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PRELIMINARY ANALYSIS OF LONG-RANGE AIRCRAFT DESIGNS FOR FUTURE HEAVY AIRLIFT MISSIONS

Walter P. Nelms, Jr.
Ames Research Center
Moffett Field, California 94035

Ronald Murphy
Air Force Flight Dynamics Laboratory
Wright Patterson Air Force Base
Ohio 45433

Alice Barlow
Computer Sciences Corporation
Mountain View, California 94043

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Walter P. Nelms, Jr.,* Ronald Murphy,** and Alice Barlow†

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*Ames Research Center, Moffett Field, CA 94035

**Air Force Flight Dynamics Laboratory, Wright Patterson Air Force Base, OH 45433

†Computer Sciences Corporation, Mt. View, CA 94043

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16. Abstract

A computerized design study of very large cargo aircraft for the future heavy airlift mission has been conducted by the NASA Ames Research Center, using the Aircraft Synthesis program (ACSYNT). The study was requested by the Air Force under an agreement whereby Ames provides computerized design support to the Air Force Flight Dynamics Laboratory. This effort is part of an overall Air Force program to study Advanced Technology Large Aircraft Systems (ATLAS). Included in the Air Force large aircraft program are investigations of missions such as heavy airlift, airborne missile launch, battle platform, command and control, and aerial tanker. The Ames studies have concentrated on large cargo aircraft of conventional design with payloads from 250,000 to 350,000 lb. Range missions up to 6500 n.mi. and radius missions up to 3600 n.mi. have been considered. Takeoff and landing distances between 7000 and 10,000 ft are important constraints on the configuration concepts. The results indicate that a configuration employing conventional technology in all disciplinary areas weighs approximately 2 million pounds to accomplish either a 6500-n.mi. range mission or a 3600-n.mi. radius mission with a 350,000-lb payload. Other than payload and range, some of the more important parameters affecting the vehicle design gross weight are shown to be aerodynamic efficiency (lift/drag ratio) and fuselage sizing.

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LARGE MILITARY CARGO AIRCRAFT PROJECT TEAM

This study was conducted by the Advanced Vehicle Concepts Branch, Ames Research Center, NASA, in response to a USAF-Flight Dynamics Laboratory (AFFDL) request for support. The study was conducted by:

Walter P. Nelms, Jr.	Advanced Vehicle Concepts Branch, Ames Research Center
Ronald Murphy	Design Branch, Air Force Flight Dynamics Laboratory
Alice Barlow	Computer Sciences Corporation

Aircraft synthesis in the Advanced Vehicle Concepts Branch at Ames is accomplished by a team of individuals with expertise in the various disciplines in aircraft conceptual design. Contributions to this study effort in their respective areas of expertise were:

Program Control & Optimization	Garret Vanderplaats
Geometry	Charles R. Castellano Ralph Carmichael
Aerodynamics	John A. Axelson Alan D. Levin
Mission Trajectory	Michael Tauber John Paterson
Propulsion	Jack Morris Rodney O. Bailey
Weights	Walter P. Nelms Alice Barlow
Economics	Charles R. Castellano

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SYMBOLS

$\mathcal{R}$	aspect ratio
b	span, ft
BPR	bypass ratio
C_D	drag coefficient, $\frac{\text{drag}}{qS}$
C_{D_i}	induced drag coefficient, $\frac{\text{induced drag}}{qS}$
$C_{D_{min}}$	minimum drag coefficient, $\frac{\text{minimum drag}}{qS}$
C_{D_W}	wave drag coefficient, $\frac{\text{wave drag}}{qS}$
C_L	lift coefficient, $\frac{\text{lift}}{qS}$
$C_{L_{APP}}$	landing approach lift coefficient, $\frac{\text{approach lift}}{qS}$
$C_{L_{LOF}}$	lift-off lift coefficient, $\frac{\text{lift-off lift}}{qS}$
$C_{L_{TRIM}}$	trimmed lift coefficient, $\frac{\text{trimmed lift}}{qS}$
c_R	root chord, ft
c_t	tip chord, ft
CG	center of gravity location, percent M.A.C.
CPR	compressor pressure ratio
d	diameter, ft
FPR	fan pressure ratio
G	acceleration of gravity, 32.2 ft/sec^2
h	height, ft
h_{BC}	altitude at beginning of cruise, ft
l	length, ft
L/D	lift/drag ratio

$(L/D)_{APP}$	landing approach lift/drag ratio
M	Mach number
$M_{cr_{DES}}$	design cruise Mach number
M_{max}	maximum Mach number
M.A.C.	mean aerodynamic chord, ft
$N_{Z_{LIM}}$	limit load factor
$N_{Z_{ULT}}$	ultimate load factor
q	free-stream dynamic pressure, lb/ft^2
q_{max}	maximum dynamic pressure, lb/ft^2
S	planform area, ft^2
S_W	wetted area, ft^2
sfc	specific fuel consumption, lb/lb-hr
sfc_{cr}	specific fuel consumption at cruise, lb/lb-hr
T	thrust, lb
T_{SLS}	maximum sea-level static thrust, lb
t_{ENDUR}	endurance time, hr
TIT	turbine inlet temperature, °R
TOFL	takeoff field length, ft
T/W	sea-level static thrust-to-weight ratio of aircraft
$(T/W)_{ENG}$	sea-level static thrust-to-weight ratio of aircraft
t/c	thickness-to-chord ratio
$(t/c)_R$	root thickness-to-chord ratio
$(t/c)_t$	tip thickness-to-chord ratio
V_{LOF}	lift-off velocity, knots
W_g	gross takeoff weight, lb
W_i	weight of individual component, lb

W_p	payload, lb
$W_{p_{DES}}$	design payload, lb
W_{wing}	wing weight, lb
$W_{wing/S}$	wing unit weight, lb/ft^2
$W_{wing/Wg}$	wing weight fraction
w	width, ft
W/S	takeoff wing loading, lb/ft^2
α	angle of attack, deg
δ_{FLAP}	flap angle, deg
Λ	sweep, deg
$\Lambda_{c/4}$	sweep of quarter chord, deg
Λ_{LE}	sweep of leading edge, deg
λ	taper ratio
ρ	density, slugs/ft^3
σ	ratio of density at altitude to density at sea level

PRELIMINARY ANALYSIS OF LONG-RANGE AIRCRAFT DESIGNS FOR FUTURE HEAVY AIRLIFT MISSIONS

Walter P. Nelms, Jr., Ronald Murphy,* and Alice Barlow**

Ames Research Center

INTRODUCTION

If the United States is to meet future world-wide commitments, the strategic airlift mission will likely require a capability to provide continuous and sustained air movement of high-value military equipment between the continental United States and overseas areas without reliance on foreign refueling bases. The Air Force has responsibility to insure that the technology is available for the timely development of such an aircraft system. Because of the long development time required for any major system, it is appropriate to initiate a program to develop an adequate data base for these possible future aircraft. A new strategic airlift aircraft will become advantageous when technology advances make possible a new system with significantly increased performance and operational flexibilit: at substantially reduced cost. Therefore, to determine whether or not there has been sufficient progress, an assessment of the total spectrum of advanced technology is required. To this end, the Air Force initiated the Advanced Technology Large Aircraft System (ATLAS) program. The objective of this program is to establish the mission utility of very large, advanced technology, military aircraft and to assess and promote the technology data base relevant to development of these future systems. The program will consider whether an aircraft designed for the strategic airlift mission has the potential to perform alternate missions when necessary modifications are incorporated. Therefore, the rol :: of airborne missile launcher, battle platform, command and control, and aerial tanker will be studied later in the Air Force ATLAS program.

In support of the ATLAS program, the USAF Flight Dynamics Laboratory (AFFDL) initiated a multiphased study to assess technology and evaluate large-aircraft concepts. The initial phase of this program is an in-house design study conducted by the AFFDL with support from NASA Ames Research Center, and includes discussions with Air Force operational commands and planning organizations. The purpose of the present report is to document the results of the design studies by Ames Research Center of conventional "state-of-the-art" large military cargo aircraft. These results will be used as a starting point for continued Air Force studies.

The computerized design studies have been conducted by the Advanced Vehicle Concepts Branch of Ames using the Aircraft Synthesis program (ACSYNT).

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*Air Force Flight Dynamics Laboratory, Wright Patterson Air Force Base, Ohio 45433.

**Computer Sciences Corporation, Mountain View, California 94043

The studies consider "conventional designs" with payloads from 250,000 to 350,000 lb for ranges up to 6500 n.mi. Takeoff and landing distances between 7000 and 10,000 ft are important constraints on the configuration concepts. For specified missions and payloads, vehicles utilizing conventional technology are optimized for minimum gross weight, and parametric sensitivity analyses of selected designs are given. The effects of design parameters such as wing loading, thrust-to-weight ratio, aspect ratio, and fuselage sizing, and mission parameters such as payload, range, and cruise Mach number are presented. The configurations developed in this study will serve as baselines in later work to assess the effects of improvements in technology such as supercritical aerodynamics, advanced control systems, composite materials, and advanced propulsion systems.

The main body of the present report describes the mission, the design guidelines, and the study results. In appendix A, the methods of analysis employed in the study are presented, including a brief description of the ACSYNT program, modifications required to the program for the present study, and a description of the takeoff and landing calculations. As part of the study, the ACSYNT program has been correlated with the C-5A aircraft, and this activity is described in appendix B.

MISSION

The missions used in the present study were specified by the AFFDL. The baseline payload is 350,000 lb, and the aircraft is to be sized such that this payload can be carried on a 6100-n.mi. range mission or a 3600-n.mi. radius mission (returning empty) with the more critical of these two conditions serving as the design point. Details of the basic range and radius missions are presented in figures 1 and 2, respectively. Once baseline designs have been established, parametric variations in mission and configuration parameters will be made.

As part of the study, the effects of takeoff and landing field lengths on the vehicle design are to be investigated. The rules for the baseline design specify that takeoff and landing shall be accomplished within 8000 ft over a 50-ft obstacle at full gross weight. Figure 3 shows a diagram of these requirements. The reasoning behind the 8000-ft distance and details of the takeoff and landing calculations are included in appendix A.

DESIGN GUIDELINES

General

In addition to the mission rules and field length requirements previously described, several other guidelines were specified by the Air Force for the study as follows:

- Conventional wing-body-tail design
- "State-of-the-art" baseline vehicle

-463 L palletized cargo
-Floor bearing strength sufficient to support three M60A main battle tanks at 110,000 lb each
-Pressurized fuselage (including cargo compartment)
-Forward loading of main cargo
-Provision for loading small cargo through aft fuselage (drive-through capability not essential)
-JT9D engine technology
-JP fuel located all in wing
-Limit load factor = 2.50 (ultimate = 3.75)
-No kneeling capability in landing gear
-Minimum cruise altitude greater than 30,000 ft
-Service ceiling equal to or greater than 32,000 ft

Throughout the study, the figure of merit for the baseline designs will be minimum gross weight to accomplish a specified mission. Minimum gross weight is obtained in the automated optimization process by determining the optimum combination of wing loading, thrust-to-weight ratio, wing sweep, and wing aspect ratio, subject to appropriate constraints such as required takeoff distance. The above list of parameters initially included wing thickness-to-chord ratio and taper ratio, but the gross weight was found to be essentially insensitive to small changes in these two parameters. Therefore, to conserve computer time, these are not considered in the optimization parameter list in most of the study.

As a result of correlating the ACSYNT program with the C-5A aircraft, multiplying factors were applied to several of the component weight estimating relationships in the present study to account for special features of large cargo type of aircraft. These multiplying factors are described in the C-5A correlation activities in appendix B. One additional factor has been used, which was a result of early synthesis attempts in the present large-aircraft study. Based on the judgment of the Air Force and NASA design engineers, the wing unit weight, as predicted by the relationship described in appendix A, appeared to increase too rapidly with increasing gross weight. As a result, the estimated wing weight for these large cargo aircraft was judged to be somewhat high. Therefore, a multiplying factor of 0.90 was applied to the wing weight estimations for this study. Further discussion and substantiation of the resulting final wing weights are given in appendix C.

Fuselage Sizing

To achieve the "best" fuselage configuration for various design payloads, three cargo compartment cross sections have been generated by the AFFDL. These cross sections are shown in figure 4. The internal cargo arrangement of configuration A (fig. 4(a)) is the same as the C-5A arrangement. Configuration B (fig. 4(b)) is the cross section used for the baseline vehicle. Configuration C (fig. 4(c)) is the largest cross section considered in the study. Figure 4(d) shows the C-5A fuselage cross section for comparisons. The cargo compartment cross sections were developed for compatibility with existing pallets and containers (military/commercial 463 L pallets at 4500 lb/pallet or air/surface intermodal containers) and for the requirements of outsized cargo. The floor-load bearing must be sufficient to support three M60A main battle tanks at 110,000 lb/tank. The entire cargo compartment is pressurized, and quarters for a relief crew are provided.

The cargo floor length is a function of design payload as shown in figure 5. The floor lengths are 295, 227, and 212 ft for fuselage configurations A, B, and C, respectively, at the baseline design payload of 350,000 lb.

Forward and aft fuselage contours are defined in figures 6(a) and (b), respectively, for all fuselage configurations. The aft fuselage contour, with a small opening for access and loading of small cargo, is considered adequate for this particular configuration. Drive-through loading is not seen as an essential design feature for a future large aircraft of this type, since it is unlikely such aircraft would be exposed to a forward combat area. Thus, necessary ground-handling equipment could be provided, and the time for off-loading would not be expected to be a critical factor in aircraft survivability. Elimination of the aft-loading requirement results in considerable weight and cost reductions.

The combination of lengths in figures 5 and 6(a) and (b) results in the total fuselage lengths shown in figure 7. For the baseline design payload of 350,000 lb, the following fuselage configurations are obtained:

	Conceptual fuselage designs (conventional aircraft)		
	A	B	C
Total pallets	77	77	77
Cargo floor length, ft	296	227	211
Total fuselage length, ft	383	326	316

Fuselage B is used with a length of 326 ft for the initial baseline designs with a payload of 350,000 lb. The effect of fuselage sizing on the aircraft gross weight is considered in the parametric variations.

RESULTS

Unconstrained Field Lengths

Figures 8(a) and (b) present carpet plots of aircraft gross weight as a function of wing loading and design cruise Mach number for the basic range and radius missions, respectively, with a payload of 350,000 lb. In all cases, the configurations are optimized for minimum gross weight but are without constraints for takeoff and landing field length. These configurations are designed by the cruise conditions. The results show a decrease in gross weight with an increase in wing loading. Also note that there is roughly a 10 percent increase in minimum gross weight in going from a design cruise Mach number of 0.75 to 0.80.

Limit lines for various takeoff distances and for an 8000-ft landing field length are superimposed on the plots. It can be seen that for both missions, configurations with wing loadings below about 145 are able to meet the 8000-ft landing requirement. However, none of these configurations can achieve a takeoff of 8000 ft or less, indicating that the takeoff conditions are critical. Takeoff field length will therefore be the primary constraint in the study.

Another important conclusion from these results is that the gross weights for the radius mission are greater than those for the range mission (compare figs. 8(a) and (b)). Therefore the radius mission will be considered as the critical mission in the study.

Constrained Takeoff Field Lengths

The results shown in figure 8(b) for the radius mission are reoptimized with an 8000-ft takeoff field length constraint. A carpet plot of these minimum gross weight configurations for various wing loadings and design cruise Mach numbers is shown in figure 9. These configurations are primarily designed by the takeoff conditions. Reducing the required takeoff field length obviously results in a heavier aircraft (compare figs. 9 and 8(b)) because of the increased thrust-to-weight ratio that is necessary. Note the development of a minimum gross weight for a given cruise Mach number in figure 9. Figures 8(a) and (b) did not display a minimum of this sort. There is still roughly a 10 percent increase in going from a design cruise Mach number of 0.75 to 0.80.

Two baseline design points are identified in figure 9. The first (Baseline I) is the minimum gross weight for the design matrix. This is 1.97 million pounds and occurs at a wing loading of 125 psf for a design cruise Mach number of 0.75. Again it should be emphasized that this configuration employs conventional technology in all disciplinary areas thus contributing to the large gross weight. Figure 10 shows three views of the Baseline I configuration, and table 1 presents a summary of the vehicle characteristics including a weight statement. Additional information in the form of computer graphics output is included in appendix D. For the Baseline I configuration shown, the wing span is about 400 ft and the overall length is about 360 ft. For comparison, the C-5A aircraft has a span of 222 ft, a length of 246 ft, and a gross weight of about 728,000 lb, depending on the mission. In other words, the Baseline I configuration has a gross weight about 2.5 times that of the C-5A and almost twice the span. As figure 10 shows, the configuration is of conventional wing-body-tail design, and the overall features are seen to be similar to the C-5A aircraft, that is, a high wing, tee-tail concept. The configuration has nose-loading capability similar to the C-5A, but has only limited access in the rear. Eight turbofan engines of bypass ratio 5.1, each with 58,561 lb of sea-level-static thrust, are mounted on the wings. Alternate numbers of engines are considered in the parametric variations.

Baseline I is the configuration of primary interest in most of this study. However, a second baseline (baseline II) is identified in figure 9 and assumes a design cruise requirement of Mach 0.80. For this configuration, the minimum gross weight of 2.2 million pounds occurs close to a wing loading of 130 psf as shown. Table 2 gives details of the baseline II configuration.

The minimum gross weight points for all design cruise Mach numbers in figure 9 are shown plotted in figure 11. These represent the minimum gross weights for a payload of 350,000 lb, a radius mission of 3600 n.mi., and an 8000-ft takeoff distance. Baselines I and II are indicated on this plot. Note again that the minimum gross weight occurs at a design cruise Mach number of 0.75.

In considering the optimized configurations in figure 11, it is pertinent to mention some of the checks or verifications that are made of the design results. These checks are desirable to verify that the study configurations have certain characteristics that, according to engineering judgment, are not unlike those of similar existing aircraft. For example, it is expected that the wing weight on a per-square-foot basis for these large study configurations should be similar or somewhat heavier than that of existing large transport aircraft. This, in fact, was found to be the case. A brief discussion of this and examples of other checks on the results are given in appendix C.

Optimum Sensitivities

It is important to determine if any particular parameters, other than obvious ones such as range and payload, have major effects on the configuration gross weight. This is accomplished by performing optimum sensitivity studies. That is, as a parameter is varied from its baseline value, the entire configuration is reoptimized to obtain the minimum gross weight for a configuration with the new value of the parameter. Several of these optimum sensitivity studies are presented in this section.

An important item to be investigated in the present study is the effect of fuselage cross section. Figure 12 shows the effect of three fuselage cross sections previously described (fig. 4) on the gross weight of the Baseline I design. Payloads from 250,000 lb to the baseline value of 350,000 lb are considered for a takeoff distance of 8000 ft. To accommodate the various payloads, the fuselage cross section is held constant while the length varies as indicated in figure 7. Again, it is important to note that all configurations in figure 12 are optimized for minimum gross weight. Fuselage cross section A gives the lowest gross weights by about 3 to 4 percent, but the accompanying fuselage length is much greater than the other two fuselage lengths. This increased fuselage length presents a serious problem for rotation during takeoff and landing. For these reasons, fuselage B is used as the baseline in the study.

Two alternate numbers of engines other than the baseline of eight are considered in figure 13 for both the baselines I and II configurations. The optimized designs employing 6 and 10 engines are shown with the sea-level static thrust per engine indicated parenthetically. For these configurations, there appears to be very little effect of the number of engines on gross weight. Therefore, eight engines will remain as the baseline value throughout the study.

The effect of load factor on baseline I is presented in figure 14. Optimized configurations are shown for limit load factors from 2.25 to the baseline value of 2.5. The corresponding ultimate load factors are also indicated. If it is possible to design to the lowest load factor shown, the gross weight can be reduced by about 5 percent from the baseline I value.

Another item of interest is the sensitivity of the baseline design to reserve fuel requirements. Figure 15 shows the effect of reducing the reserve fuel (percent of initial fuel) below the baseline I value of 5 percent. The configurations have each been reoptimized for the reduced reserve values. If the reserve requirements can be relaxed by about half (to 2.5 percent), a 3 percent reduction in gross weight of baseline I is possible.

It is of interest to determine the effect of combining the reduced load factor and fuel reserve requirements. Figure 16 shows configurations that have been reoptimized for an ultimate load factor of 3.375, a fuel reserve of 2.5 percent, and for various design cruise Mach numbers. Also shown for comparison is the minimum gross weight curve taken from figure 11. Baseline configurations I and II are identified on the plot. As the figure indicates, combining the above two effects results in about an 8 percent reduction in gross weight.

Reducing the structural weight of this wing is a plausible design improvement. Using advanced composite materials instead of aluminum might result in wing weight savings of 20 to 30 percent. The effect of such a reduced wing weight on the gross weight of the present configuration is shown in figure 17. Optimum configurations for a wing weight reduction of up to 20 percent (multiplying factor of 0.8) are shown for both baselines I and II designs. In both cases, a 20 percent saving in wing weight translates into roughly a 9 percent reduction in gross weight.

Nonoptimum Sensitivities

The previous section presented optimum sensitivity information or results obtained by reoptimizing the entire configuration for minimum gross weight when a given parameter is varied. Another type of sensitivity information that is useful to the designer is in the form of nonoptimum sensitivities. In this case, as a given parameter is varied, the configuration is not reoptimized. In other words, for small changes about the nominal or baseline value, the effect of changing only a given parameter can be assessed. This information is generally presented in the form of numerical, nondimensional, sensitivity factors. Figure 18 presents two examples of nonoptimum sensitivities. The effect of thrust-to-weight ratio (T/W) and of wing loading (W/S) on the gross weight (W_g) of baseline I is demonstrated. Using the T/W curve as an example, the normalized values of gross weight (W_g divided by the baseline or nominal W_g) are plotted versus normalized T/W (T/W divided by the nominal T/W). As indicated on the figure, the slope of this curve about the baseline value point gives the sensitivity, or in other words, the percent change in gross weight for a 1-percent change in T/W. For example, a 1-percent increase in T/W increases (plus sign) the gross weight by 0.22 percent (or a 10-percent increase in T/W increases the gross weight by 2.2 percent). In the second example, a 1-percent increase in W/S decreases (negative sign) the gross weight by 0.25 percent. This process can be repeated for any number of aircraft and mission parameters. These nondimensional sensitivities can then be directly compared and ranked, thus providing the designer with a rapid relative indication of the effect of changing various parameters. This permits an assessment of critical areas and thereby helps focus research on the most significant of these areas. These sensitivity values are particularly useful for rapid identification of overall improvements in the vehicle design due to changes in the technology of a specific area.

Table 3 presents sensitivity factors as described above for the baseline I configuration. The sensitivities are divided into three categories: design parameters, efficiency indicators, and mission parameters. Design parameters can be directly controlled by the designer, efficiency indicators are essentially intermediate results of the particular design, and mission parameters are specified to the designer as basic requirements.

Of the design parameters, gross weight is most sensitive to fuselage sizing (table 3). A 10-percent increase in either fuselage diameter or length increases gross weight by 5.4 and 4.5 percent, respectively. This indicates that the fuselage aerodynamics and structural weight are important considerations in the design of this configuration.

Of all the efficiency parameters considered, it is of primary interest that the aerodynamic efficiency (lift/drag ratio) has the greatest effect on gross weight of the Baseline I configuration (table 3). This implies that considerable effort should be devoted to achieving an efficient aerodynamic design. Other aerodynamic parameters, such as minimum drag and induced drag (which, of course, relate to the aerodynamic efficiency), are shown to have a significant effect on gross weight of the baseline concept. Wing and fuselage weights are also of prime concern in the design of this configuration, implying that significant improvements in gross weight may be obtained through the use of advanced technology in the form of composite materials. As may have been expected, the mission parameters, specifically design radius and design payload, have a significant effect on the gross weight of baseline I.

Performance Factors

During the design studies, certain performance factors that are of interest are determined for the resulting configurations. Two of these are briefly described here for the baseline I and II concepts. Figure 19 shows maximum Mach number contours as a function of altitude for both baselines. Superimposed is a line of maximum design dynamic pressure. Thus the maximum Mach numbers are established for the configurations as 0.82 and 0.84 for baselines I and II, respectively. The figure also indicates the altitudes at which these occur.

