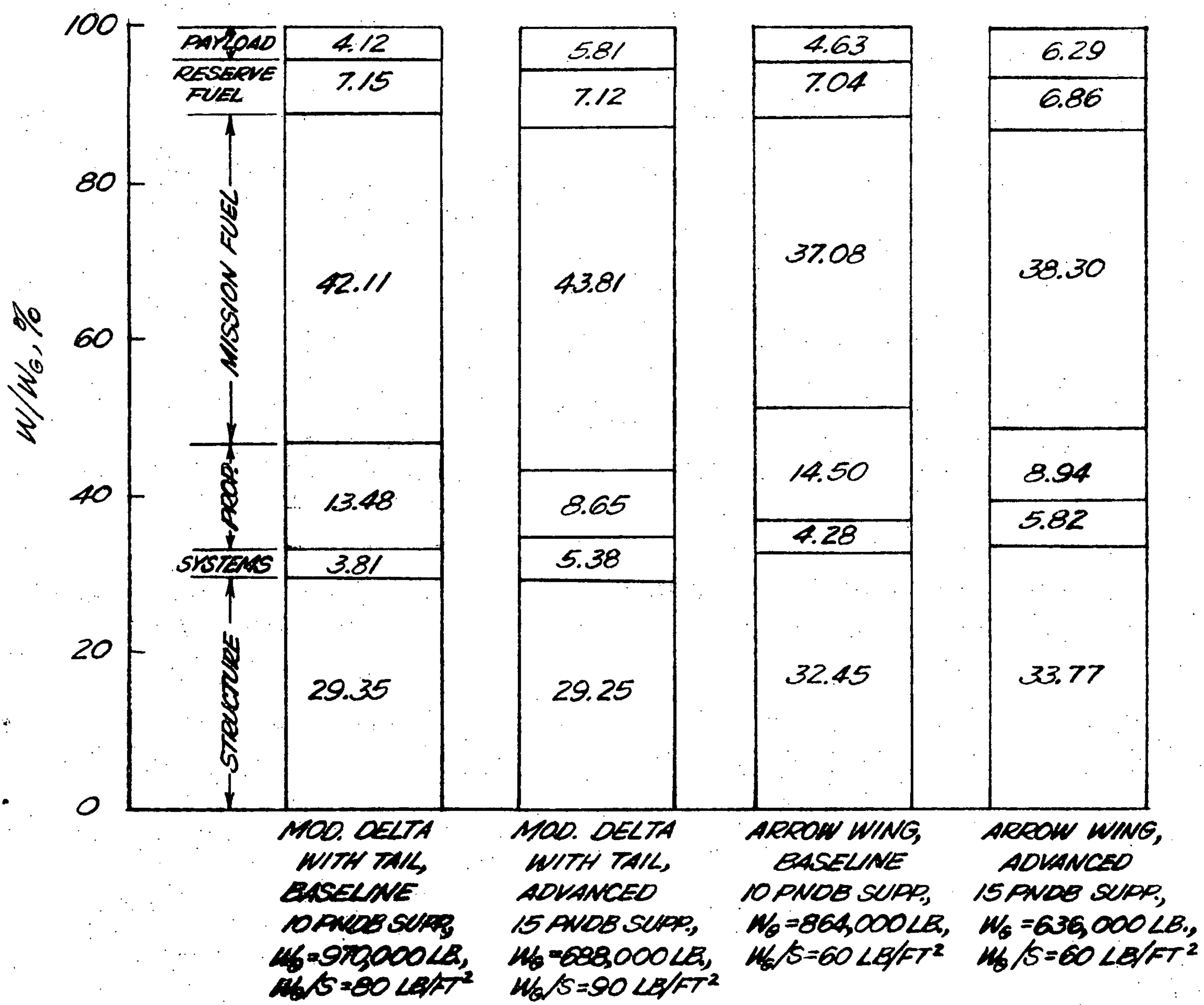


FIGURE 18.- SST WEIGHT BREAKDOWN FOR 4200 N. MI. DESIGN MISSION WITH 200 PASSENGERS. DRY TURBOJET ENGINES, $(P_3/P_2)_{DES}=10$, $T_{4_{DES}}=2400\,°F$. SIDELINE NOISE, 108 EPNDB.