The maximum endurance times for the baseline designs are of interest since the aircraft could be used on other missions where endurance may be a prime factor. For baselines I and II, maximum endurance times are shown in figure 20 at various Mach numbers for two conditions. First, only the mission fuel is used, with the 5 percent reserve fuel remaining onboard, and this is shown in the lower portion of the figure. Second, if the entire payload could be converted to fuel, the endurance times possible are as shown in the upper portion of the figure (5 percent reserve fuel remaining). Using only the mission fuel (the case of most interest), maximum endurance times of 20 to 21 hours are achievable with either baseline design. The Mach numbers for the greatest endurance are seen to be about 0.43 and 0.46 for baselines I and II, respectively.

Off-Design Results

This section describes the effects of operating the baseline configurations at off-design conditions of speed and range. One figure will be for baseline II, and the remaining figures will concentrate on baseline I, which is of greatest interest in the present study.

The baseline II configuration is designed to cruise at a Mach number of 0.80 as previously described. If this configuration is then fixed, with fuel weight being the only variable, an assessment can be made of operating at off-design conditions. Figure 21 shows the effect of operating fixed baseline II at lower cruise Mach numbers on the 3600-n.mi. radius mission with the basic payload of 350,000 lb. As indicated, there is no particular advantage in terms of fuel usage in cruising this design at lower Mach numbers, as the gross weight remains essentially constant. Also shown in figure 21 is the reduction in gross weight possible (approximately 2.5 percent) through reduced fuel consumption as a result of operating fixed baseline II on the 6100-n.mi. range mission. This again indicates that the basic radius mission specified for the study is the most critical when compared to the basic range mission.

Figure 22 presents the results of operating the baseline I configuration (cruise Mach number of 0.75) at off-design conditions. The basic payload of 350,000 lb and the takeoff distance of 8000 ft or less are maintained. Figure 22(a) shows the effect of a reduced radius mission on the gross weight of the baseline I configuration. The upper curve is for a fixed configuration (designed for a 3600-n.mi. radius) with fuel weight the only variable. The lower curve represents the baseline I configuration resized for each radius. For this latter case, the configuration is scaled in size (geometry and weight) but has a fixed wing loading and thrust-to-weight ratio.

The baseline I configuration was designed for the basic radius mission, which has been shown to be most critical. If this configuration is fixed and then "flown" on a range mission, it is possible to achieve a distance of

6500 n.mi. as shown in figure 22(b). This, of course, is 400 n.mi. further than the baseline range mission requirements of 6100 n.mi. Thus, the baseline I configuration is capable of accomplishing either a 3600-n.mi. radius mission or a 6500-n.mi. range mission for the same gross weight. Figure 22(b) also shows results comparable to those of figure 22(a); that is, operating baseline I at reduced ranges either as a fixed design or as a configuration resized for each range.

During the course of the study, the Air Force expressed interest in the baseline I configuration (M_{cruise} = 0.75) at different design ranges and payloads. Therefore, using the computerized design process, additional results were generated for the range mission with the baseline I configuration as a basepoint. Figure 23 presents the results of this effort.

Figure 23(a) is a carpet plot showing the effect of gross weight of resizing the baseline I configuration for various payloads and ranges with three different takeoff constraints. The baseline I configuration is shown in the figure as having a payload of 350,000 lb, a range of 6500 n.mi., and a takeoff distance of 8000 ft. The baseline I design is then resized (geometry and weight scaled) for the various payloads and ranges at a constant wing loading of 125 psf. Since the lift coefficient for lift off is maintained at 1.47, the thrust-to-weight ratio varies in figure 23(a) for the three different takeoff distances (see appendix A) as follows:

Takeoff distance, ft	Thrust-to-weight ratio
7000	0.27
8000	0.24
9000	0.22

From figure 23(a), the gross weight of configurations with various combinations of payload, range, and takeoff distance can be determined.

Of particular interest to the Air Force is a configuration identified as baseline III in figure 23(a). This design has a gross weight of 1.3 million lb, a payload of 350,000 lb, a range of 3500 n.mi., and a takeoff distance of 8000 ft. Details of the baseline III configuration are given in table 4.

The effect of takeoff field length requirements on the configuration gross weight is more readily apparent in figure 23(b). These results were taken from figure 23(a) for a payload of 350,000 lb. Baselines I and III are identified in the figure at a takeoff distance of 8000 ft.

Range-Payload Tradeoff

Figure 24 shows the range-payload performance of the three baseline designs identified in this study. This figure is for configurations of constant gross weight as noted with an 8000-ft takeoff distance for the range mission. The maximum ranges at a payload of 350,000 lb are shown to be 6500, 6400, and 3500 n.mi. for baselines I, II, and III, respectively. As fuel is substituted for payload while maintaining a constant gross weight, the range of the various configurations increases until the payload is completely replaced by fuel. At this point (zero payload) the maximum ranges possible for the baselines I, II, and III designs are 10,740, 9930, and 8780 n.mi., respectively. The entire payload can be replaced by fuel for all three configurations since there is more than adequate fuel volume available in the wings (see appendix C).

Economics

Using the computer programs described in appendix A, the development and production costs of the three baseline designs have been calculated. As indicated in appendix A, operation cost is not included in this study. Program inputs and assumptions for the cost analysis of the present configurations are as follows:

- Hourly rates are based on approximate 1974 dollars

Engineering	\$20.00
Tooling	17.00
Manufacturing	15.00

- An engine development is assumed (sizing the configuration so that an existing engine could be used would obviously be a cost reduction item)
- Prototype aircraft are not considered and five flight-test aircraft are assumed
- A fee of 10 percent is used in the calculations

Figure 25(a) shows the total development and production cost as a function of the number of aircraft produced for the three baseline concepts. The cumulative average cost per aircraft is indicated parenthetically in the figure. The important thing to note is the relative comparison between the costs of the three concepts. There is approximately a 5 percent increase in cost due to increasing the design cruise Mach number from 0.75 for baseline I to 0.80 for baseline II. Reducing the range requirements from 6500 n.mi. for baseline I to 3500 n.mi. for baseline III results in approximately an 18-percent reduction in the total development and production costs.

A bar chart showing the breakdown of the development and production costs for 100 aircraft is shown in figure 25(b). Results are presented for the baselines I and II concepts. A total development and production cost for 100 aircraft is seen to be about \$11.8 billion for the baseline I design, or \$118 million per aircraft. From figure 25(b), it is apparent that the airframe is the major cost item for both development and production.

CONCLUDING REMARKS

A computer design study of very large cargo aircraft for the future heavy airlift mission has been conducted by the NASA Ames Research Center, using the Aircraft Synthesis program, ACSYNT. The study was requested by the Air Force under an agreement whereby Ames provides computerized design support to the Air Force Flight Dynamics Laboratory. The study concentrated on conventional designs employing no advanced technology and with payloads from 250,000 to 350,000 lb. Range missions up to 6500 n.mi. and radius missions up to 3600 n.mi. have been considered. Takeoff and landing distances between 7000 and 10,000 ft are important constraints on the configuration concepts. As part of the study, the ACSYNT program has been correlated with the C-5A aircraft. Some of the more important results are:

1. A baseline configuration is established with a gross weight of approximately 2 million pounds, which can accomplish either a 6500-n.mi. range mission or a 3600-n.mi. radius mission with a 350,000-lb payload. This configuration has been optimized for minimum gross weight with a takeoff and landing distance within 8000 ft. The optimum cruise Mach number is 0.75, and the wing loading is 125 psf at takeoff. It should be emphasized that this configuration employs conventional technology in all disciplinary areas thus contributing to the large gross weight. This configuration will serve as a baseline in later work to assess the effects of improvements in technology, such as supercritical aerodynamics, advanced control systems, composite materials, and advanced propulsion systems.

2. To increase the cruise Mach number from an optimum of 0.75 to 0.80 would require approximately a 10-percent increase in gross weight.

3. The above baseline design has a maximum endurance of approximately 20 hr with a 350,000-lb payload. If all of this payload was converted to additional fuel (which could easily be accommodated in the wing along with the regular fuel), either a maximum range of 10,700 n.mi. or a maximum endurance of 35 hr is possible.

4. Other than range and payload parameters having a major effect on the vehicle design gross weight are aerodynamic efficiency (lift/drag ratio), fuselage sizing, minimum drag, and wing and fuselage weight; of these, aerodynamic efficiency has the greatest effect.

5. The results show that the total development and production cost would be approximately $12 billion for a fleet of 100 of these aircraft.

6. On the basis of the C-5A aircraft correlation, it is indicated that the ACSYNT program should give good results for heavy aircraft of the C-5A type.

APPENDIX A

METHOD OF ANALYSIS

This appendix briefly describes the Aircraft Synthesis program used in the present study and the modifications made to the weight-estimating routine as a consequence of the special requirements of the study. Also, the methods used to calculate the takeoff and landing distances are described, and the philosophy behind the 8000-ft field length requirement is discussed.

Synthesis Program

A computerized Aircraft Synthesis program (ACSYNT) developed by the NASA Ames Research Center to provide rapid conceptual design information was used for this study (ref. 1). This modularized program consists of a control module and technology modules for geometric, mass, aerodynamic, propulsive, and economic information. In addition, there are modules to provide automatic design convergence, sensitivity, and optimization calculations, as well as graphical output. Figure 26 presents a block diagram of the ACSYNT program.

Inputs to the various modules include control parameters, initial vehicle definition parameters, mission profile, and several initial assumptions to start the program. Output includes vehicle characteristics required to accomplish the specified mission, such as component weights and geometry, fuel and time requirements for the various phases of the mission profile, aerodynamic and propulsion characteristics, and vehicle cost information.

Control program- The control program sequences the order in which the modules are executed and transfers information to all the other modules. This module controls the limits of the various program loops, number of passes to be made through the program, and criteria for convergence of the vehicle. Convergence of the vehicle is determined by a regula-falsi procedure (ref. 2). If the vehicle is either too light or too heavy compared to the input estimate of the vehicle gross weight, the entire synthesis program is recycled until the predicted and calculated gross weights agree within a specified tolerance.

Geometry module- Based on input configuration parameters, some of which are fixed and others requiring only an initial estimate, this module defines the vehicle size and shape to be used in the remaining parts of the program. Initial size estimates of the fuselage, engine, wing and tail surfaces are made. The characteristics of these components are updated at each pass through the program based upon information supplied by the other technology modules or by the control program. The fuselage is sized to contain the cargo compartment, crew and relief crew, and electronic equipment. The wing is sized on the basis of input wing loading and shape parameters, including the constraint of having sufficient volume for the entire fuel supply. Balance is calculated on the basis of a specified static margin and tail volume coefficient, or the static margin is determined for a fixed ratio of tail area to wing area. This module calculates the final geometric vehicle properties that will satisfy the mission.

Aerodynamics module– The aerodynamic characteristics for a given altitude and Mach number are determined from the geometric characteristics. The trajectory module specifies lift, drag, or angle of attack at a Mach number and altitude, and the aerodynamics module determines the remaining variables. Calculation procedures employ both theoretical methods and empirical information. Results have been calibrated with existing aircraft and with wind tunnel data for configurations at both high and low angles of attack.

Friction drag estimates are based on the method of Bertram (ref. 3), with an empirical correction for thickness-induced pressure fields made according to the method of Koelle (ref. 4). Base drag is computed using base pressure coefficient as a function of Mach number. Lift and drag-due-to-lift are calculated for angles of attack from zero to beyond maximum lift using a nonlinear theory currently under development at Ames. This method (ref. 5) is derived from a combination of potential theory and momentum integrations.

Propulsion module– The propulsion module is a one-dimensional cycle analysis program developed at Ames (ref. 6) that calculates engine performance and other propulsion characteristics at any specified altitude, Mach number, and power setting. Required inputs include engine type (turbojet, turbofan, etc.) and component efficiencies, pressure ratios, and temperature limits. The engine weight and length are calculated using results obtained from the MARS system (ref. 7), and the inputs required include thrust, turbine inlet temperature, bypass ratio, compressor pressure ratio, number of compressor and turbine stages, and year of first installation. Variations in power settings available are: maximum afterburning, 100 percent rpm, maximum continuous, and percentages of maximum continuous. The throttle setting and specific fuel consumption are calculated from information supplied by the trajectory and aerodynamic modules. The basic engine thrust and fuel consumption are corrected for the appropriate installation losses associated with the inlet and nozzle. The program has the capability of correcting for additive drag, subsonic spillage drag, bleed and bypass drags, auxiliary air systems drag, inlet diverter drag, and nozzle boattail and interference drags. For the present study, the standard AIA ram recovery schedule less 3 percent is employed. The basic engine characteristics in this study are essentially state-of-the-art, and no performance improvements have been used that might be considered advanced propulsion system technology.

Trajectory module– This module computes a vehicle trajectory for a specified mission from information generated in the aerodynamic and propulsion modules. The trajectory consists of takeoff, climb, acceleration, cruise, descent, loiter, and landing segments. Equations of motion neglect flight-path-angle rate terms, and integration is by approximate step-by-step procedures. Climb options include an approximate minimum fuel trajectory or a constant indicated air speed climb. Either a Breguet (maximum range) cruise or a segmented, constant altitude cruise is available. This module calculates the fuel used for each phase of the specified mission including the reserve fuel phase, thus establishing the total fuel requirement. A more in-depth discussion of the takeoff and landing phases will be presented later in this appendix since these have important implications on the design of the present vehicle.

Mass properties module– Weights are calculated in this module from empirical equations based on correlations of existing aircraft data. The wing weight is a function of load factor, aspect ratio, leading-edge sweep, taper ratio, thickness-to-chord ratio, design dynamic pressure, and vehicle gross weight. Load factor, surface area, maximum Mach number, and vehicle gross weight are the parameters used in determination of the fuselage weight. Weights of the remaining items, such as tail surfaces, fixed equipment, and engine installation, are determined by similar empirical methods. This module has several options for calculating wing and fuselage weights, and those employed in the present study are discussed under program modifications in this appendix.

Economics module– The development and fleet acquisition costs were determined using a modified version of the cost-estimating relationships developed by the Rand Corporation (ref. 8). The Development And Procurement Costs for Aircraft (DAPCA) computer program used was supplied by USAF-Aeronautical Systems Division (ASD/ACCX). The estimating equations were derived by statistical multiple regression techniques. Several of the cost-estimating relationships (CER) in the DAPCA program were modified (by ASD) from ones based on the total aircraft spectrum to equations based only on heavy cargo, tanker, and subsonic bomber aircraft. The results from the modified DAPCA program correlated well with the C-5A and 747 aircraft (ref. 9). It should be noted that the DAPCA program does not compute operational costs.

Optimization– This module is coupled to the synthesis program to provide an automatic closed-loop optimization of the vehicle. The optimization algorithm is based on Zoutendijk's method of feasible directions (ref. 10). The optimization procedure and computer program are described in references 11 and 12. The best combination of user-specified design variables to minimize vehicle weight (or to minimize or maximize any other parameter) is determined subject to prescribed bounds on the vehicle or mission parameters.

Sensitivity– This module determines the sensitivity of gross weight or aircraft performance to changes in design or mission parameters. Using gross weight as an example, the term "sensitivity" as used here is defined as the change in gross weight/gross weight divided by the change in parameter/nominal value of parameter. In other words, the sensitivity is the relative change in gross weight divided by the relative change in the design parameter. Two types of sensitivity analysis are available from the ACSYNT program. The first, referred to as "nonoptimum sensitivity," considers the effect on vehicle weight or performance of changing only a single design variable. With this type of sensitivity, there is a possibility of violating constraints on the design if the parameter is varied too far from its nominal value. For example, increasing wing thickness-to-chord ratio alone may reduce the vehicle performance below some specified level. An alternate capability of ACSYNT is the use of "optimum sensitivity" analysis. In this case, when a single design parameter is changed, all the selected vehicle design parameters are reoptimized with respect to each other to maintain minimum gross weight (assuming the objective is minimum weight) while satisfying all the design constraints.

Program Modifications

For the present study, modifications were made to the ACSYNT weights program in two primary areas, wing and fuselage structural weight estimating methods. The intent was to provide procedures more appropriate for use on large cargo type of aircraft. After consultation between Air Force and NASA design, weights, and structural engineers, the methods indicated here were incorporated into the ACSYNT program. These equations were provided by the Air Force but are still considered proprietary. Therefore, the actual equations cannot be given, but a brief description and sample correlation will be presented.

The wing weight is a function of the following parameters, which are raised to various powers and multiplied by various constants: design gross weight, ultimate load factor, aspect ratio, quarter chord sweep, span, reference area, taper ratio, root chord, tip chord, maximum thickness at root and tip, and maximum equivalent airspeed. Figure 27 presents the results of the present equation correlated with seven cargo transport aircraft. The calculated wing weights are compared to the actual wing weights for these aircraft, and the agreement is excellent. The figure also indicates the wing weight calculated for a typical design of a large cargo aircraft in the present study.

An equation to calculate the fuselage weight of cargo-type aircraft with heavy flooring and with various loading-door arrangements had to be incorporated into ACSYNT. The parameters in the equation provided by the Air Force are: design gross weight, ultimate load factor, fuselage wetted area, and maximum Mach number. These parameters are raised to various powers and multiplied by various constants. In addition, there are factors included in the equation to account for the following features: front door loading, rear door loading, main cargo floor system, upper floor, and miscellaneous doors, panels, and fairings. The fuselage weights calculated by the present equation compared to the actual fuselage weights of the C-141 and C-5A aircraft are shown in figure 28. Again, the calculated fuselage weight of a typical design in the present study is indicated in the figure.

Takeoff and Landing Considerations

General.– The heavy airlift mission suggests that these airplanes should have basing characteristics similar to the C-5A. Specifically, this implies a 5000-ft landing distance at the design payload and compatibility with runway surfaces that handle aircraft in the 747 and C-5A class. Realizing these requirements to be potentially quite stringent, the baseline design field lengths have been relaxed by the Air Force to 8000 ft (all engine operation over a 50-ft obstacle). For background, the following table provided by the Air Force shows the number of airfields having hard surface runways equal to or greater than the 8000-ft baseline length. The advantage (in terms of number of airfields that can be used) of providing takeoff and landing distances as short as practical is obvious from the table. The effects on gross weight of takeoff and landing field length requirements are investigated in the study.

Field length, ft	Number of Airfields			
	CONUS civil	CONUS military	CONUS total	61 Other countries
7,000	215	184	399	561
8,000	126	170	296	262
9,000	75	115	190	123
10,000	36	90	126	48

For the present study, the takeoff and landing calculations are accomplished at full gross weight using the rules specified in figure 3. The following sections give a brief description of the calculation methods employed.

Takeoff— The takeoff considers all engines operating over a 50-ft obstacle, and the distance is calculated using a procedure similar to that of reference 13. The total distance consists of a ground roll and an air distance to the 50-ft obstacle,

$$S_{total} = S_{GR} + S_{AIR}$$

The ground distance is

$$S_{GR} = \frac{(W_g)\ 2.849(V_{LOF})^2}{2G[(T - \mu W) - (C_D - \mu C_L)qS]_{0.707V_{LOF}}}$$

where the following are assumed,

$$\mu = 0.025$$

$$(C_D - \mu C_L) = 0.09$$

and the values in parenthesis in the denominator are taken at 0.707 of V_{LOF}. The air distance is:

$$S_{AIR} = \frac{50}{(T/W)_{V_{LOF}} - 1/(L/D)_{V_{LOF}}}$$

where T/W and L/D are at lift-off velocity. Lift-off velocity in knots is:

$$V_{LOF} = \sqrt{\frac{295\ (W/S)}{C_{L_{LOF}}}}$$

The takeoff and landing lift characteristics used in the present study are shown in figure 29 for a flap system similar to that of the C-5A. The lift-off lift coefficient ($C_{L_{LOF}}$) is governed by the capability of the aircraft to rotate. Many large designs are not able to rotate or can rotate only through a small angle, and therefore several $C_{L_{LOF}}$ values are considered in this study. For the baseline design, the configuration is capable of a 7° rotation angle. With about 2.5° of wing incidence, the maximum achievable C_L during ground roll would be approximately 1.71 for a 25° flap setting (fig. 29). Since this is a geometry-limited condition, only an 8-percent margin is placed on the lift-off velocity (similar to commercial FAR rules). The baseline $C_{L_{LOF}}$ is then

$$C_{L_{LOF}} = \frac{1.71}{(1.08)^2} = 1.47$$

Two other $C_{L_{LOF}}$ values have been assumed, 0.95 for no rotation, and 2.06 for full rotation.

Using the above three lift coefficients, as well as several W/S values from 120 to 160 psf, and several T/W values from 0.2 to 0.3, a series of takeoff distances were calculated using the method described above. For these calculations, the thrust lapse rate for the study engine shown in figure 30 was used. These takeoff results were then generalized in the form of reference 14 and presented in figure 31 (symbols) for sea-level altitude. Since the scatter of the detailed calculations was small, the generalized curve shown was used throughout the study. Figure 32(a) shows a carpet plot of the above results, and figure 32(b) presents the required T/W for the baseline $C_{L_{LOF}}$ and 8000-ft takeoff distance.

Landing– The landing field length was calculated in two segments, air distance over a 50-ft obstacle and ground roll, using a method similar to that of references 14 and 15. The total landing distance is:

$$X_{TOTAL} = X_{AIR} + X_{GR}$$

The air distance is:

$$X_{AIR} = \left[\frac{(V_{APP}{}^2 - V_{TD}{}^2)}{2G} + 50\right]\left(\frac{L}{D}\right)_{APP}$$

where

$$\text{approach velocity, } V_{APP} = 1.3\ V_{STALL}$$

$$\text{touchdown velocity, } V_{TD} = 1.15\ V_{STALL}$$

and

$$V_{STALL} = \sqrt{\frac{(2)(W_{LAND})}{(S)(C_{L_{APP}})(\rho)}}$$

where landing weight, W_{LAND}, has been assumed as takeoff gross weight for this study.

The ground roll is:

$$X_{GR} = \frac{(V_{TD})^2}{-2(a)}$$

where a is the average deceleration during ground roll, fps^2.

Using the above procedure, two air distances were calculated with approach lift coefficients ($C_{L_{APP}}$) suggested by the Air Force (see fig. 29 for landing lift characteristics). The first air distance calculation assumed a flare capability prior to touchdown with a $C_{L_{APP}} = 1.66$. The second assumed no flare capability and a $C_{L_{APP}} = 1.25$. Both methods yield a touchdown rate of sink of 6 to 7 fps and a glide scope from 1° to 2°.

Two ground roll distances were also calculated, one with all engine thrust reversing and one with no thrust reversing. Braking deceleration for both cases was 6 fps^2 with an additional 1 fps^2 for aerodynamic and ram drag. For full thrust reversing, an additional 2 fps^2 was included. The sequencing of landing decelerating devices to produce these numbers is as follows:

Item	Time (sec) from touchdown
Move throttles	-3
Spoiler application	1.3
Brake application	1.6
Reverser doors start to open	1.8
Spoilers deployed	2.4
Reversers fully deployed	2.7
Full brake pressure	3.3
Maximum reverse thrust achieved	10.3

The four conditions for calculation of landing field length described above, that is, two $C_{L_{APP}}$ values (with and without rotation), each with and without thrust reversing, are presented in figure 33. Landing field length is shown for various wing loadings, and the design field length of 8000 ft is indicated. The landing conditions assumed for the study baseline vehicle are no rotation and full thrust reversing (average deceleration of 9 fps^2).

APPENDIX B

CORRELATION OF ACSYNT WITH THE C-5A AIRCRAFT

General

This appendix summarizes the results of the C-5A correlation activities. The correlation was performed in order to assess the accuracy of the ACSYNT program for prediction of weights and geometries of very large military cargo aircraft. A presentation of all the details of the correlation is beyond the scope of the present report, but a brief summary of the more important results will be given here.

The correlation is accomplished in two parts using the actual characteristics of the C-5A aircraft as the basis for comparison. The first part examines the accuracy of the individual disciplines (aerodynamics, propulsion, etc.) independent of each other. To do this, the various modules are run in a stand-alone manner, and the results are compared to the appropriate characteristics of the actual C-5A aircraft. The second part of the correlation examines the accuracy of the overall design, which consists of an integration of all the disciplines. In this case, the ACSYNT program is run in its entirety using the C-5A mission, and the resulting geometry and weights (fuel, structure, etc.) are compared to the actual C-5A values.

Figure 34 shows the C-5A general arrangement. This aircraft can perform a number of missions of varying payload and range. Figures 35 and 36 present the C-5A basic range and radius mission, respectively, which have been used in this correlation activity. With a payload of 265,000 lb and a gross weight of 706,913 lb, the C-5A can accomplish either a 1700-n.mi. range mission or a 1000-n.mi. radius mission. These mission rules (figs. 35 and 36) will be used in the overall integrated design part of the correlation.

Individual Disciplines

In this section, selected results are presented from the individual discipline correlations using the geometry, aerodynamics, propulsion, and weights modules.

Geometry module— The purpose of the correlation in this discipline is to determine how accurately the geometry module simulates the wetted areas of the actual C-5A. Using the geometry program in a stand-alone fashion, overall dimensional data of the actual C-5A is input to the module. These inputs include span, sweep, taper ratio and thickness-to-chord ratio of the lifting and stabilizing surfaces, fuselage length and maximum diameter, and bare engine length and maximum diameter. The geometry module then generates a configuration based on these inputs and calculates the appropriate wetted areas. Figure 37 shows a computer graphic display of the ACSYNT simulation of the C-5A. A summary of the wetted areas computed by the geometry module compared to the actual C-5A values is presented below:

Component	ACSYNT-simulated C-5A, ft^2	Actual C-5A, ft^2
Fuselage	15,350	16,646
Wing	12,486	10,358
Nacelles	2,186	939
Horizontal tail	1,994	1,844
Vertical tail	1,242	1,759
TOTAL	33,258	31,545

Error in total wetted area is approximately 6 percent. From this table it can be seen that there are discrepancies for certain components. Most notable on a percentage basis is the nacelle wetted area. The ACSYNT program as presently structured does not do a detailed layout of the nacelles, and the maximum diameter resulting from high bypass turbofans is used for the entire pod length (see fig. 37). However, as the table shows, the total aircraft wetted area calculated by the ACSYNT program is within approximately 6 percent of the actual value, and the estimate is on the conservative side.

Aerodynamic module— To correlate the aerodynamic module on a stand-alone basis, C-5A geometric data similar to that for the geometry module are used as inputs. The resulting lift, minimum drag, and drag due-to-lift are then compared to the actual C-5A values.

Figure 38(a) is a plot of minimum drag coefficient calculated by the program compared to the actual C-5A value at 30,000-ft altitude. An error analysis for minimum drag is shown in figure 38(b). In the Mach number region of interest for the C-5A (below about 0.85), the calculated minimum drag is within about 5 percent of the actual value.

Trimmed and untrimmed lift characteristics predicted by the aerodynamic module are presented in figure 39 for several Mach numbers. A limited number of C-5A experimental data points are shown, indicating good agreement with the ACSYNT predicted results. The results of correlating the induced drag are presented in figure 40 for several Mach numbers. Trimmed and untrimmed induced drag coefficients are shown. For lift coefficients of interest to the C-5A (cruise $C_L \approx 0.3$ to 0.4) and for Mach numbers below the drag rise, the agreement between the ACSYNT calculations and the experimental data is acceptable.

Propulsion module— This section gives a comparison of the engine characteristics as predicted by the ACSYNT propulsion module to those of actual engines. As previously indicated, the propulsion module of the ACSYNT program is a one-dimensional cycle analysis routine that calculates thrust and specific fuel consumption (sfc). The engine geometry and weight are calculated using empirical relationships.

For the present correlation, the maximum predicted sea-level static thrust point is matched to the actual engine data for that point. The program then calculates the thrust and specific fuel consumption at all other altitudes, Mach numbers, and power settings. Figure 41 presents correlated results for the C-5A engine (TF39-1). Predictions of the thrust and installed sfc are compared to the actual values for several Mach number and altitude combinations. A similar comparison is presented in figure 42 for the JT9D-25 engine which is used as the basic propulsion cycle for the present large cargo aircraft study. For both engines, the results correlate well with a few exceptions at the deep-throttled conditions. In regions where the aircraft would most likely operate, the ACSYNT-predicted sfc values for a given thrust are within about 6 percent of the actual engine values.

The bare engine geometry and weight are estimated in ACSYNT using empirical techniques based on a number of existing engines. The results predicted for the two engines used in this correlation are summarized below and compared to the actual values. For the TF39-1, the propulsion module does

Parameter	Actual engine	ACSYNT estimate	Percent error
TF39-1			
Diameter	8.3 ft	7.7 ft	7.2
Length	16.9 ft	13.0 ft	23.1
Weight	7222 lb	8219 lb	-13.8
JT9D-25			
Diameter	8.0 ft	7.2 ft	10.0
Length	12.9 ft	14.1 ft	-9.3
Weight	8550 lb	8836 lb	-3.3

not properly model the engine length and weight, underpredicting the length by over 20 percent and overpredicting the weight by 14 percent. The results are much better for the JT9D-25 engine, which is the cycle used in the present study. Improved methods for estimating geometry and weight for a wide variety of engines are presently being developed and incorporated into the ACSYNT program.

Mass properties module— This section compares the aircraft component weights predicted by the ACSYNT weights module to the actual aircraft values. To run the weights routine in a stand-alone fashion, aircraft geometric data similar to that previously described is input to the program. Also, payload, bare engine weight, and fuel weight are required inputs when using the weights module in a stand-alone mode.

Two types of results are presented here. The first consists of a weight statement comparing predicted and actual component weights for the C-5A aircraft. These results are shown in table 5. Again, it should be emphasized that the payload, bare engine weight, and fuel weight are those of the actual C-5A. Also, it should be noted that the results here are for a different C-5A mission than that which is used for the overall design correlation. The first error column (table 5) compares the components themselves, and the second error error column shows the error as a percentage of the actual gross weight of the C-5A. From the table it is obvious that in certain areas, particularly fixed equipment, the error between some components themselves is large, but is not as significant on the basis of gross weight. In any event, fixed equipment is an area where improvements are being made to the ACSYNT program. For the major components of structures and propulsion, the computed and actual results are seen to be within 10 percent (table 5).

A second type of weights correlation consists of comparing the ACSYNT-computed versus actual weights of the individual aircraft components (wing, fuselage, empennage, fixed equipment, etc.) for seven large transport aircraft. The aircraft used in this correlation were the 990, DC-8, DC-10, L-1011, 747, C-141, and C-5A. Computed versus actual weights were prepared for each aircraft component, and both a mean of the errors and a standard deviation were calculated, but these results will not be shown here because they are proprietary. However, with the exception of several minor components, the mean of the errors was within 10 percent of the actual component weight, considering all aircraft. Also, it is important to note that in all cases the C-5A component-weight estimates were within 10 percent of the actual with exception of the fixed equipment which was underpredicted by about 15 percent.

As a result of these correlations, weighting factors have been established to modify a number of the component-weight relationships used in the ACSYNT program to account for special features of large cargo aircraft. These factors were arrived at following consultation between NASA and Air Force design engineers. The factors will be used in the following overall integrated design portion of the C-5A correlation and will also be used in the present large cargo aircraft synthesis studies. The multiplying factors are shown in the following table. The normal landing gear weight relationship in ACSYNT

Component	Weighting factors for large cargo aircraft
Landing gear	1.20
Propulsion	0.89
Electrical	0.80
Avionics	1.80
De-ice/air conditioning	10.00
Flight controls	1.50

is for conventional transport aircraft, and as such, it underpredicts the weight for large cargo aircraft as seen in table 5 for the C-5A. Thus, the above multiplying factor is applied to account for these special types of landing gear. Since many of the items in the fixed-equipment area are mission dependent, this is usually a difficult area in which to provide general weight equations that are accurate. Fortunately, the fixed-equipment weights usually account for only a small part of the gross weight. In any event, to improve the present estimates, the above multiplying factors were established on the basis of the C-5A stand-alone weights correlation (table 5) and will be used in the remainder of the study. It was also judged that the ACSYNT engine weight estimating method gives weights that are high when compared to similar propulsion systems. Therefore, the above multiplying factor was applied to the total propulsion system weight in the remainder of the study.

Overall Design

This section examines the accuracy with which the ACSYNT program can predict the results of the overall integrated design of the C-5A. To do this, the entire ACSYNT program was employed, combining all the disciplines. The C-5A mission rules previously described (figs. 35 and 36) with a payload of 265,000 lb were used. Two examples of the results from the overall design correlation are presented here.

In the first case, the C-5A overall external dimensions and the propulsion system size and weight (fixed to represent the TF39 engine) were simulated as closely as possible with the ACSYNT program. Then the gross weights to accomplish the basic range and radius missions were determined. This essentially involves a correlation of the fuel, structure, and fixed equipment required for the mission, since the payload and propulsion system weights are fixed. The results of this correlation are shown in figure 43. The weights calculated by the ACSYNT program are essentially identical for the range and radius missions as shown. This is as it should be since the C-5A can accomplish either of these missions at the same gross weight of 706,913 lb. As figure 43 shows, the gross weights computed by the ACSYNT program for the simulated C-5A are approximately 0.3 percent over that of the actual C-5A. There are compensating errors in fuel and structure weight as noted. Table 6 gives a breakdown of the computed weights for the simulated C-5A for both basic missions, along with actual C-5A values. Since the computed weights for both missions are essentially the same, only the range mission will be considered in the next comparison.

The second example of the C-5A overall design correlation is referred to here as an optimized aircraft. In this case, the ACSYNT program determines the minimum gross weight configuration to accomplish the basic range mission (fig. 35) with a payload of 265,000 lb while satisfying the takeoff and landing field length constraints of 5000 ft or less. To do this, the program employs an optimization procedure (previously described) to find the optimum combination of W/S, T/W and wing $A\!R$, Λ, λ, and t/c for minimum gross weight. It should be noted that in this case the program is free to resize the propulsion system, whereas before, the engine was fixed to represent the TF39. Also, the fuselage size is constrained to assure that it can accommodate the required payload volume. The results of this overall design correlation are presented in figure 44 for the basic range mission, and a breakdown of the computed weights is given in table 6. The computed gross weight is seen to be essentially that of the actual C-5A, implying no improvement through optimization. This is not totally true, however, since there is approximately a 1.2-percent reduction in fuel compared to the actual C-5A and about a 4-percent reduction compared to the ACSYNT-simulated C-5A (fig. 43). The offsetting feature is the weight of the resized propulsion system which is approximately 6 percent greater than the actual C-5A (table 6). (This item is not shown in the bar chart of fig. 44, however.) The primary reason for this greater weight is that the ACSYNT aircraft optimized at a slightly higher value of T/W than the actual C-5A (0.242 compared to 0.230). This resulted in somewhat larger and therefore heavier engines, thereby accounting for the difference in propulsion system weight. Table 7 is a summary of some of the wing and tail geometric features of the ACSYNT-optimized aircraft compared to those of the actual C-5A, indicating very small differences between the two sets of values.

Summary

It is difficult to establish an absolute number for the expected accuracy of the overall ACSYNT predictions because of the many different disciplines, possible missions, and types of configurations that may be considered. Obviously, a better assessment of this credibility could be obtained by performing correlations on a large number of different aircraft; this is a very time-consuming task, however. To date, the only aircraft that has been correlated which would be representative of very large cargo/transport-type aircraft is the C-5A. A brief summary of this activity has been presented. On the basis of this limited correlation and considering the results in their entirety, it is suggested that the ACSYNT program should be capable of giving gross weight and geometric results that are within 10 percent of actual values for aircraft similar to the C-5A. This is considered sufficiently accurate for computerized design at the early stages of vehicle definition.

APPENDIX C

DESIGN ASSESSMENT

During the course of computerized aircraft design studies, there are occasions when checks are made to substantiate or assess certain design results. Most often these checks occur internally in the computer program itself, but at times they are made by the design engineer following the synthesis of a vehicle. This section gives several examples of these types of verifications.

Wing Weight

Where it is necessary to extrapolate beyond the characteristics of existing aircraft, caution should be exercised to assure that the results are reasonable. An example of this occurs in the present study where the gross weights of the study configurations are two to three times that of any existing aircraft. Here it is important to substantiate the structural weights that are given by the computerized design process. Wing weight is used as the example in this case.

Figure 45(a) shows wing unit weight as a function of aircraft gross weight for configurations with wing-mounted engines, wing aspect ratios from about 6 to 9, and for aluminum structure. As an approximation, the designer can extrapolate to heavier configurations by fairing a wide curve through the existing aircraft points (symbols) as shown. The wing unit weights for the three baseline designs of the present study fall within the bounds of this expected trend. The values for the baselines are between 16 to 18 psf. Figure 45(b) again shows wing unit weight, but this time as a function of wing loading. As before, the three baseline values fall within the bounds of the extrapolated curve. Finally, figure 45(c) shows wing weight fraction as a function of gross weight. The baseline values follow the expected trend, and the baseline wing weights vary from about 13 to slightly over 14 percent of the aircraft gross weight. In summary then, these results indicate that the study wing weights are representative of what might reasonably be expected for these large cargo aircraft using existing structural technology.

Engine Thrust-to-Weight Ratio

As indicated in another section of this report, the engine thrust and weight characteristics are generated by a propulsion cycle program and an engine empirical weight estimation method, respectively. To assure that the engines resulting from the design studies are reasonable, a check of the engine thrust-to-weight ratios are made. For the baseline configurations, the study engines had thrust-to-weight ratios ranging from 5.71 to 5.85. These compare to a range of values from 5.67 to 6.04 for existing engines of the TF39, JT9D, and CF6 class. The study engines are therefore within the present day technology level as far as thrust-to-weight characteristics are concerned. Thus, it is quite reasonable to expect that engines of this or more advanced technology can be provided for these large cargo aircraft.

Wing Fuel Volume

One of the design guidelines for this study is that all of the JP fuel be located in the wings of the configurations. This, of course, is almost a necessity for cargo aircraft of this type. To assure that this is the case for the study configurations, the fuel volume available in the wings is checked against the fuel volume required.

A method similar to that described in reference 15 is used to calculate the volume available in the wing for fuel storage. The results from this method have been compared with more detailed procedures from the Air Force, and they correlate quite well. Inputs for the correlation are wing area, thickness-to-chord ratio, aspect ratio, taper ratio, gross weight, skin thickness, spar location, and percent of span used. In this calculation procedure, fuel can either be located in the wing carry-through structure or not.

The wing fuel volume was calculated for the three baseline designs. Fuel was included in the wing carry-through structure, and between the front and rear spars which were assumed to be at 15 and 70 percent of the chord, respectively, and inside 80 percent of the span. The calculated results are tabulated below, and the total fuel required (including reserves) for the baseline mission (payload = 350,000 lb) for each configuration is also shown.

	Baseline		
	I	II	III
Total fuel volume available, lb	1,640,000	2,140,000	890,000
Fuel required for baseline missions, lb (with payload = 350,000 lb)	759,000	947,000	336,000

Thus, there is more than adequate fuel volume available for the three designs. It appears from these results that wing volume available for fuel is not a problem for these very large aircraft. In fact, for the above three configurations, the entire payload could be converted to fuel and stored in the wings along with the existing fuel (gross weight held constant), and there would still be excess fuel volume available.

APPENDIX D

COMPUTER GRAPHICS DISPLAY

Because of the large amount of output available from the ACSYNT program, a computer graphics system is the most convenient means of displaying the results. Not only can the results of a vehicle synthesis be displayed, but input data can be checked by the design engineer prior to attempting a run of ACSYNT. References 1 and 16 discuss computer graphic systems and the merits of their use in the aircraft design process.

Selected output from the computer graphics system for the baseline I configuration is shown in figure 46. Included are views of the configuration as "seen" by ACSYNT, aerodynamic characteristics, propulsion system performance, and component weight information. These represent only a small portion of the results from ACSYNT that can be displayed.

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TABLE 1.- BASELINE I CHARACTERISTICS

General

W_g = 1,972,540 lb

W/S = 125 lb/ft^2

T/W = 0.238

$N_{Z_{LIM}}$ = 2.50

$N_{Z_{ULT}}$ = 3.75

Engine

No. = 8

Type = Turbofan

l = 15.6 ft

d = 8.3 ft

T_{SLS} = 58,561 lb

sfc_{cr} = 0.60 lb/lb-hr

$(T/W)_{ENG}$ = 5.85

BPR = 5.10

FPR = 1.6

TIT = 2835 °R

CPR = 22.9

Geometry

	Wing	Horiz. tail	Vert. tail
S, ft^2	15,779	2525	2525
S_W, ft^2	31,785	5077	3549
b, ft	397	109	56
Λ_{LE}, deg	27.3	28.9	36.6
$\Lambda_{c/4}$, deg	25.0	24.5	34.9
AR	10.0	4.74	1.24
λ	0.34	0.37	0.80
M.A.C., ft	39.0	22.4	41.1
c_R, ft	53.8	30.6	45.5
c_t, ft	18.3	11.3	36.4
$(t/c)_R$	0.13	0.105	0.13
$(t/c)_t$	0.11	0.105	0.13

TABLE 1.- Continued

Fuselage

l = 326 ft

d = 31.4 ft

S_W = 27,956 ft^2

Cargo bay

l = 227 ft

w = 29.7 ft

h = 14.2 ft

Performance

$W_{P_{DES}}$ = 350,000 lb

$M_{cr_{DES}}$ = 0.75

TOFL = 8000 ft

Max. radius = 3600 n.mi.

Max. range = 6500 n.mi.

M_{max} = 0.82

t_{ENDUR} = 20.7 hr

h_{BC} = 35,600 ft

Mission summary

	Radius			Range		
	Fuel, lb	Time, h.	Distance, n.mi.	Fuel, lb	Time, hr	Distance, n.mi.
Warm up, taxi, takeoff	29,446	--	--	14,724	--	--
Climb and accelerate	49,215	0.66	267	48,990	0.66	267
Cruise	357,046	7.75	3333	628,414	14.48	6233
Descent	--	--	--	--	--	--
Climb and accelerate	32,465	0.66	254	--	--	--
Cruise	224,876	7.78	3346	--	--	--
Descent	--	--	--	--	--	--
Reserves	64,275	0.5	--	65,195	0.5	--
Trapped fuel	1,400	--	--	1,400	--	--
Total	758,723	17.35	3600 (radius)	758,723	15.64	6500

TABLE 1.- Concluded

Weights Summary for Baseline I

Component	Pounds	Percent W_g
Airframe structure	704,651	35.72
Wing	283,066	14.35
Fuselage	258,951	13.13
Horizontal tail	21,020	1.07
Vertical tail	12,850	.65
Nacelles	35,507	1.80
Alighting gear	93,257	4.73
Propulsion	105,209	5.33
Engines (8)	97,053	4.92
Fuel system	8,156	.41
Fixed equipment	53,107	2.69
Hyd. and pneu.	11,836	0.60
Electrical	5,384	.27
Avionics	5,892	.30
Instruments	1,939	.10
De-ice/air cond.	3,643	.18
Aux. power sys.	919	.05
Furnish and eqpt.	12,750	.65
Flight controls	10,744	.54
Fuel	758,723	38.46
Payload	350,850	17.79
Crew (5)	850	.04
Cargo	350,000	17.74
Calculated gross weight	1,972,540	100.00

TABLE 2.- BASELINE II CHARACTERISTICS

General

W_g = 2,198,673 lb

W/S = 130 lb/ft^2

T/W = 0.246

$N_{Z_{LIM}}$ = 2.50

$N_{Z_{ULT}}$ = 3.75

Engine

No. = 8

Type = Turbofan

l = 16.5 ft

d = 8.9 ft

T_{SLS} = 67,619 lb

sfc_{cr} = 0.62 lb/lb-hr

$(T/W)_{ENG}$ = 5.71

BPR = 5.10

FPR = 1.6

TIT = 28.35 °R

CPR = 22.9

Geometry

	Wing	Horiz. tail	Vert. tail
S, ft^2	16,922	2708	2708
S_W, ft^2	34,087	5445	3861
b, ft	349	113	58
Λ_{LE}, deg	33.3	28.9	36.6
$\Lambda_{c/4}$, deg	30.4	24.5	34.9
AR	7.18	4.74	1.24
λ	0.34	0.37	0.80
M.A.C., ft	52.5	25.6	46.9
c_R, ft	72.5	34.9	51.9
c_t, ft	24.6	12.9	41.5
$(t/c)_R$	0.13	0.105	0.13
$(t/c)_t$	0.11	0.105	0.13

TABLE 2.- Continued

Fuselage

l = 326 ft

d = 31.4 ft

S_W = 27,956 ft^2

Cargo bay

l = 227 ft

w = 29.7 ft

h = 14.2 ft

Performance

$W_{P_{DES}}$ = 350,000 lb

$M_{cr_{DES}}$ = 0.80

TOFL = 8000 ft

Max. radius = 3600 n.mi.

Max. range = 6450 n.mi.

M_{max} = 0.84

t_{ENDUR} = 20.4 hr

h_{BC} ≈ 33,078 ft

Mission summary

	Radius			Range		
	Fuel, lb	Time, hr	Distance, n.mi.	Fuel, lb	Time, hr	Distance, n.mi.
Warm up, taxi, takeoff	34,000	--	--	17,000	--	--
Climb and accelerate	51,568	0.52	219	71,977	0.87	311
Cruise	462,180	7.35	3381	777,543	13.36	6139
Descent	--	--	--	--	--	--
Climb and accelerate	33,558	.53	206	--	--	--
Cruise	286,293	7.40	3394	--	--	--
Descent	--	--	--	--	--	--
Reserves	77,777	.5	--	78,856	.5	--
Trapped fuel	1,400	--	--	1,400	--	--
Total	946,776	16.3	3600 (radius)	946,776	14.23	6450

TABLE 2.- Concluded

Weights Summary for Baseline II

Component	Pounds	Percent W_g
Airframe structure	724,125	32.93
Wing	277,925	12.64
Fuselage	266,367	12.11
Horizontal tail	23,210	1.06
Vertical tail	14,395	.65
Nacelles	39,582	1.80
Alighting gear	102,647	4.67
Propulsion	122,304	5.56
Engines (8)	112,823	5.13
Fuel system	9,481	.43
Fixed equipment	54,618	2.48
Hyd. and pneu.	13,194	.60
Electrical	5,633	.26
Avionics	6,289	.29
Instruments	2,068	.09
De-ice/air cond.	3,643	.17
Aux. power sys.	956	.04
Furnish. and eqpt.	12,750	.58
Flight controls	10,085	.46
Fuel	946,776	43.06
Payload	350,850	15.96
Crew (5)	850	.04
Cargo	350,000	15.92
Calculated gross weight	2,198,673	100.00

TABLE 3.- SENSITIVITY FACTORS FOR BASELINE I

<u>Design parameters</u>	
Fuselage diameter	0.54
Fuselage length	.45
Wing loading	-.25
Thrust/weight ratio	.22
Wing aspect ratio	-.11
Wing sweep (c/4)	.11
Wing taper ratio	.04
Wing thickness/chord ratio	.03
<u>Efficiency indicators</u>	
Lift/drag ratio	-0.78
Minimum drag	.47
Wing weight	.42
Fuselage weight	.39
Induced drag	.37
Propulsion weight	.16
Landing gear weight	.13
<u>Mission parameters</u>	
Mission radius	0.78
Payload	.75
Takeoff distance	-.29
Fuel reserve (% initial)	.06
Loiter time	.04

TABLE 4.- BASELINE III CHARACTERISTICS

General

W_g = 1,317,077 lb

W/S = 125 lb/ft^2

T/W = 0.238

$N_{Z_{LIM}}$ = 2.50

$N_{Z_{ULT}}$ = 3.75

Engine

No. = 8

Type = Turbofan

l = 13.3 ft

d = 6.8 ft

T_{SLS} = 39,103 lb

sfc_{cr} = 0.60 lb/lb-hr

$(T/W)_{ENG}$ = 5.85

BPR = 5.10

FPR = 1.6

TIT = 2835 °R

CPR = 22.9

Geometry

	Wing	Horiz. tail	Vert. tail
S, ft^2	10,576	1692	1692
S_W, ft^2	21,304	3403	2158
b, ft	325	90	46
Λ_{LE}, deg	27.3	28.9	36.6
$\Lambda_{c/4}$, deg	25.0	24.5	34.9
Æ	10.0	4.74	1.24
λ	0.34	0.37	0.80
M.A.C., ft	35.2	20.2	37.1
c_R, ft	48.5	27.6	41.0
c_t, ft	16.5	10.2	32.8
$(t/c)_R$	0.13	0.105	0.13
$(t/c)_t$	0.11	0.105	0.13