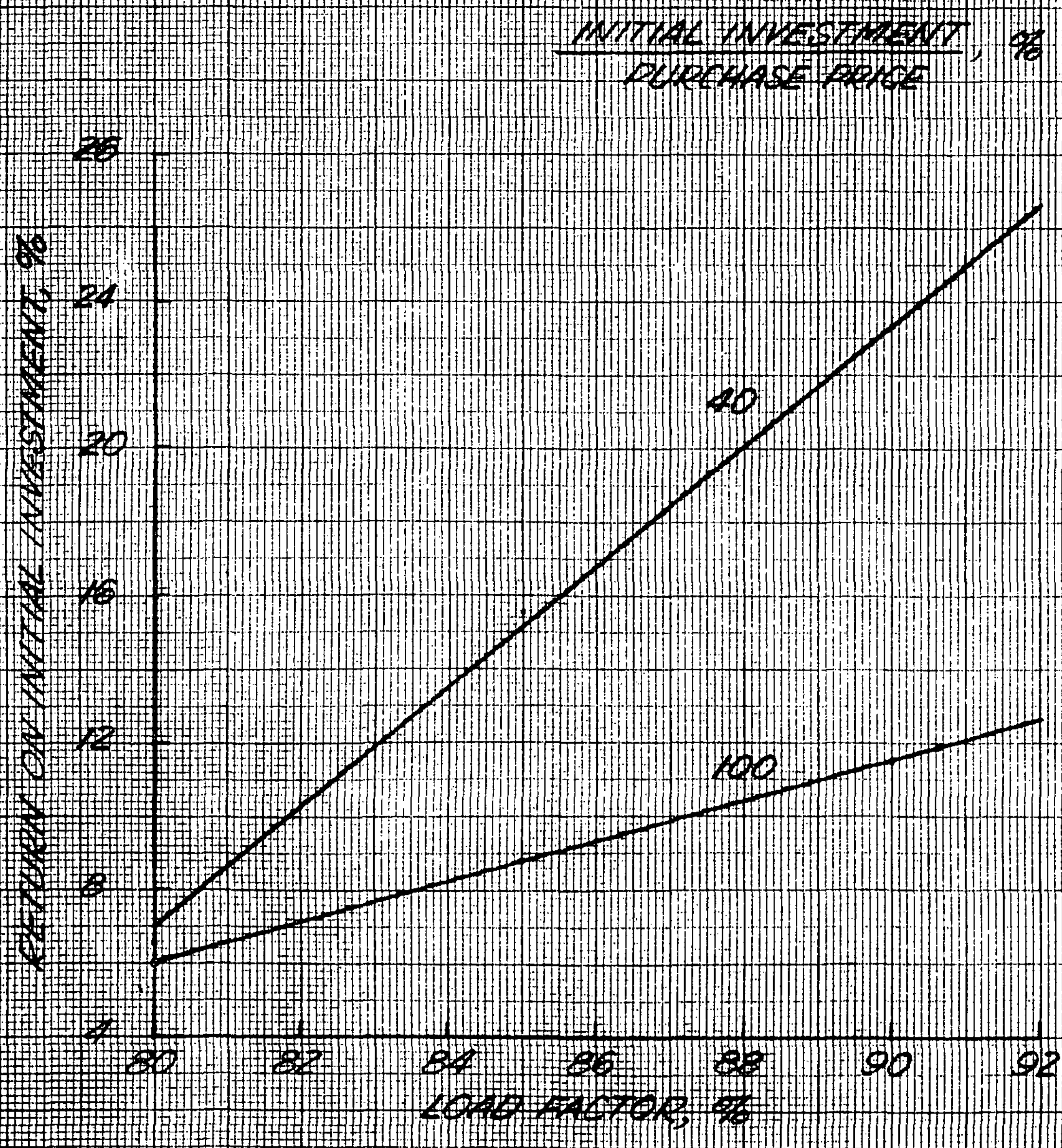


FIGURE 17. - EFFECT OF LOAD FACTOR ON R.O.I. WITH NOISE GOAL OF FAR 36-10 EPNdB (98 EPNdB). ARROW WING SST; WING LOADING, 60 LB/FT^2; RANGE, 4200 N. MI; 200 SEATS; FARE, 5 CENTS/REV-PASS.-S.-MI; PURCHASE PRICE, \$107.7×10^6 (INCLUDING SPARES). DRY TURBOJET CYCLE WITH ADVANCED 15 PNdB SUPPRESSORS.