TABLE 4.- Continued

Fuselage

l = 326 ft

d = 31.4 ft

S_W = 27,956 ft^2

Cargo bay

l = 227 ft

w = 29.7 ft

h = 14.2 ft

Performance

$W_{P_{DES}}$ = 350,000 lb

$M_{cr_{DES}} \doteq 0.75$

TOFL = 8000 ft

Max. range = 3500 n.mi.

h_{BC} = 33,468 ft

Mission summary

	Range		
	Fuel, lb	Time, hr	Distance, n.mi.
Warm up, taxi, takeoff	9,831	--	--
Climb and accelerate	33,436	0.64	250
Cruise	255,728	7.54	3250
Descent	--	--	--
Reserves	35,962	0.5	--
Trapped fuel	1,400	--	--
Total	336,357	8.68	3500

TABLE 4.- Concluded

Weights Summary for Baseline III

Component	Pounds	Percent W_g
Airframe structure	515,201	39.12
Wing	172,075	13.06
Fuselage	233,148	17.70
Horizontal tail	13,454	1.02
Vertical tail	7,461	.57
Nacelles	23,709	1.80
Alighting gear	65,355	4.96
Propulsion	68,933	5.23
Engines (8)	63,589	4.83
Fuel system	5,344	.41
Fixed equipment	45,737	3.47
Hyd. and pneu.	7,903	.60
Electrical	4,553	.35
Avionics	4,742	.36
Instruments	1,568	.12
De-ice/air cond.	3,643	.28
Aux. power sys.	809	.06
Furnish. and eqpt.	12,750	.97
Flight controls	9,768	.74
Fuel	336,357	25.54
Payload	350,850	26.64
Crew (5)	850	.06
Cargo	350,000	26.57
Calculated gross weight	1,317,077	100.00

TABLE 5.- STAND-ALONE WEIGHTS CORRELATION OF C-5A

Component	Computed, lb	% WG	Actual C-5A, lb	% Error	% Error of actual WG
Airframe structure	267,415	36.45	258,163	3.58	1.271
Wing	87,471	11.92	82,045	6.61	0.745
Fuselage	121,455	16.55	116,049	4.66	0.743
Horizontal tail	6,918	0.94	6,793	1.84	.017
Vertical tail	6,018	.82	5,603	7.40	.057
Nacelles	13,104	1.79	9,586	37.70	.483
Alighting gear	32,450	4.42	38,088	-14.80	-.774
Propulsion	37,725	5.14	36,262	4.03	0.201
Engines (4)	34,800	4.74	33,804	2.95	.137
Fuel system	2,924	.40	2,458	18.97	.064
Fixed equipment	26,018	3.55	30,838	-15.63	-0.662
Hyd. and pneu.	4,368	.60	3,978	9.80	.054
Electrical	4,450	.61	3.451	28.95	.137
Avionics	2,060	.28	3,890	-47.05	-.251
Instruments	1,172	.16	938	24.90	.032
De-ice/air cond.	364	.05	3,640	-89.99	-.450
Aux. power sys.	711	.10	971	-26.77	-.036
Furnish and eqpt.	7,735	1.05	6,835	13.17	.124
Flight controls	5,158	.70	7,135	-27.71	-.272
Fuel	319,326	43.52	319,326	Input	Input
Payload	83,261	11.35	83,261	Input	Input
Flight crew (6)	1,020	.14	1,020	Input	Input
Baggage	120	.02	120	Input	Input
Cargo	82,121	11.19	82,121	Input	Input
Gross Weight	733,745	100.00	728,000		

TABLE 6.- OVERALL DESIGN CORRELATION OF C-5A

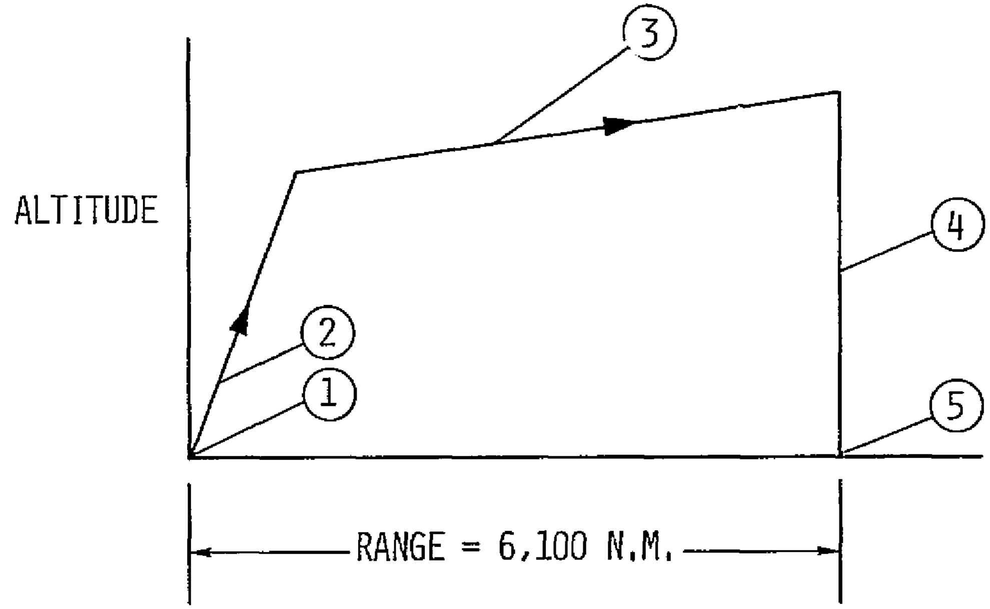
Component	ACSYNT-simulated C-5A, Basic mission, Radius, lb	ACSYNT-simulated C-5A, Basic mission, Range, lb	ACSYNT-optimized aircraft, Basic mission, Range, lb	Actual C-5A lb
Airframe structure	255,770	255,797	257,911	258,163
Wing	82,445	82,456	84,643	82,045
Fuselage	111,578	111,583	111,548	116,049
Horizontal tail	7,090	7,091	7,100	6,793
Vertical tail	3,862	3,862	3,871	5,603
Nacelles	12,755	12,758	12,742	9,586
Alighting gear	38,040	38,047	38,007	38,088
Propulsion	36,262*	36,262*	38,628	36,262
Engines (4)	33,804*	33,804*	35,634	33,804
Fuel system	2,458*	2,458*	2,994	2,458
Fixed equipment	30,879	30,881	30,903	30,838
Hyd. and pneu.	4,252	4,253	4,247	3,978
Electrical	3,521	3,521	3,519	3,451
Avionics	3,674	3,674	3,672	3,890
Instruments	1,161	1,161	1,160	938
De-ice/air cond.	3,643	3,643	3,643	3,640
Aux. power sys.	708	708	708	971
Furnish. and eqpt.	6,835	6,835	6,835	6,835
Flight controls	7,087	7,087	7,118	7,135
Fuel	119,933	119,829	114,451	115,800
Payload	265,850*	265,850*	265,850*	265,850
Crew (5)	850*	850*	850*	850
Cargo	265,000*	265,000*	265,000*	265,000
Gross weight	708,694	708,619	707,743	706,913

*Input to the ACSYNT program.

TABLE 7.- OPTIMIZED GEOMETRY CORRELATIONS

Geometry	Optimized aircraft*			Actual C-5A		
	Wing	H. tail	V. tail	Wing	H. tail	V. tail
Plan area, ft^2	6,219.9	995.2	995.2	6,200.0	965.8	961.1
Wetted area, ft^2	12,520.2	2,001.3	1,247.7	10,358.2	1,843.6	1,759.3
Span, ft	225.0	68.7	35.1	222.7	67.6	34.5
L. E. sweep, deg	27.7	28.9	36.6	27.4	28.9	36.6
c/4 sweep, deg	24.9	24.5	34.9	25.0	24.5	34.9
T. E. sweep, deg	15.6	9.4	29.4	17.3	9.4	29.4
Aspect ratio	8.1	4.7	1.2	8.0	4.7	1.2
Root chord, ft	41.5	21.2	31.5	45.5	20.8	30.9
Root thickness, in.	61.5	26.7	49.1	71.0	26.2	48.3
Root t/c	.123	.105	.130	.130	.105	.13
Tip chord, ft	13.8	7.8	25.2	15.3	7.7	24.7
Tip thickness, in.	17.3	9.9	39.3	20.2	9.7	38.5
Tip t/c	.104	.105	.130	.110	.105	.13
Taper ratio	.33	.37	.80	.34	.37	.80
Mean aero. chord, ft	30.0	15.5	28.4	30.9	15.3	28.0

*Optimized for basic C-5A range mission as computed by ACSYNT.



(1.) GROUND OPERATION AND TAKE-OFF FUEL ALLOWANCE 5 MIN. AT NORMAL POWER AT SEA LEVEL; NO RANGE CREDIT.

(2.) CLIMB AT CONSTANT INDICATED AIRSPEED ON COURSE AT NORMAL THRUST TO ALTITUDE FOR BEST LONG RANGE CRUISE AT A FIXED CRUISE MACH NUMBER*; RANGE CREDIT.

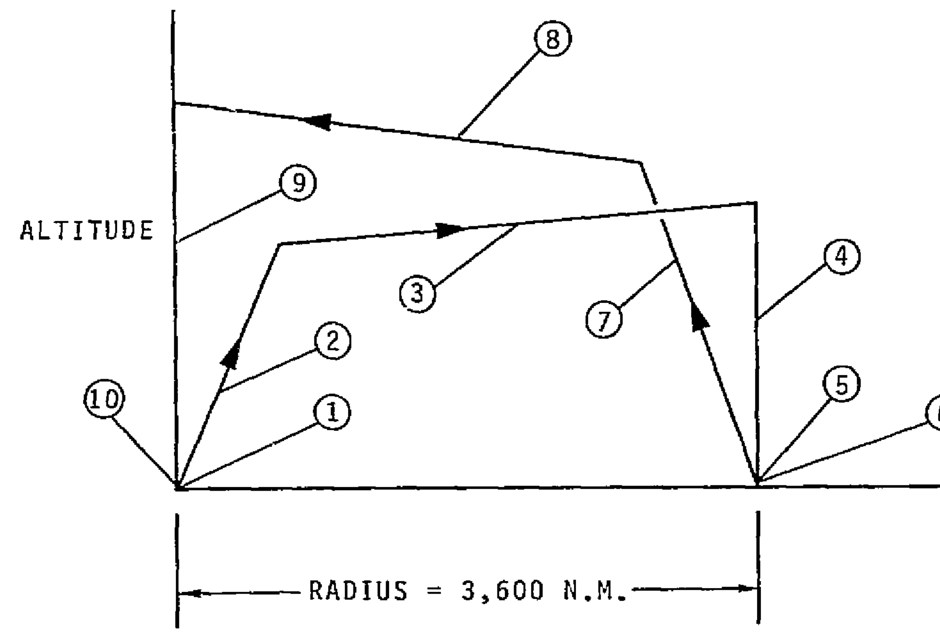
(3.) ACCELERATE FROM CLIMB SPEED TO FIXED CRUISE MACH NUMBER*, AT CONSTANT ALTITUDE, AND THEN CRUISE AT ALTITUDE FOR MAX. LONG RANGE CRUISE (BREGUET CRUISE) FOR A TOTAL DISTANCE OF 6,100 N.M. FROM TAKE-OFF POINT.

(4.) DESCEND FROM CRUISE ALTITUDE TO SEA LEVEL; NO FUEL COST; NO RANGE CREDIT.

(5.) FUEL ALLOWANCE FOR RESERVES AND LANDING IS THE SUM OF 5% OF INITIAL FUEL AND 30 MIN. LOITER AT VELOCITY FOR MAXIMUM ENDURANCE AT SEA LEVEL.

*CRUISE MACH NUMBER IS A VARIABLE IN THE STUDY

Figure 1.- Basic range mission.



1. GROUND OPERATION AND TAKE-OFF FUEL ALLOWANCE 10 MIN.* NORMAL POWER AT SEA LEVEL; NO RANGE CREDIT.
2. CLIMB AT CONSTANT INDICATED AIRSPEED ON COURSE AT NORMAL THRUST TO ALTITUDE FOR BEST LONG RANGE CRUISE AT A FIXED CRUISE MACH NUMBER*; RANGE CREDIT.
3. ACCELERATE FROM CLIMB SPEED TO FIXED CRUISE MACH NUMBER** AT CONSTANT ALTITUDE AND THEN CRUISE AT ALTITUDE FOR MAXIMUM LONG RANGE CRUISE (BREGUET CRUISE) FOR A TOTAL DISTANCE OF 3,600 N.M. FROM TAKE-OFF POINT.
4. DESCEND FROM CRUISE ALTITUDE TO SEA LEVEL; NO FUEL COST; NO RANGE CREDIT.
5. LANDING AND UNLOAD PAYLOAD; NO RANGE CREDIT.
6. TAKE-OFF; NO RANGE CREDIT.
7. CLIMB AS IN 2 ON RETURN LEG.
8. ACCELERATE AND CRUISE AS IN 3 ON RETURN LEG.
9. DESCEND AS IN 4.
10. FUEL ALLOWANCE FOR RESERVES AND LANDING IS THE SUM OF 5% INITIAL FUEL AND 30 MIN. LOITER AT VELOCITY FOR MAXIMUM ENDURANCE AT SEA LEVEL.

*USE OF 10 MIN. NORMAL POWER IN 1 PROVIDES FOR GROUND OPERATIONS AND 2 TAKE-OFFS IN 1 AND 6.

**CRUISE MACH NUMBER IS A VARIABLE IN THE STUDY.

Figure 2.- Basic radius mission.

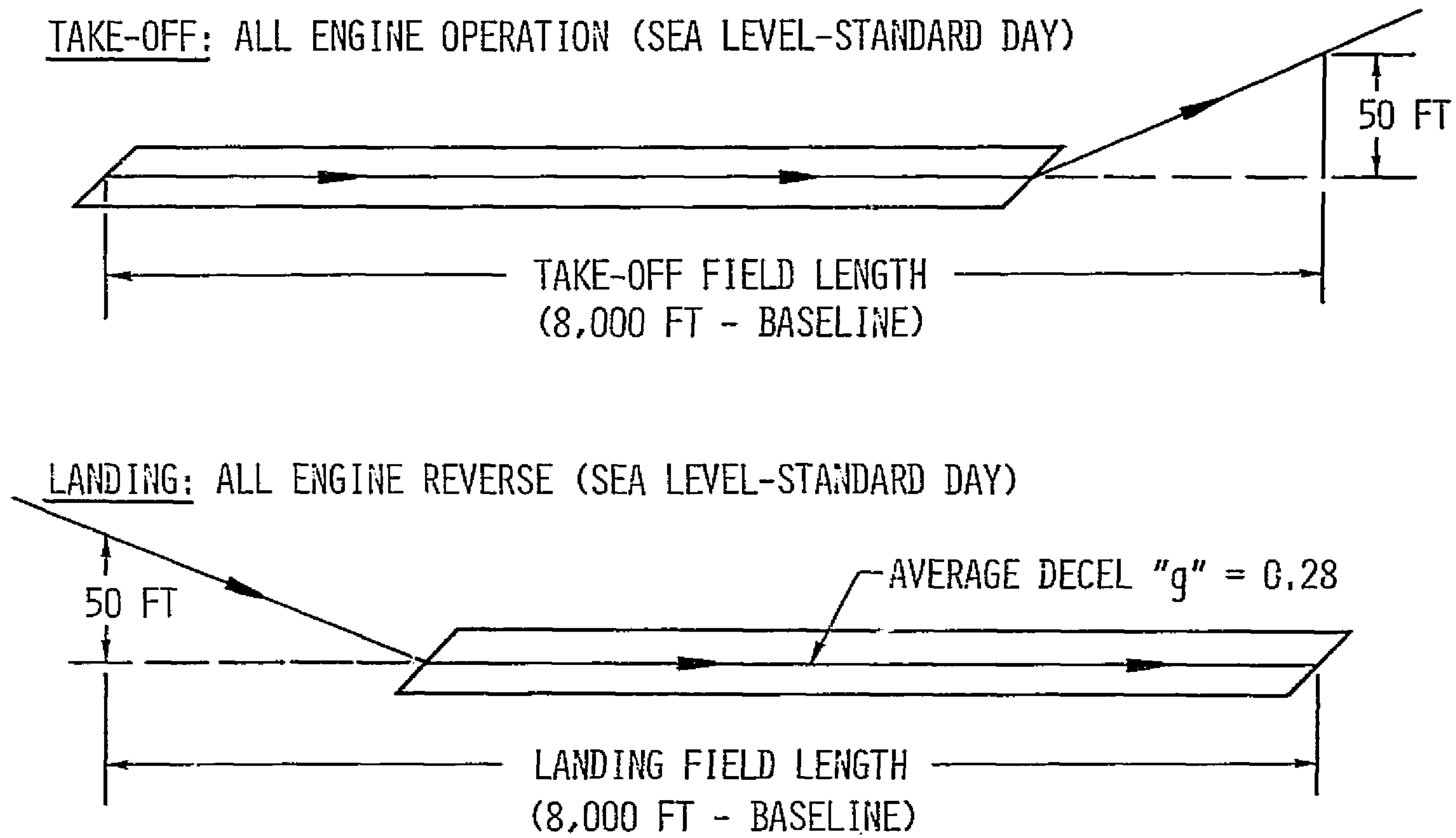
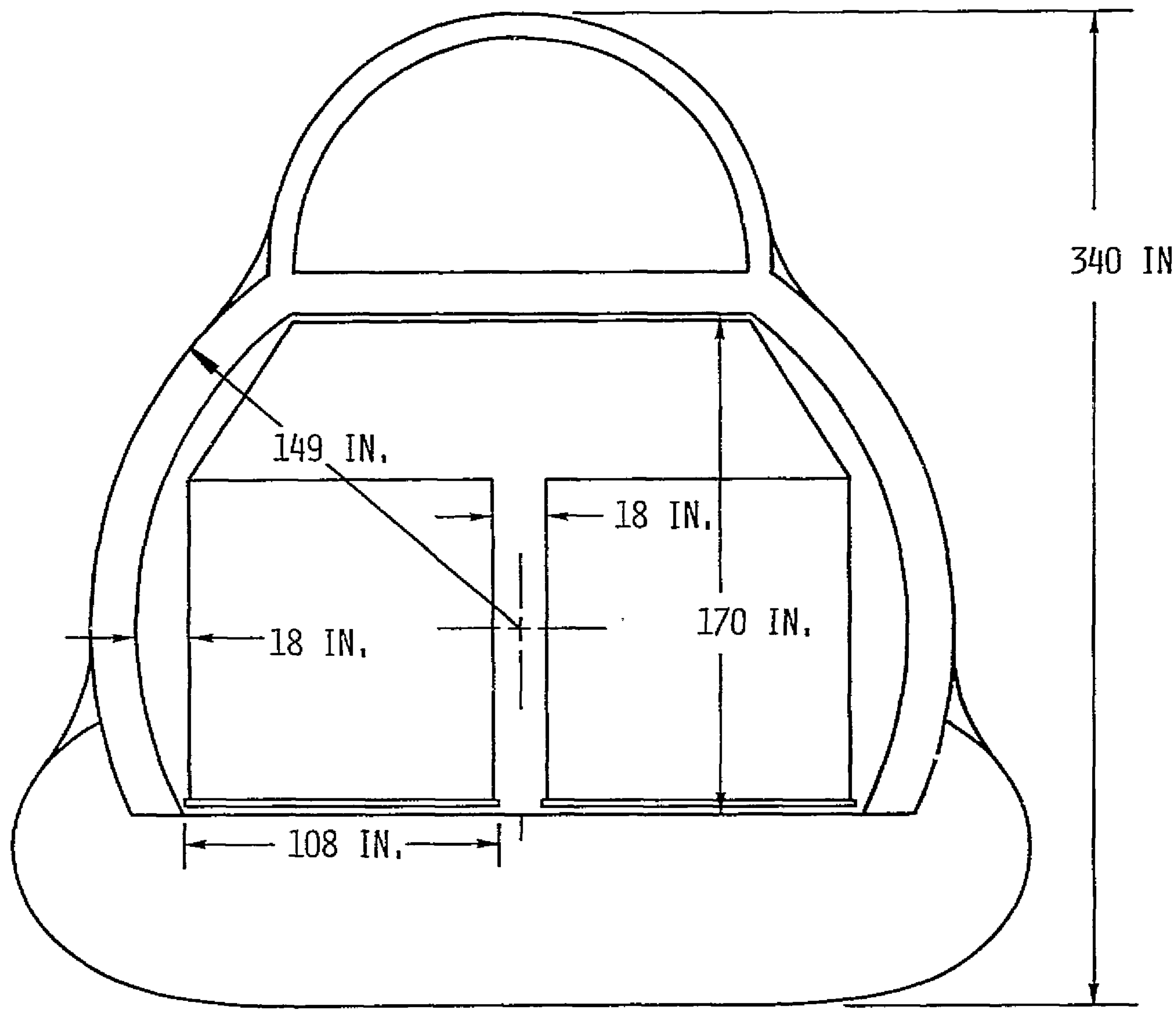
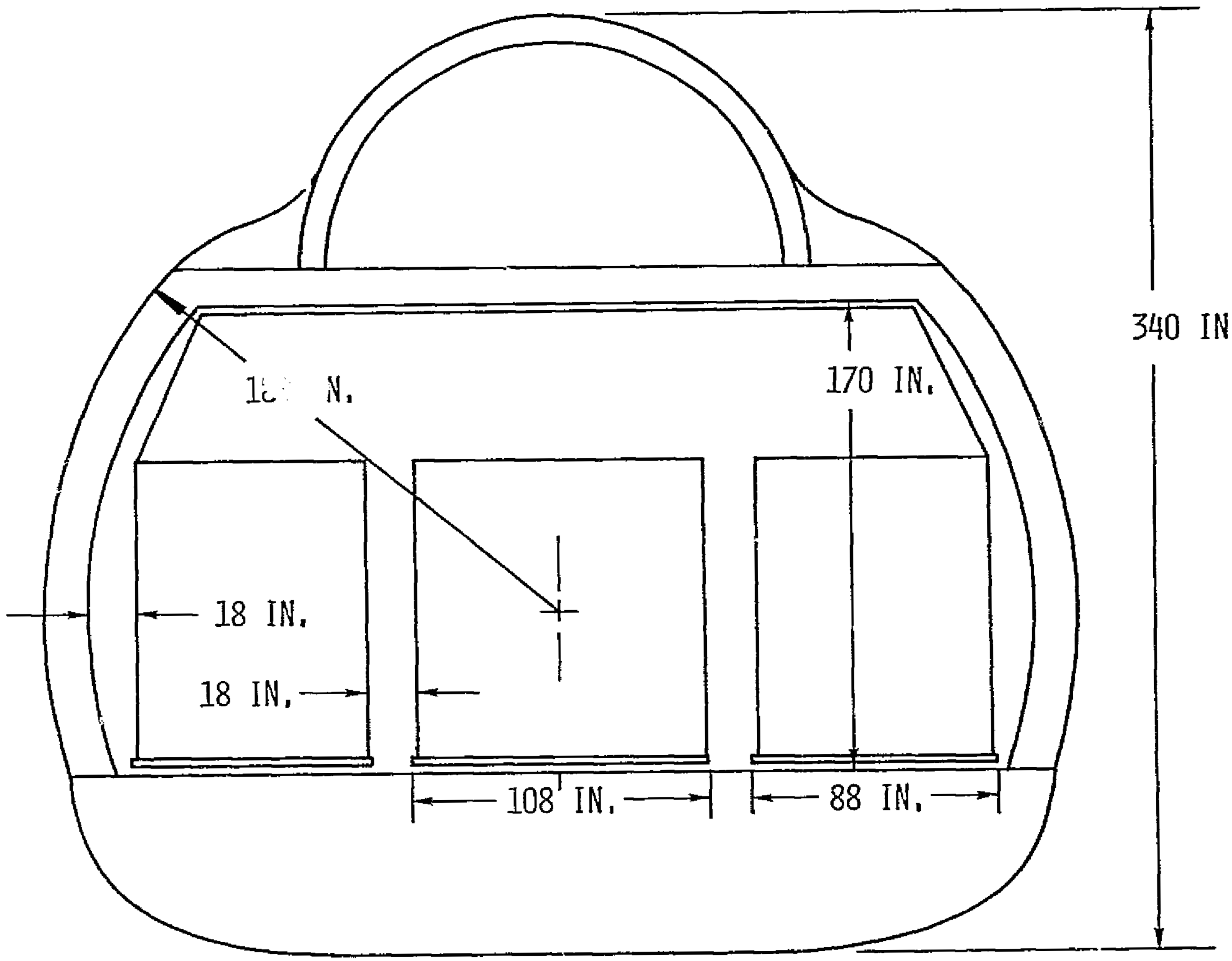


Figure 3.- Takeoff and landing field length diagrams; at takeoff gross weight.



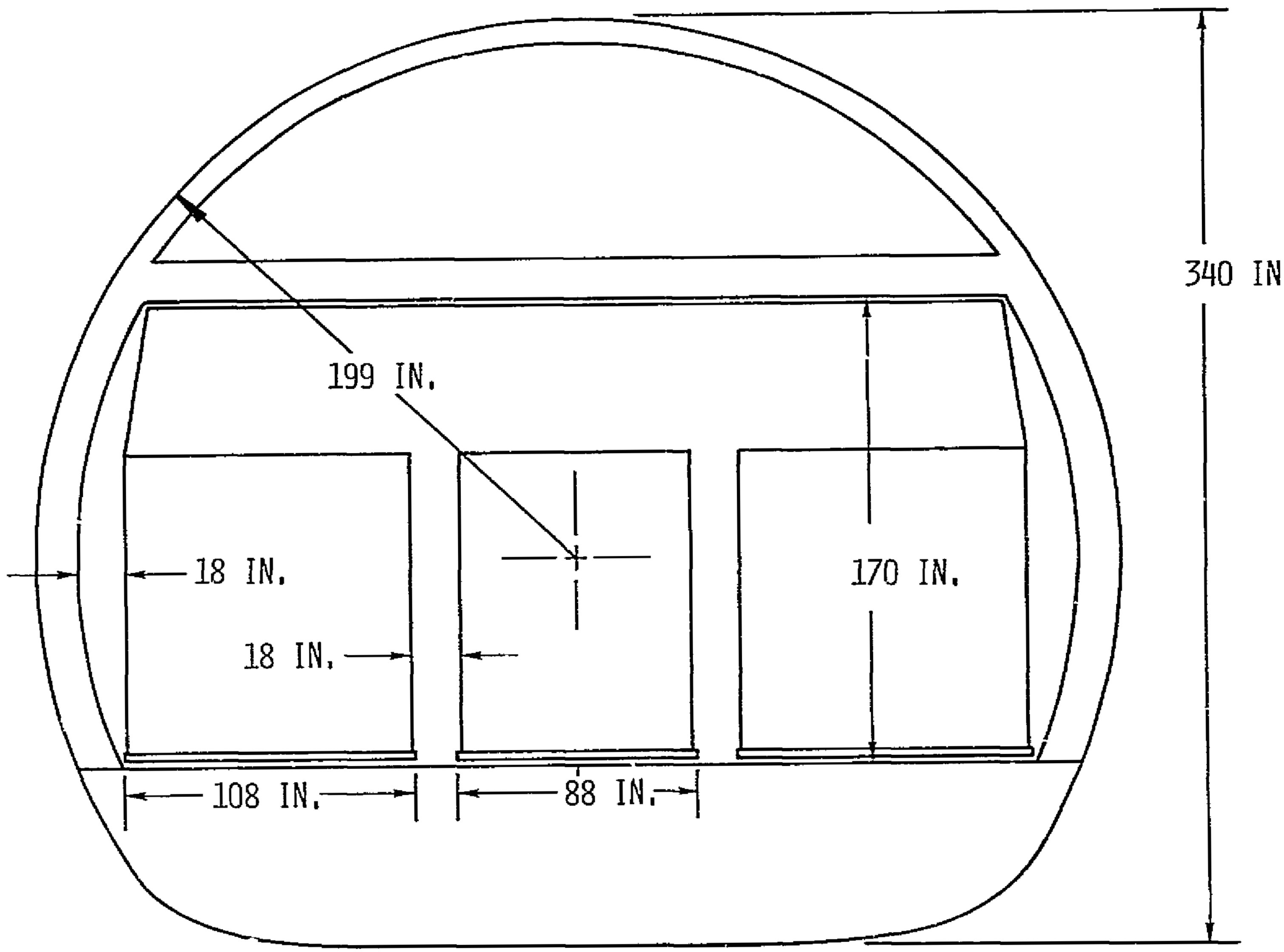
(a) Fuselage configuration A.

Figure 4.- Cross section of fuselage configurations; 463 L palletized cargo.



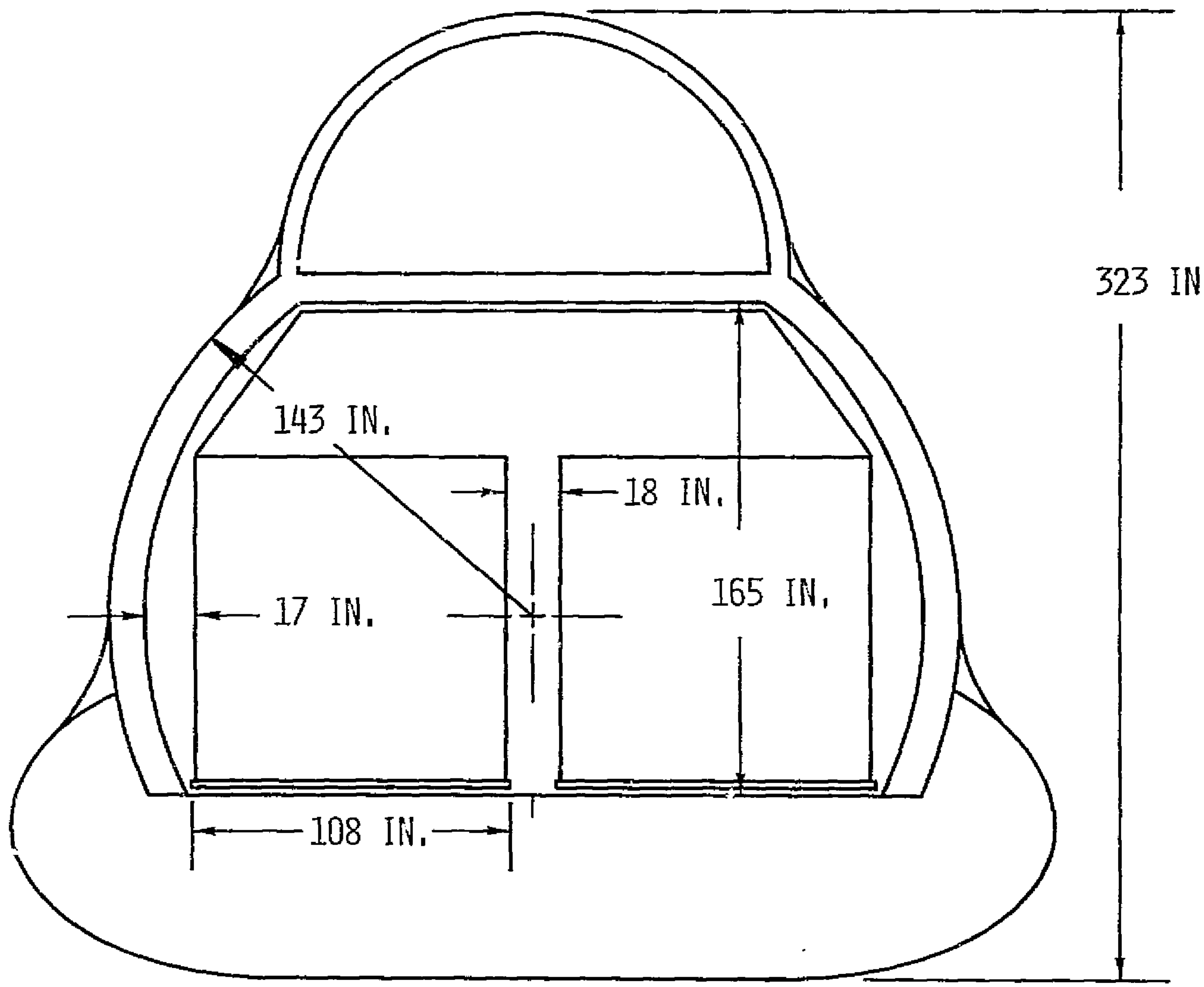
(b) Fuselage configuration B.

Figure 4.- Continued.



(c) Fuselage configuration C.

Figure 4.- Continued.



(d) C-5A fuselage cross section.

Figure 4.- Concluded.

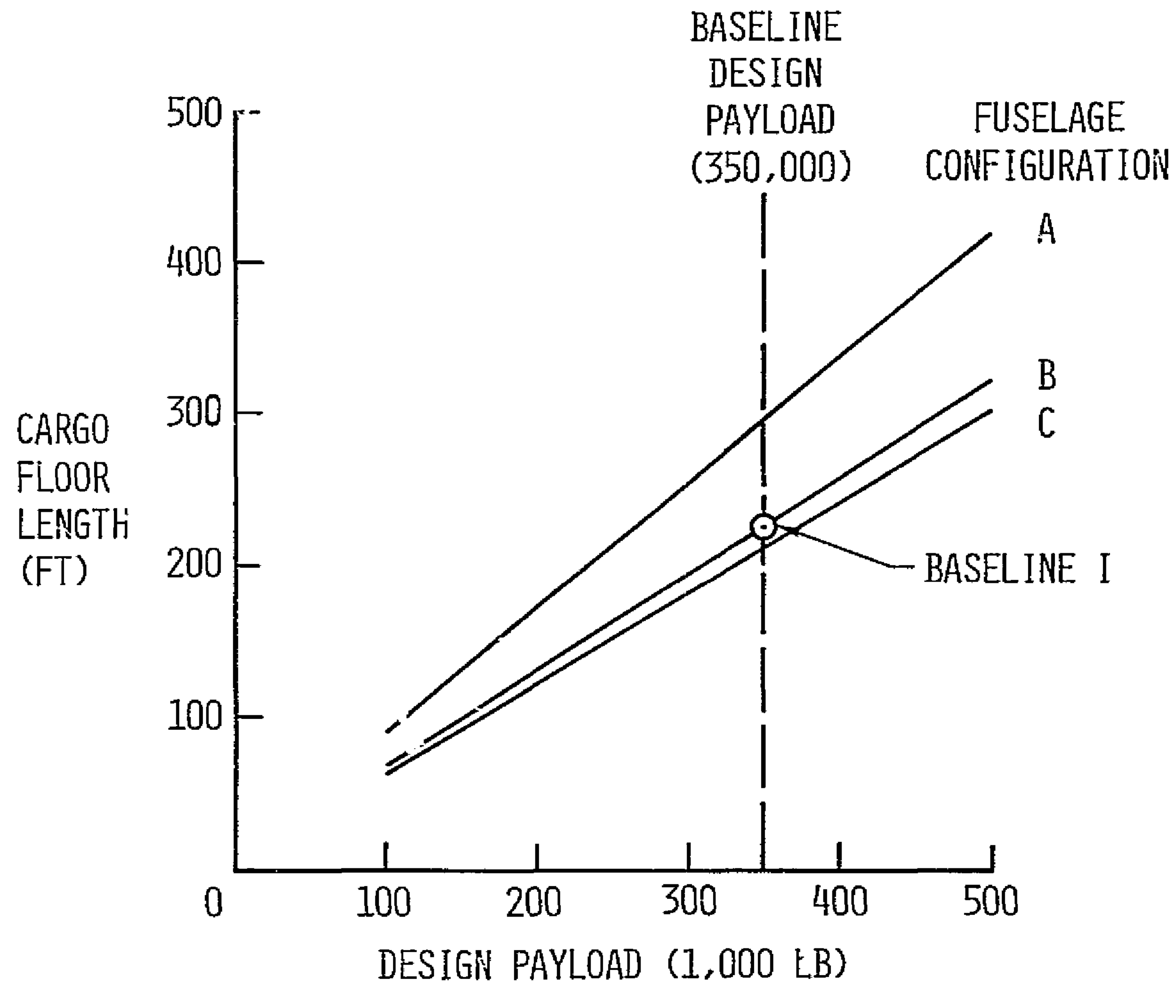
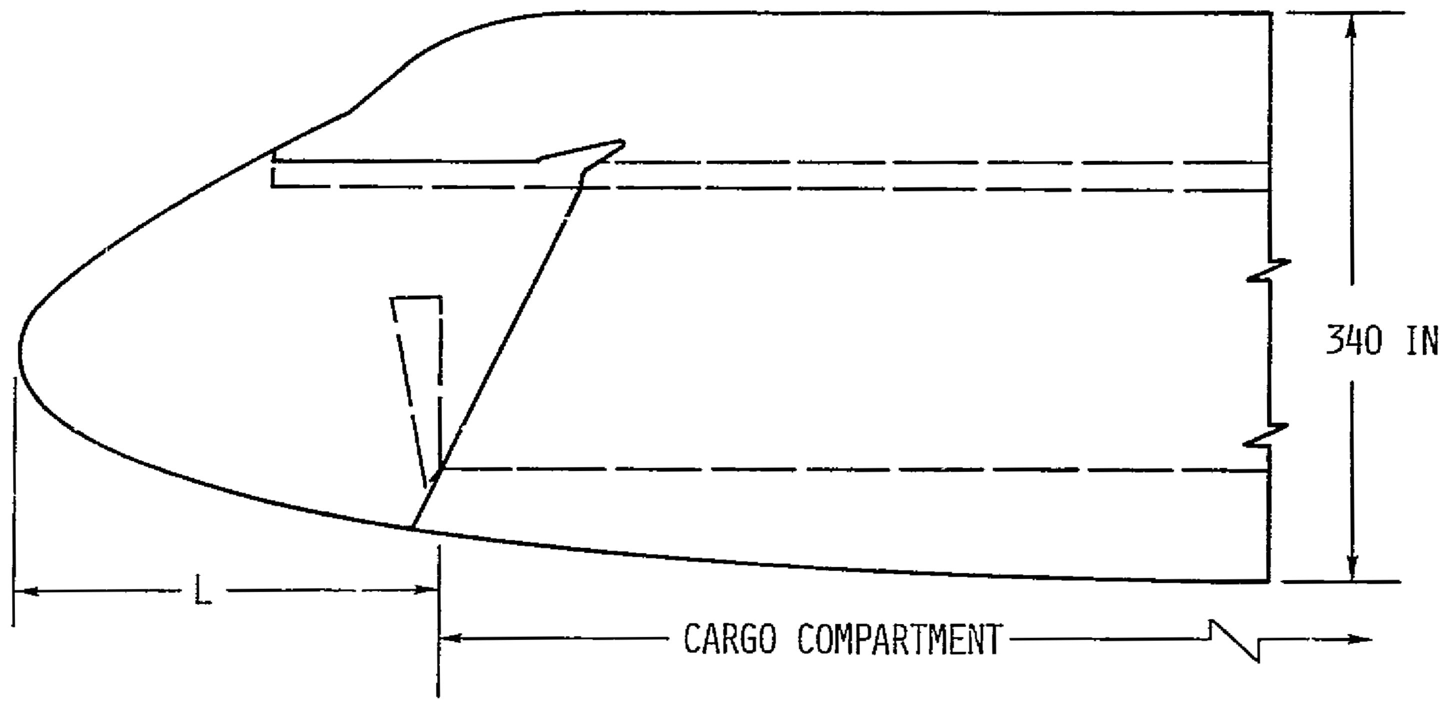


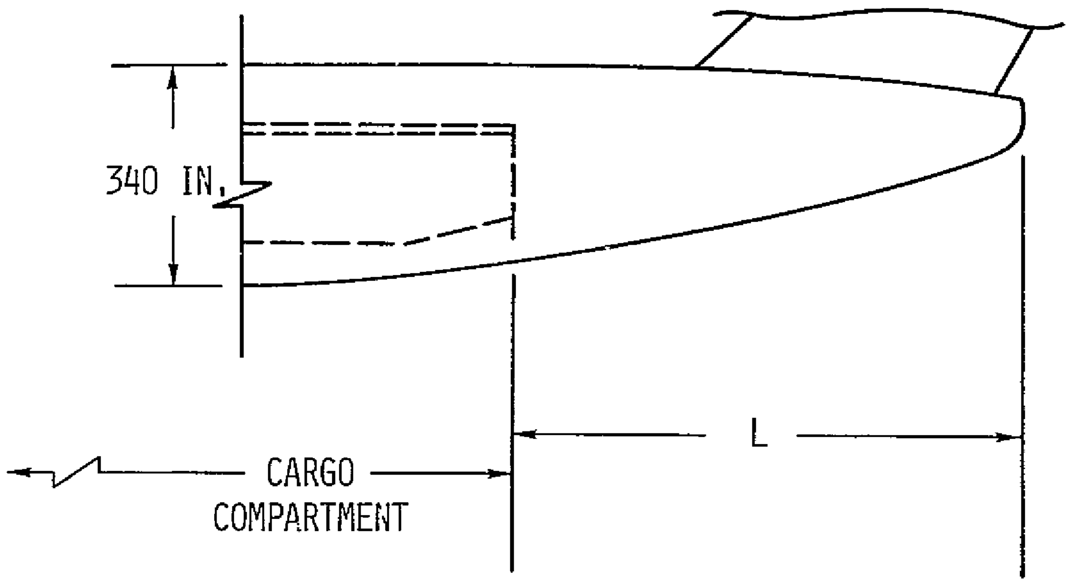
Figure 5.- Cargo compartment floor length; 463 L palletized cargo.



FUSELAGE CONFIGURATION	L (IN.)	L (FT)
A	253	21.0
B	285	23.75
C	300	25.0

(a) Forward fuselage.

Figure 6.- Fuselage contours.



FUSELAGE CONFIGURATION	L (IN.)	L (FT)
A	802	66.8
B	900	75.0
C	952	79.3

(b) Aft fuselage.

Figure 6.- Concluded.

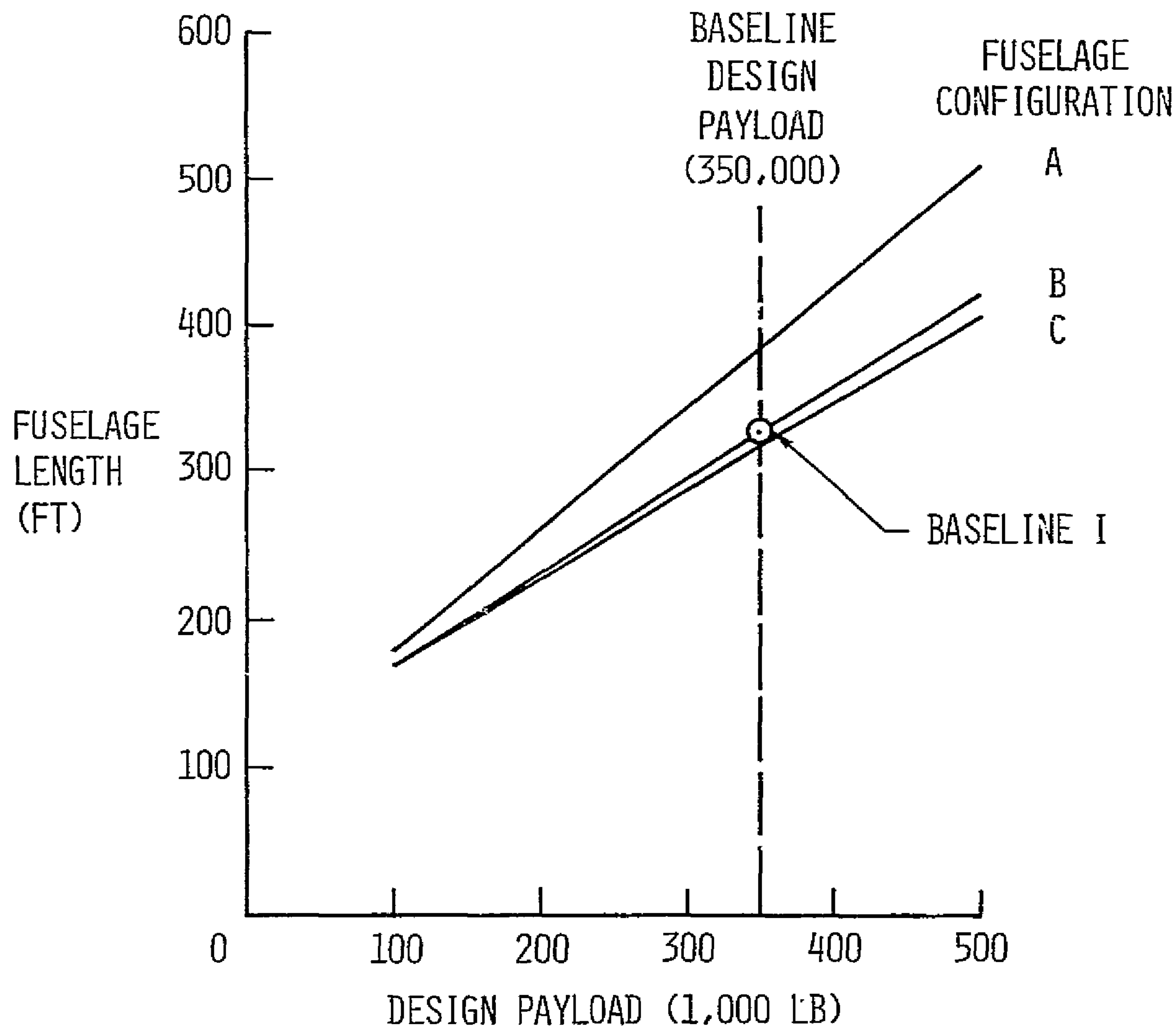
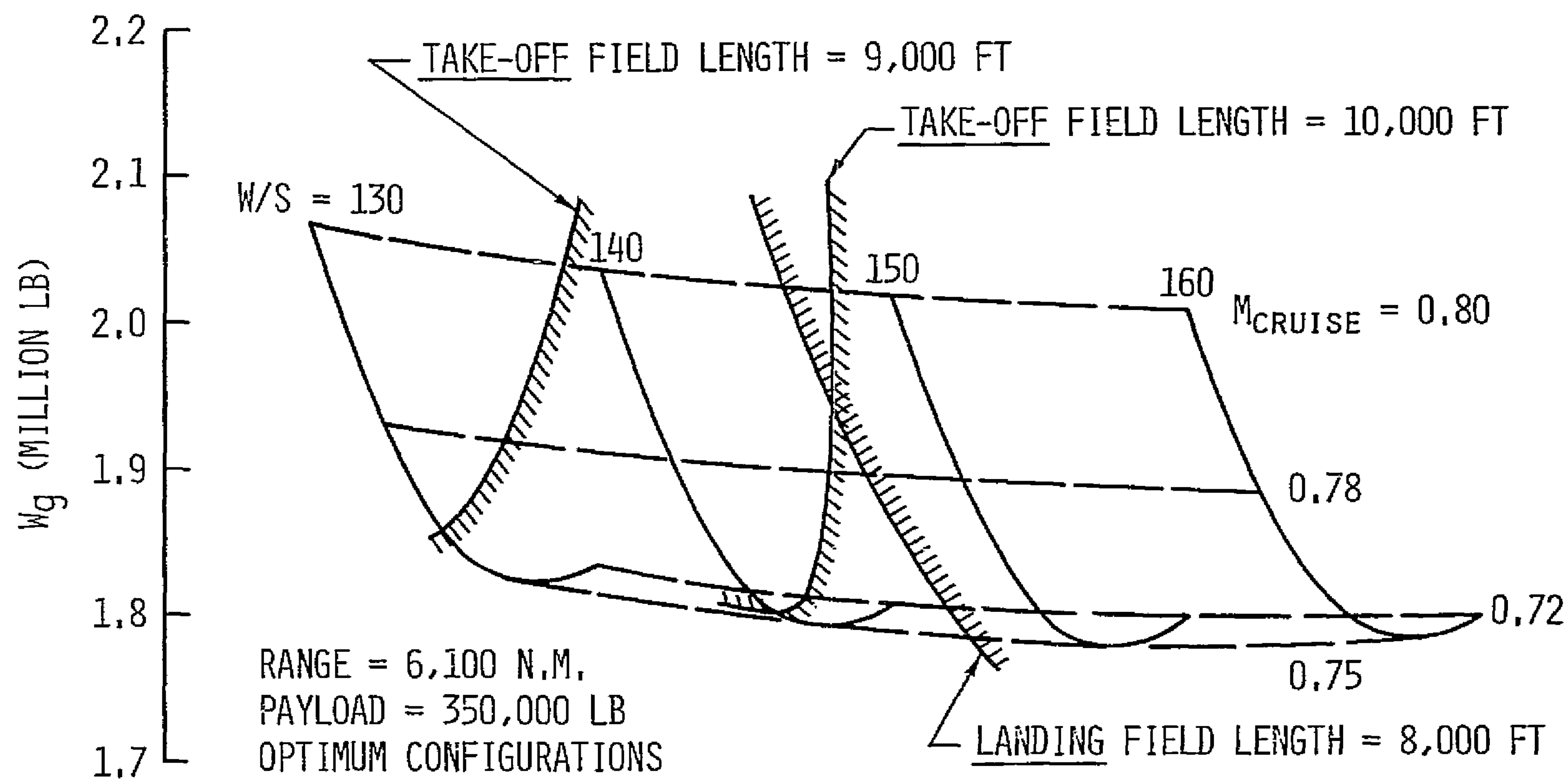
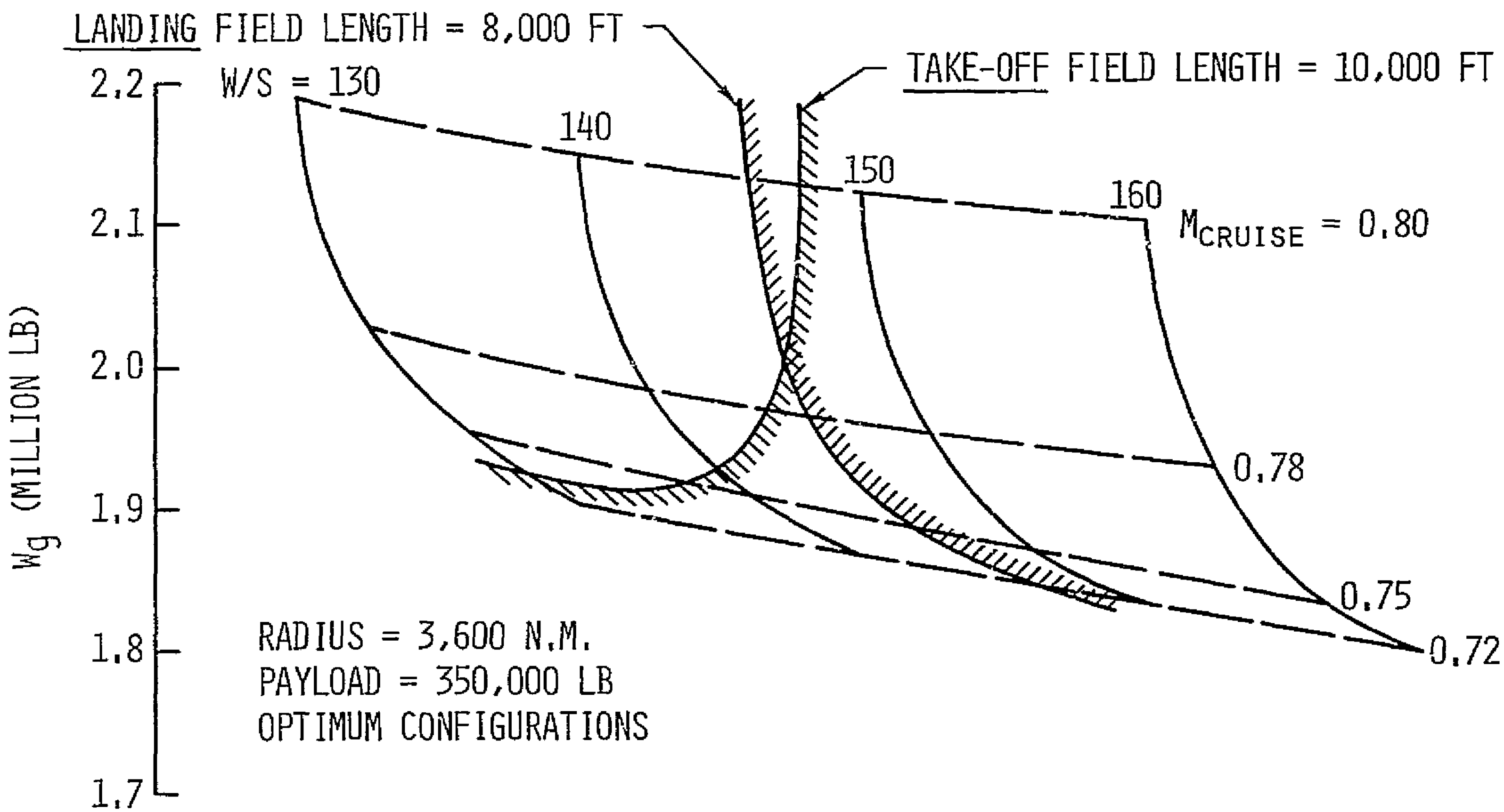


Figure 7.- Total fuselage length; 463 L palletized cargo.



(a) Basic range mission.

Figure 8.- Configurations optimized for minimum gross weight without takeoff or landing field length constraints.



(b) Basic radius mission.

Figure 8.- Concluded.

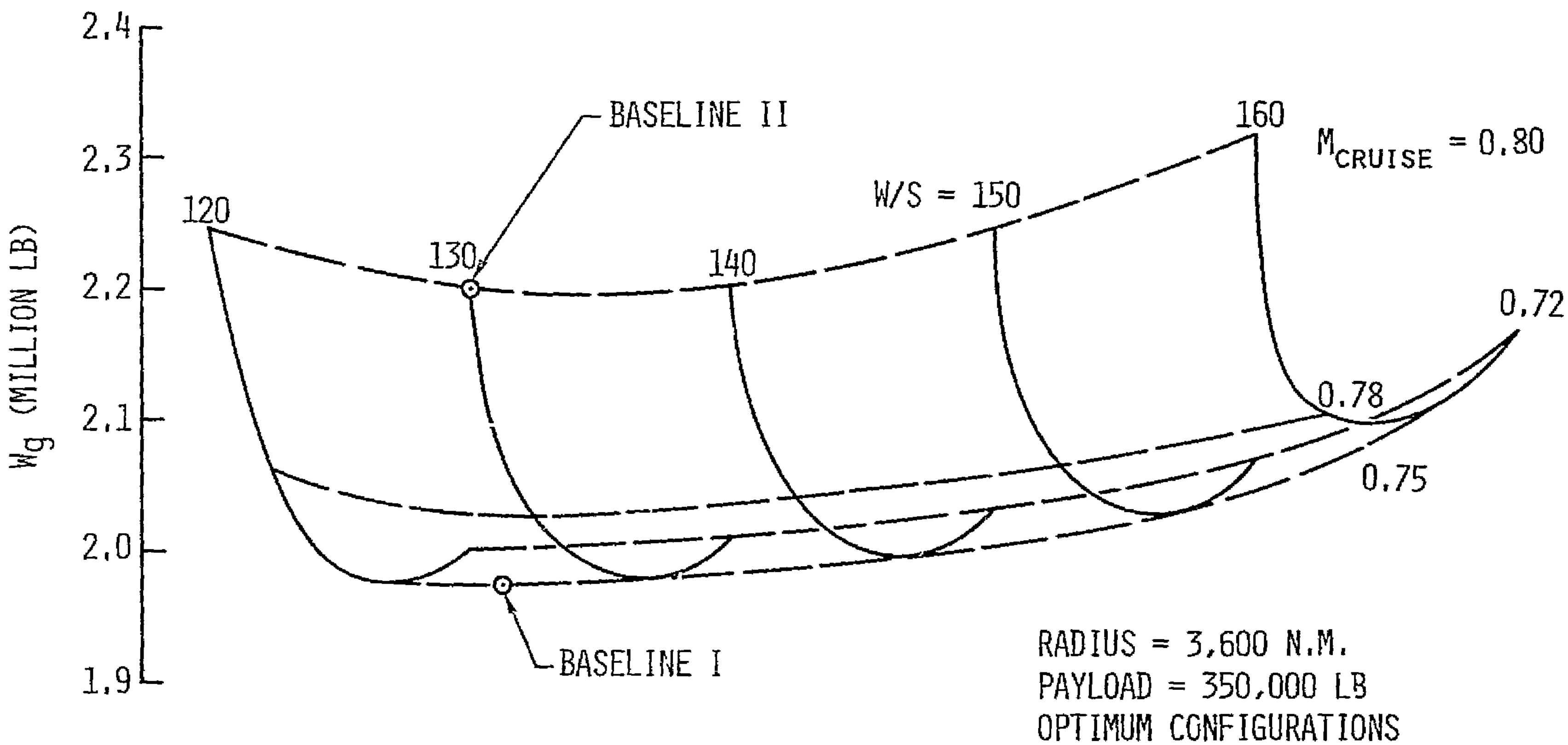


Figure 9.- Configurations optimized for minimum gross weight with an 8000-ft takeoff field length constraint; basic radius mission.

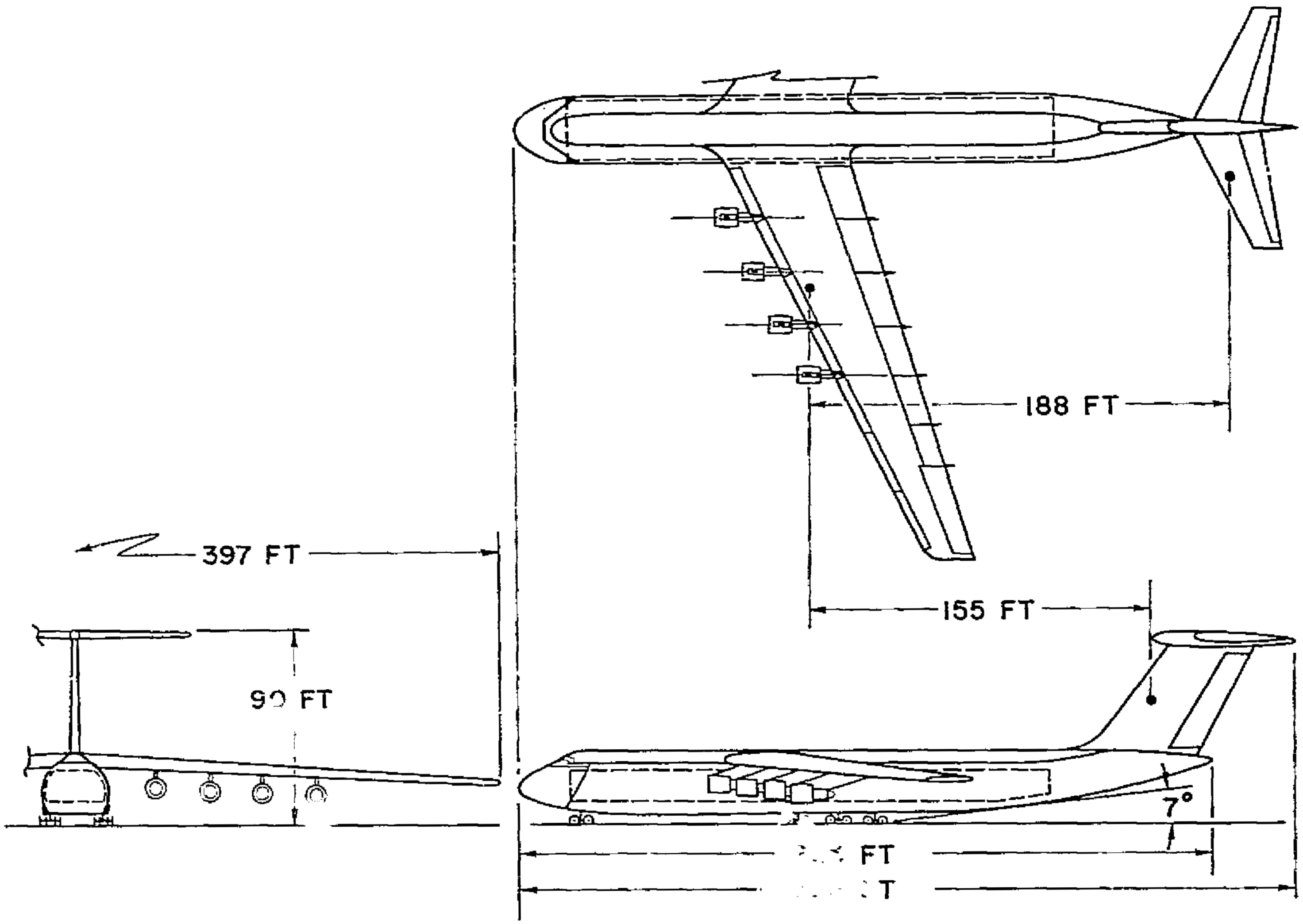


Figure 10.- Baseline I configuration arrangement.

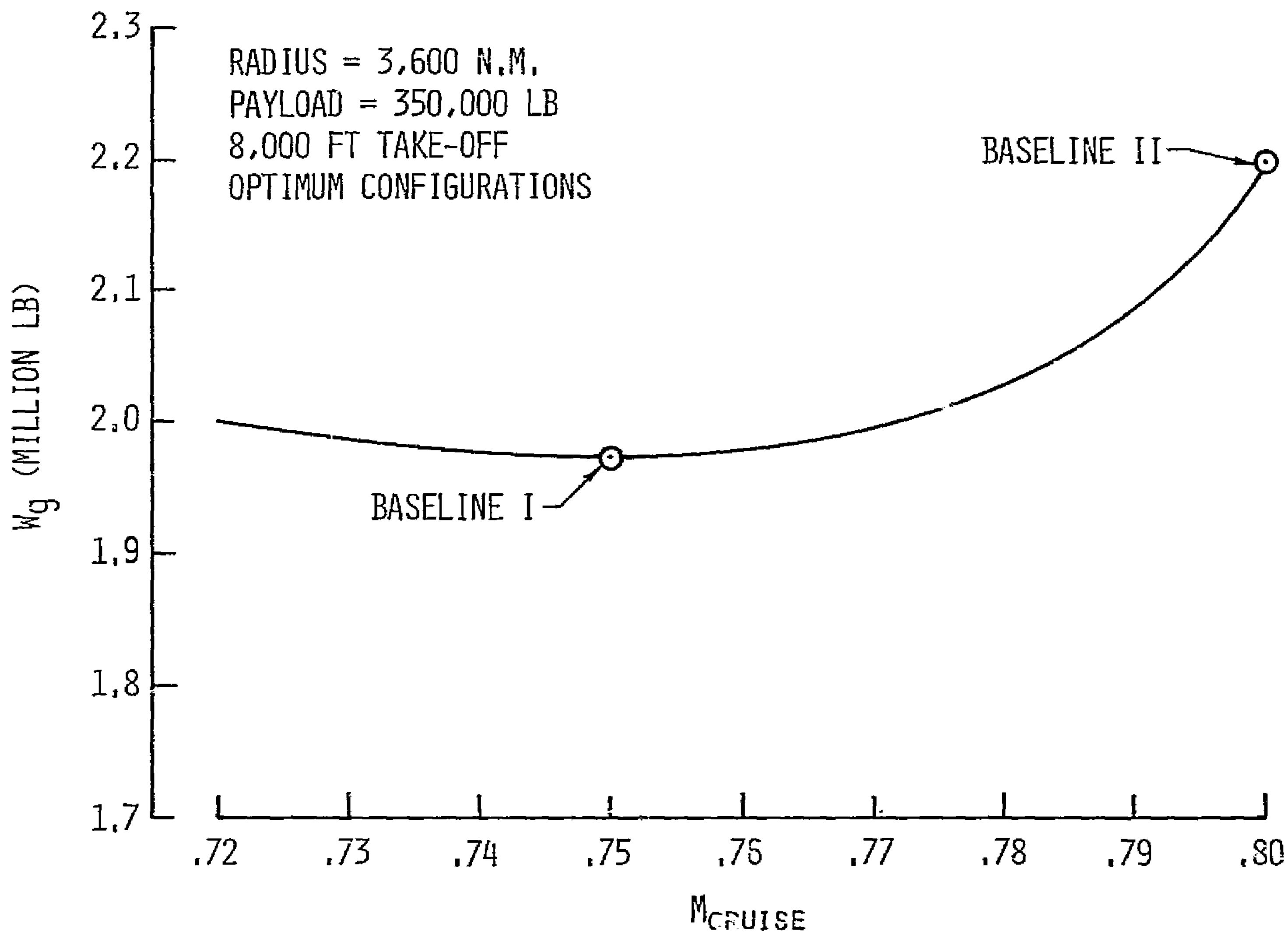


Figure 11.- Minimum gross weight configurations for the basic radius mission; 8000-ft takeoff field length.

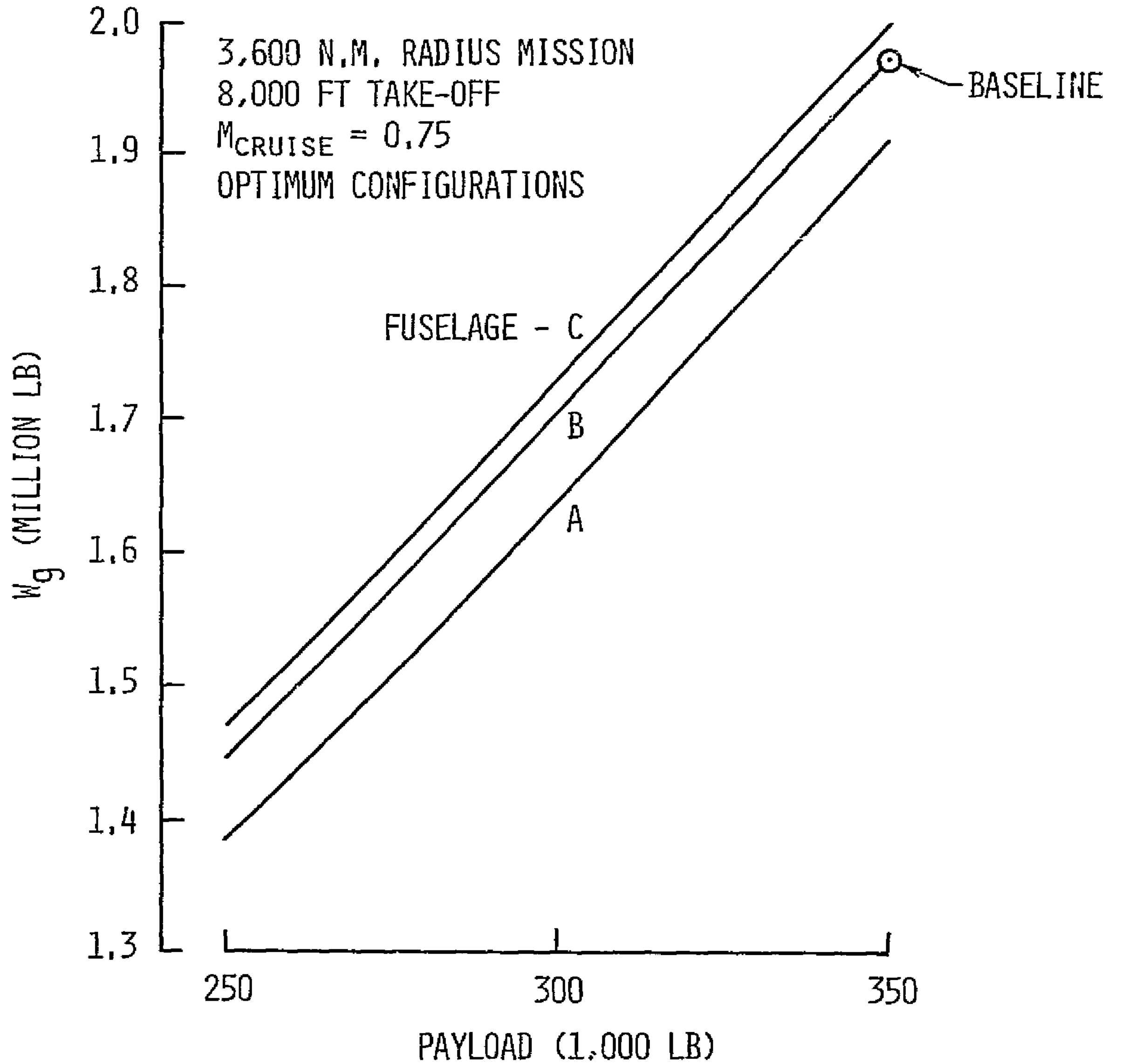


Figure 12.- Effect of payload and fuselage sizing on configurations optimized for minimum gross weight.

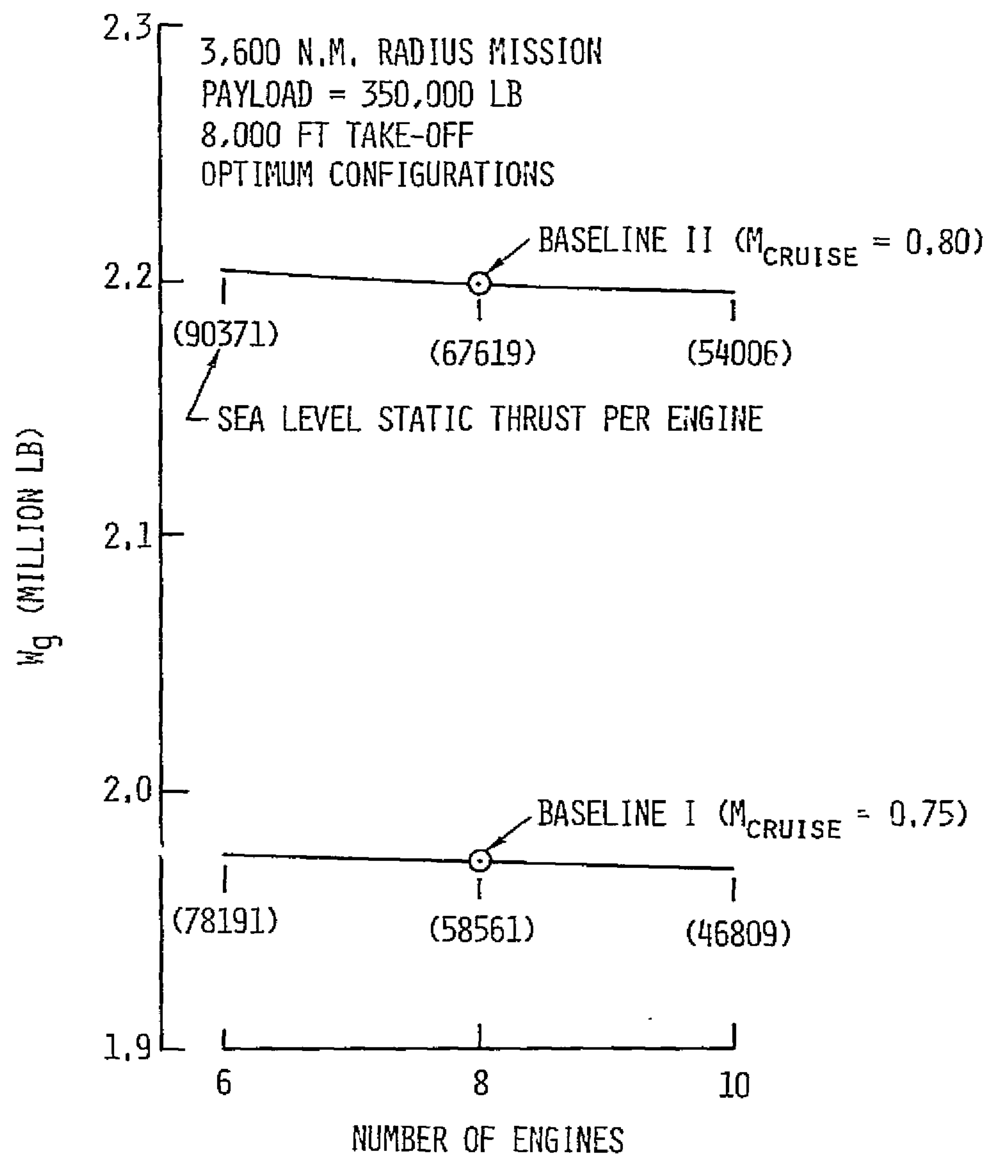


Figure 13.- Effect of number of engines on configurations optimized for minimum gross weight.

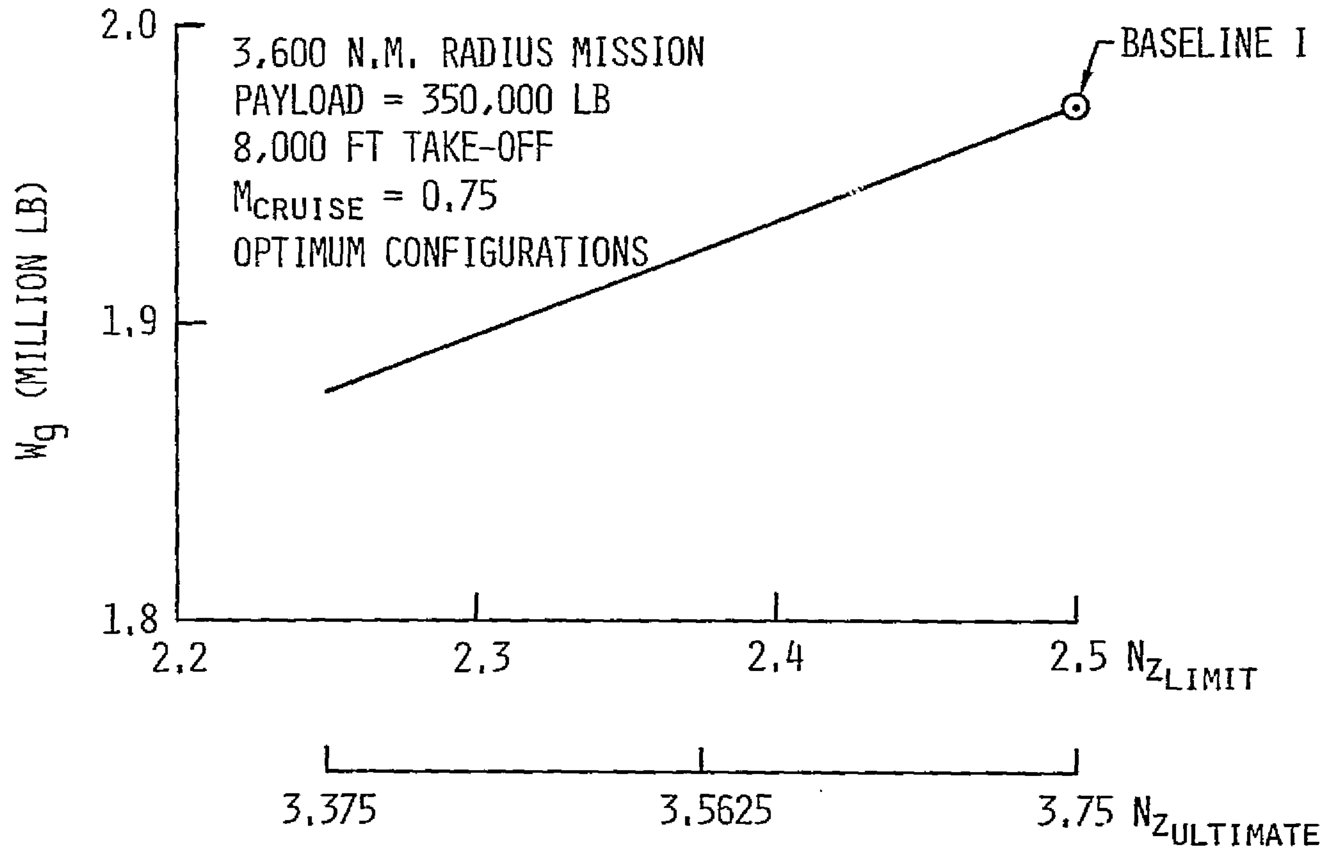


Figure 14.- Effect of load factor on configurations optimized for minimum gross weight.

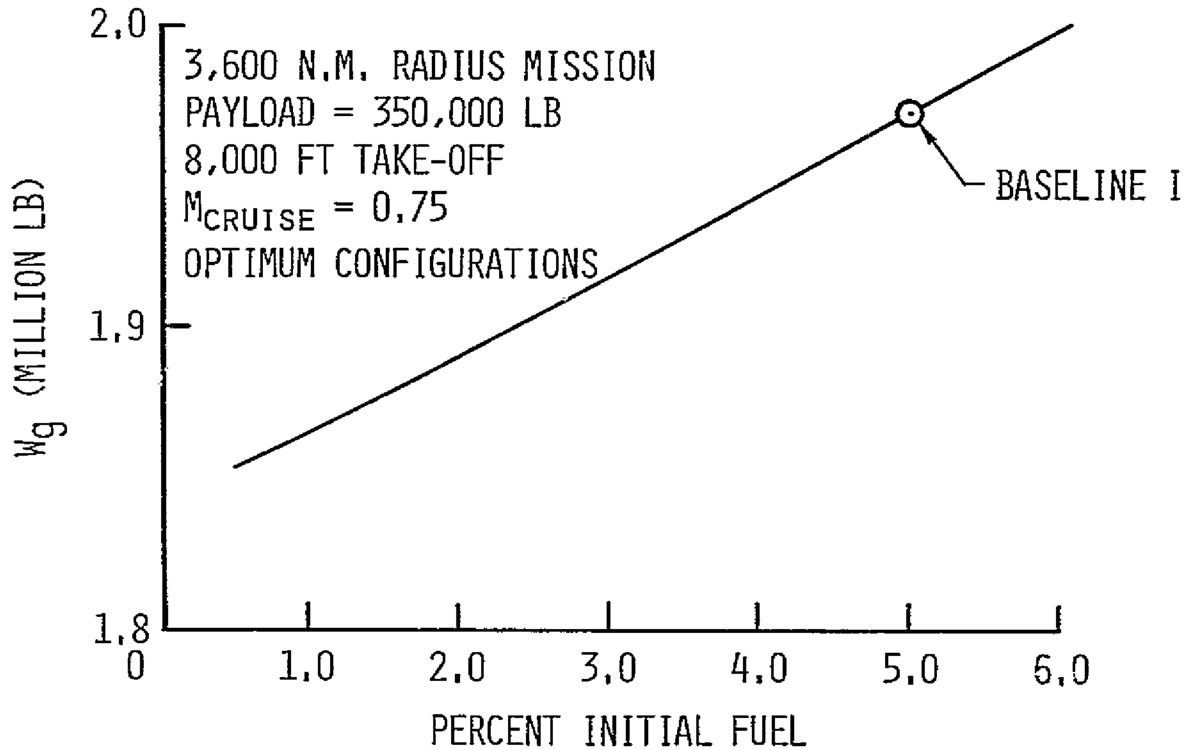


Figure 15.- Effect of reserve fuel requi..ements on configurations optimized for minimum gross weight.

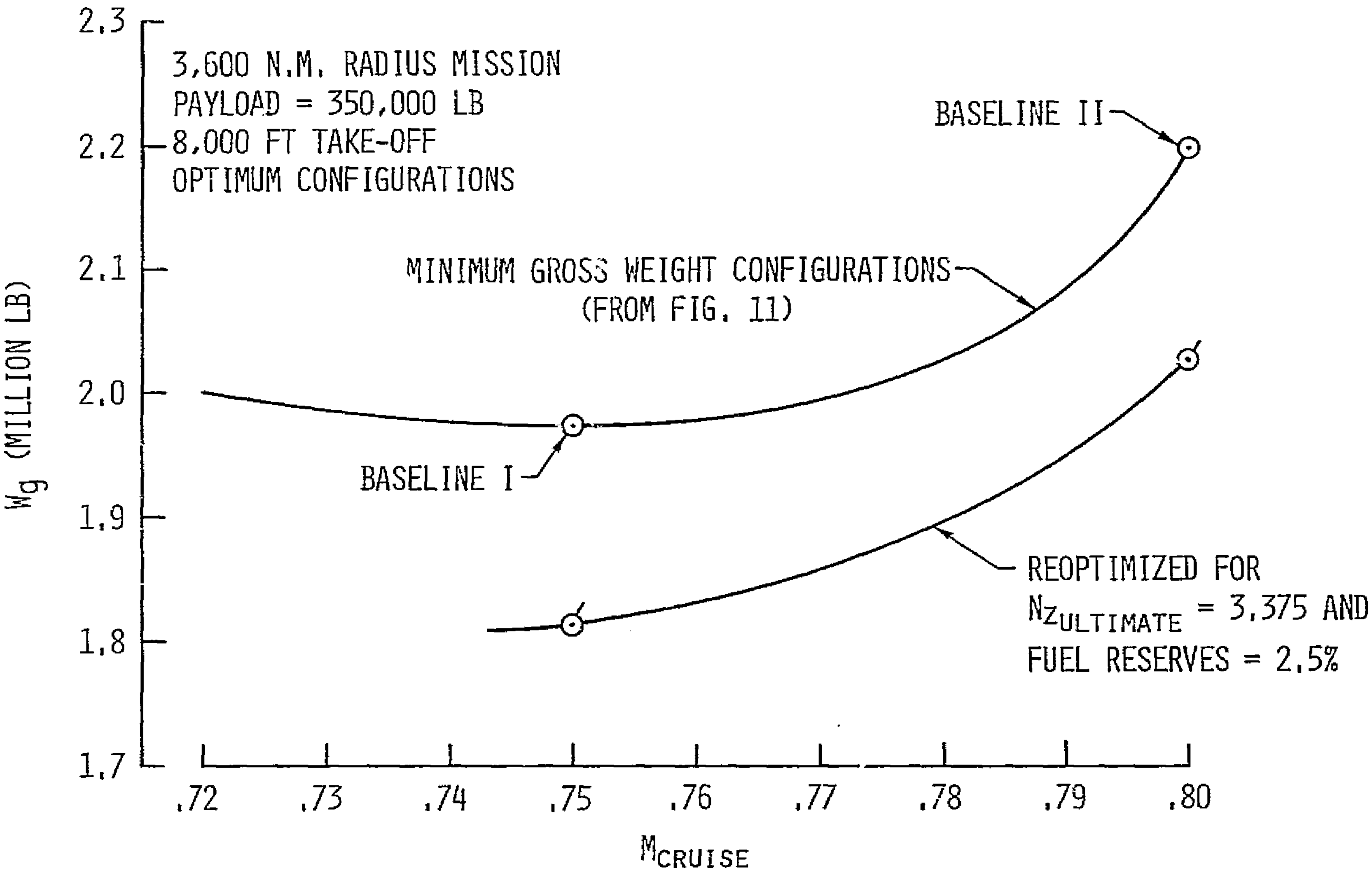


Figure 16.- Combined effect of load factor and reserve fuel requirements on configurations optimized for minimum gross weight.

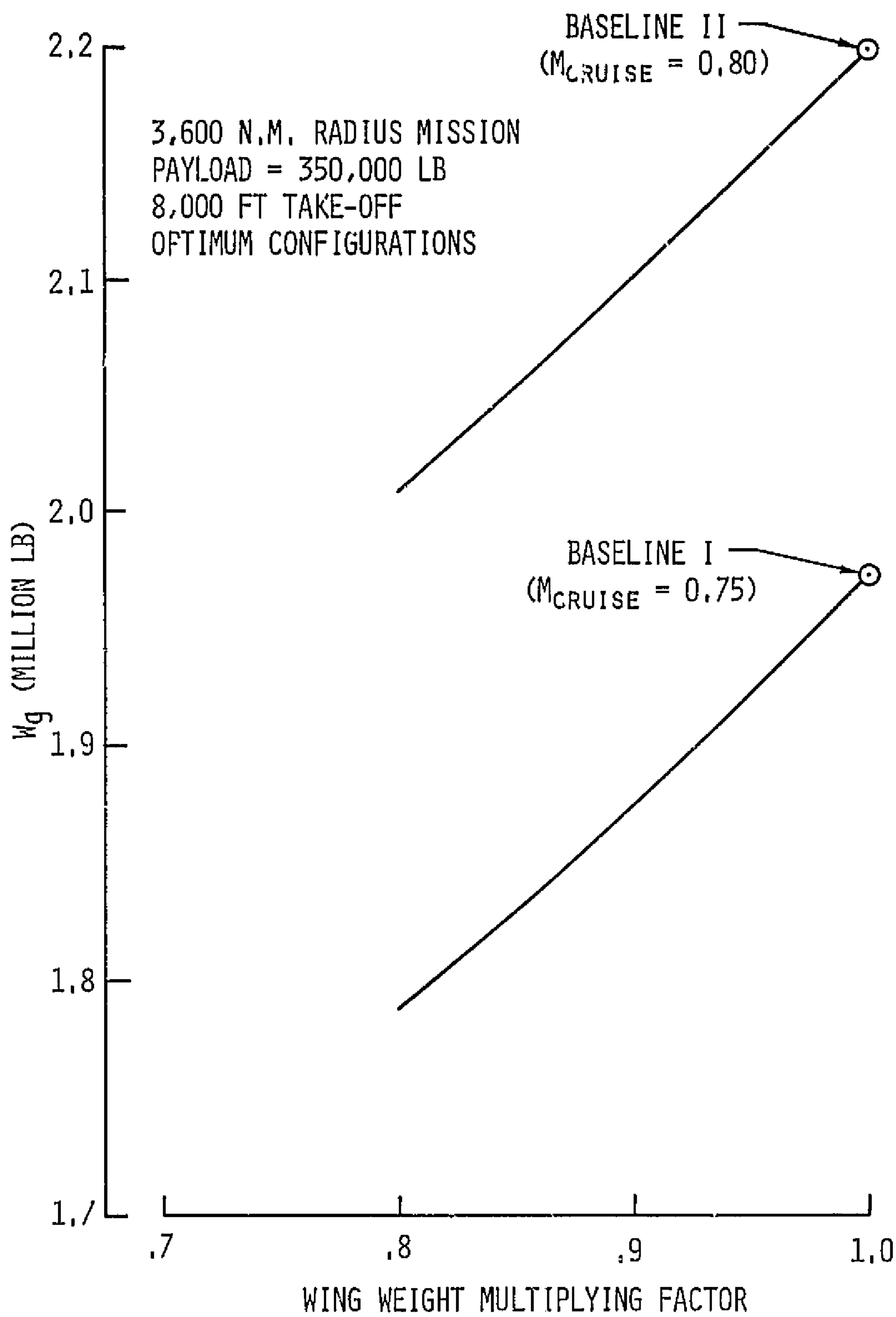


Figure 17.- Effect of wing weight on configurations optimized for minimum gross weight.

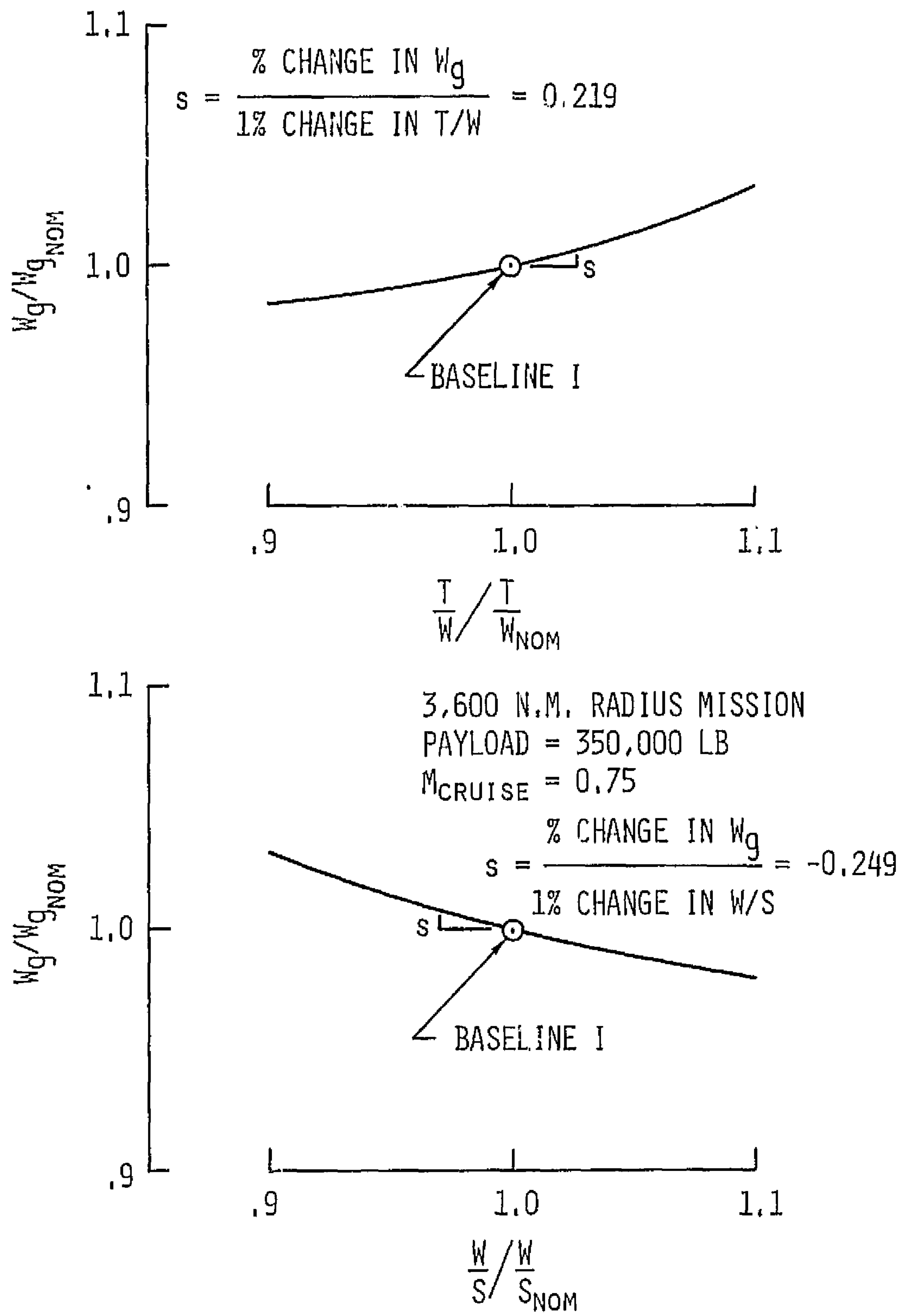


Figure 18.- Examples of gross weight sensitivities; nonoptimum.

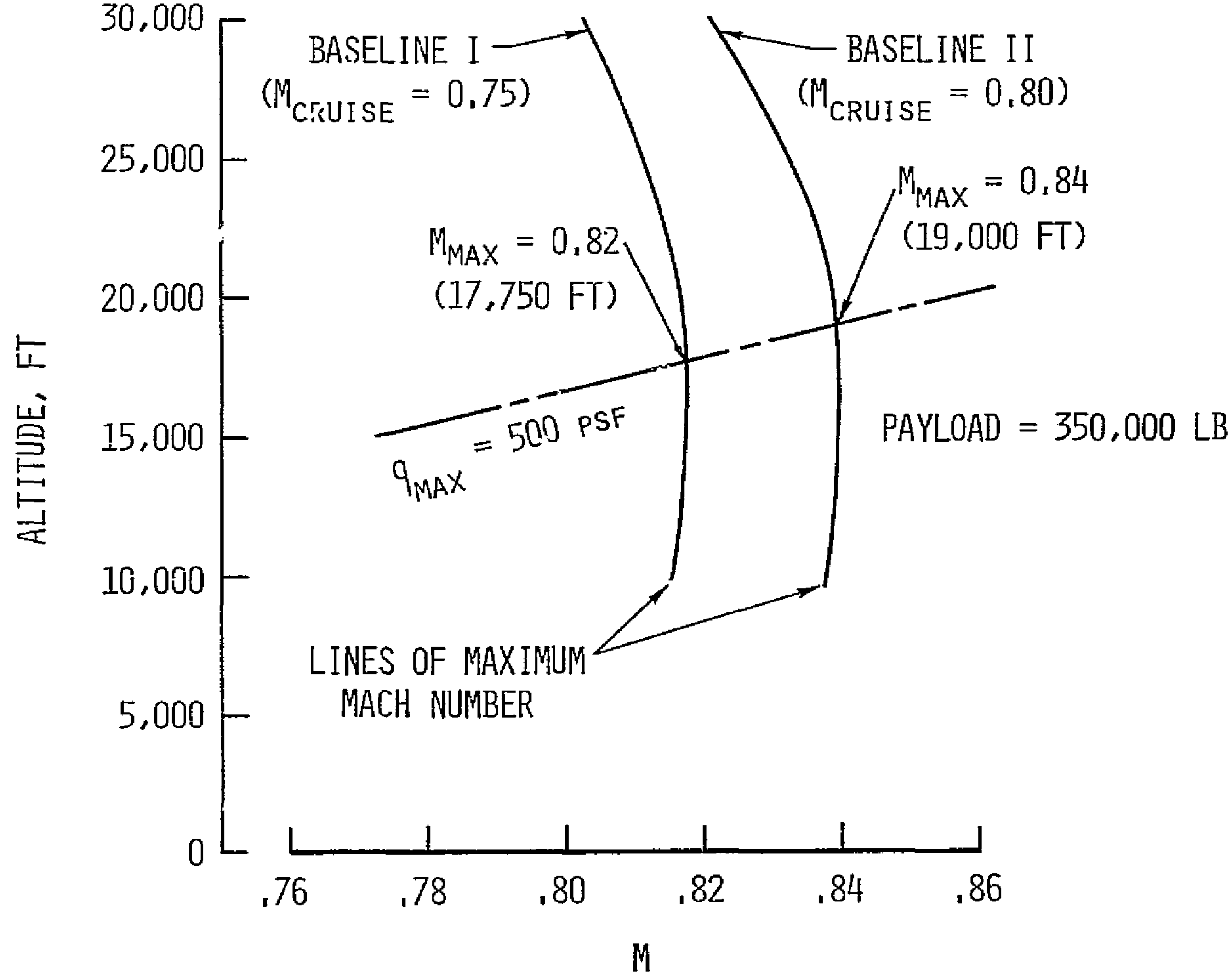


Figure 19.- Maximum Mach number capability.

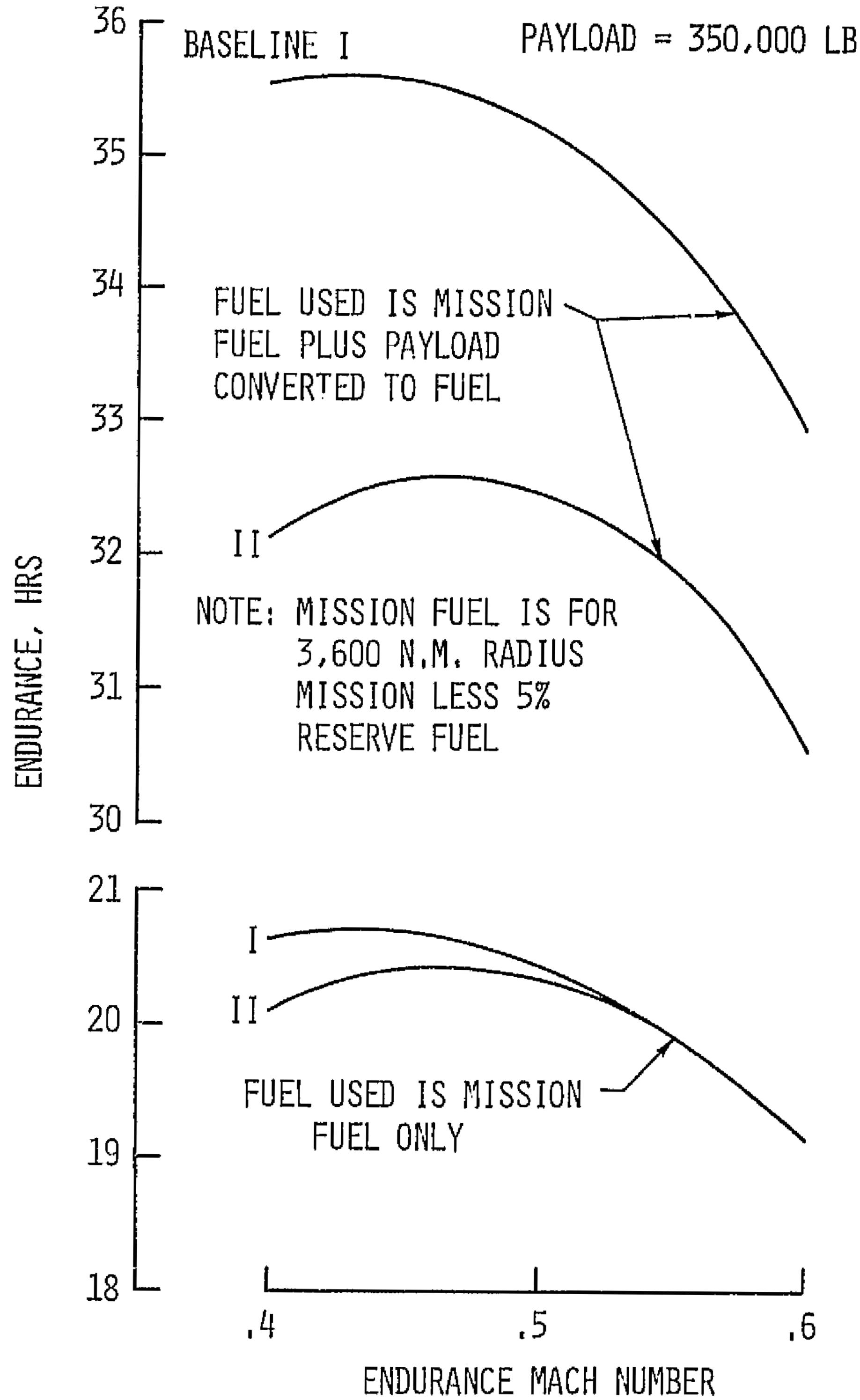


Figure 20.- Maximum endurance capability.

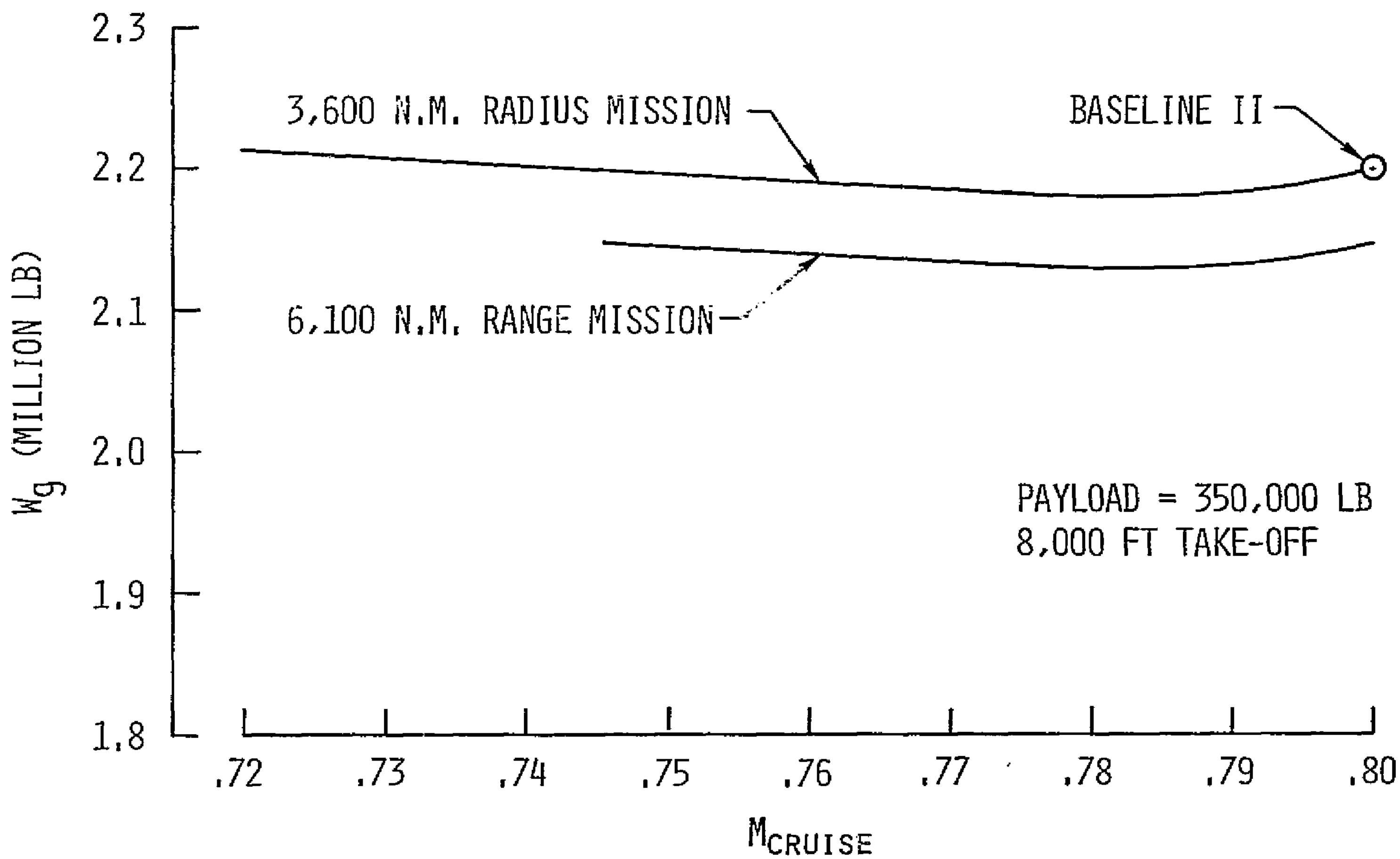
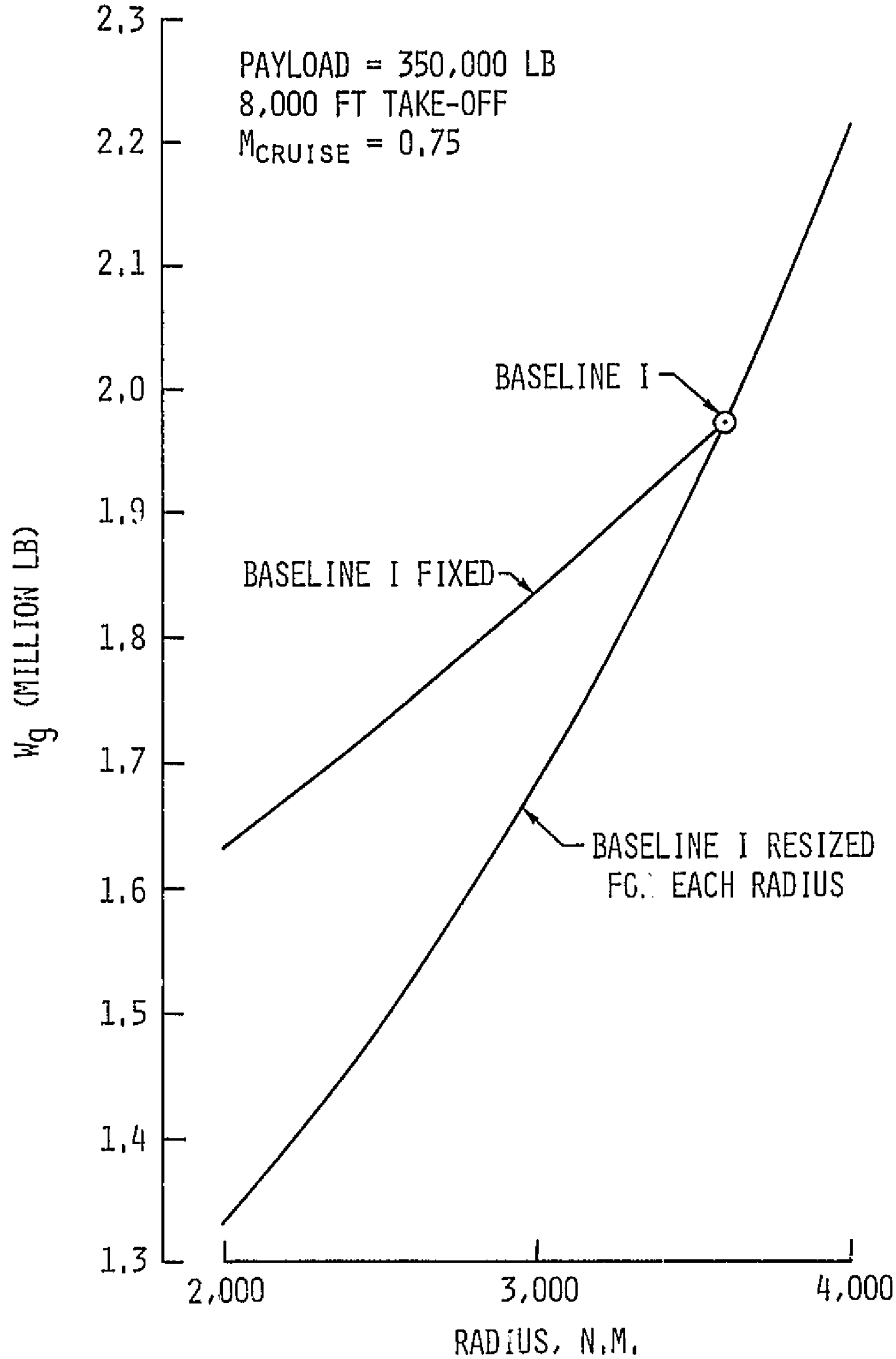
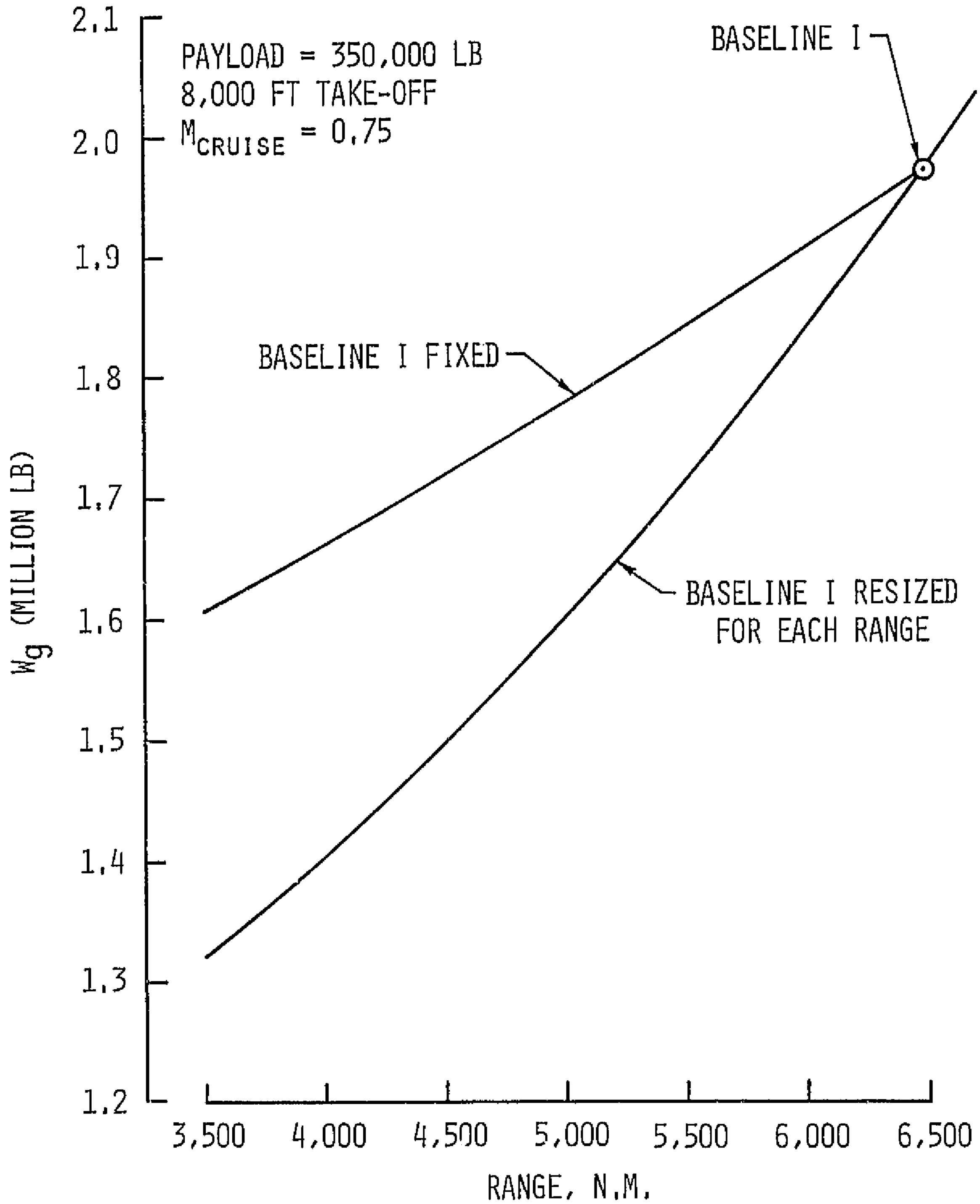


Figure 21.- Baseline II configuration at off-design conditions.



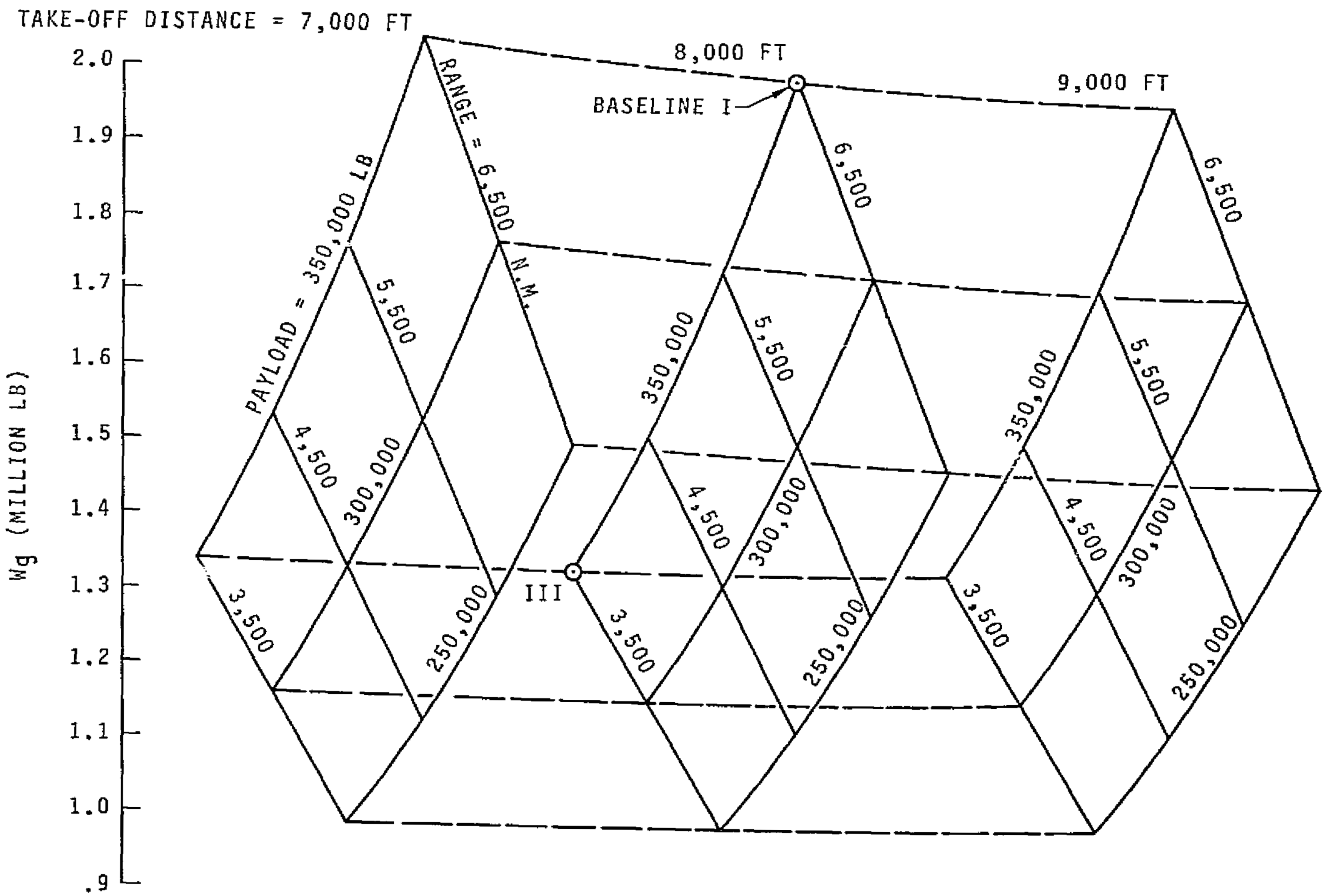
(a) Radius mission.

Figure 22.- Baseline I configuration at off-design conditions.



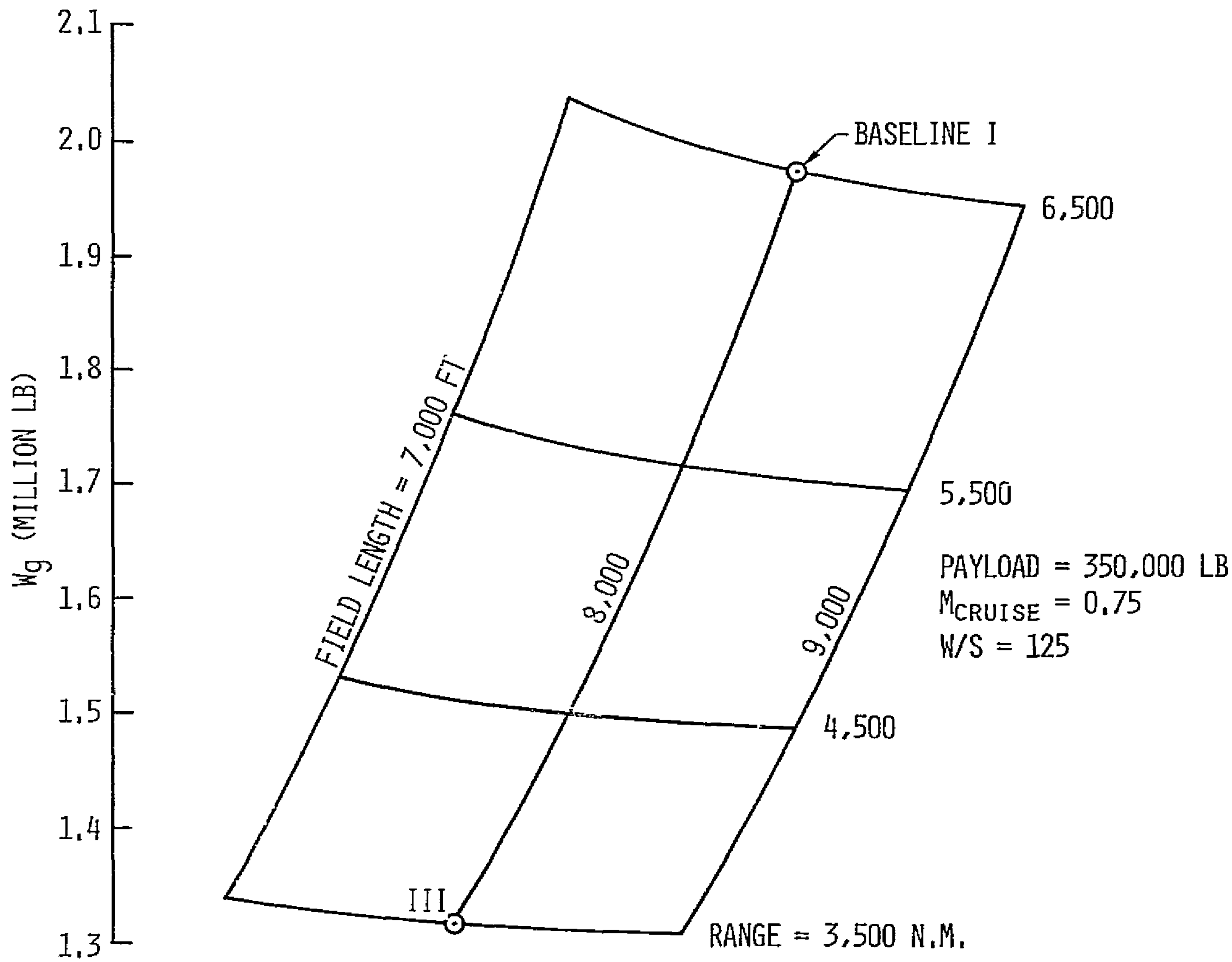
(b) Range mission.

Figure 22.- Concluded.



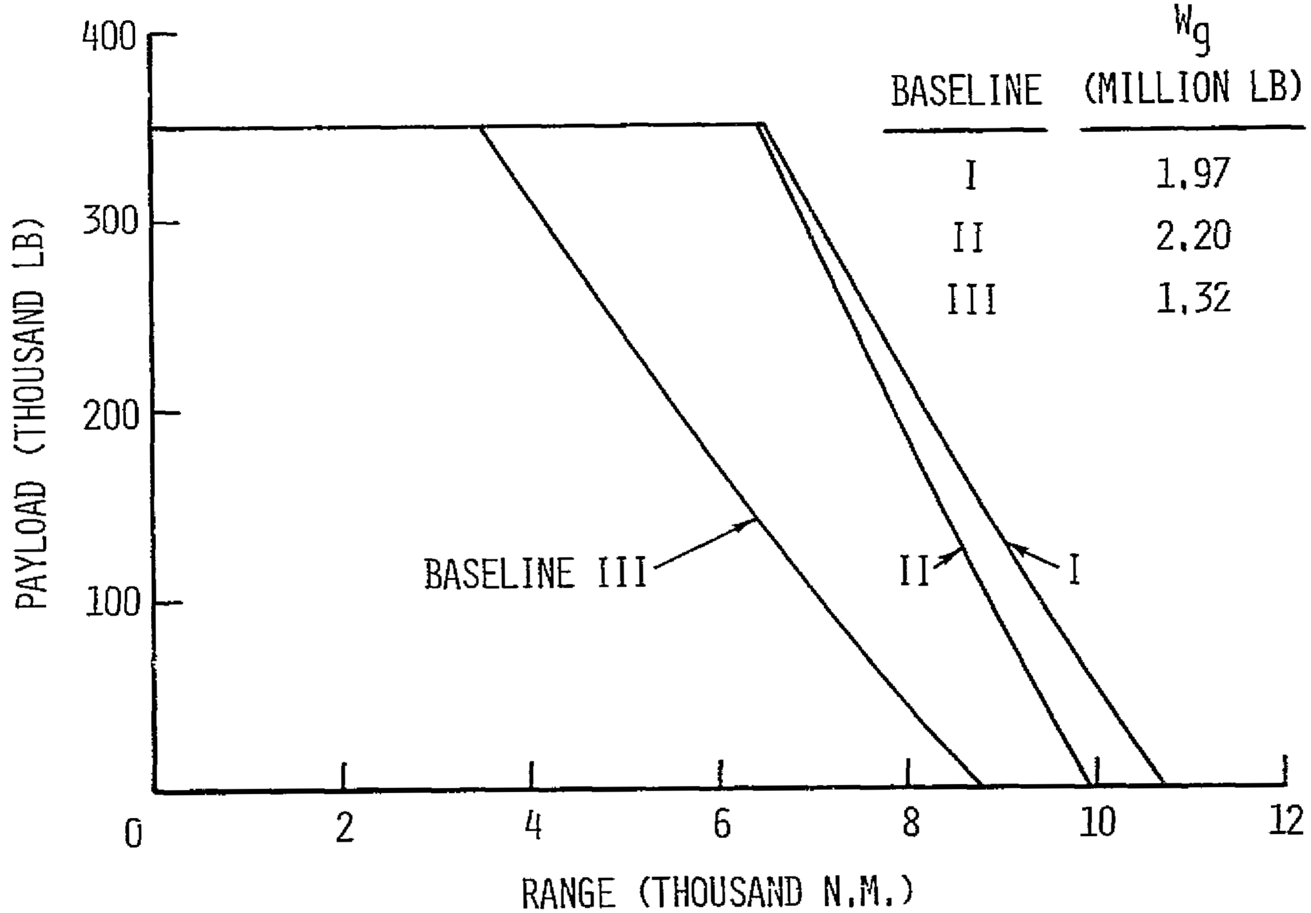
(a) Carpet plot.

Figure 23.- Baseline I configuration resized for various payloads and r ..ge missions; M_{cruise} = 0.75, W/S = 125.



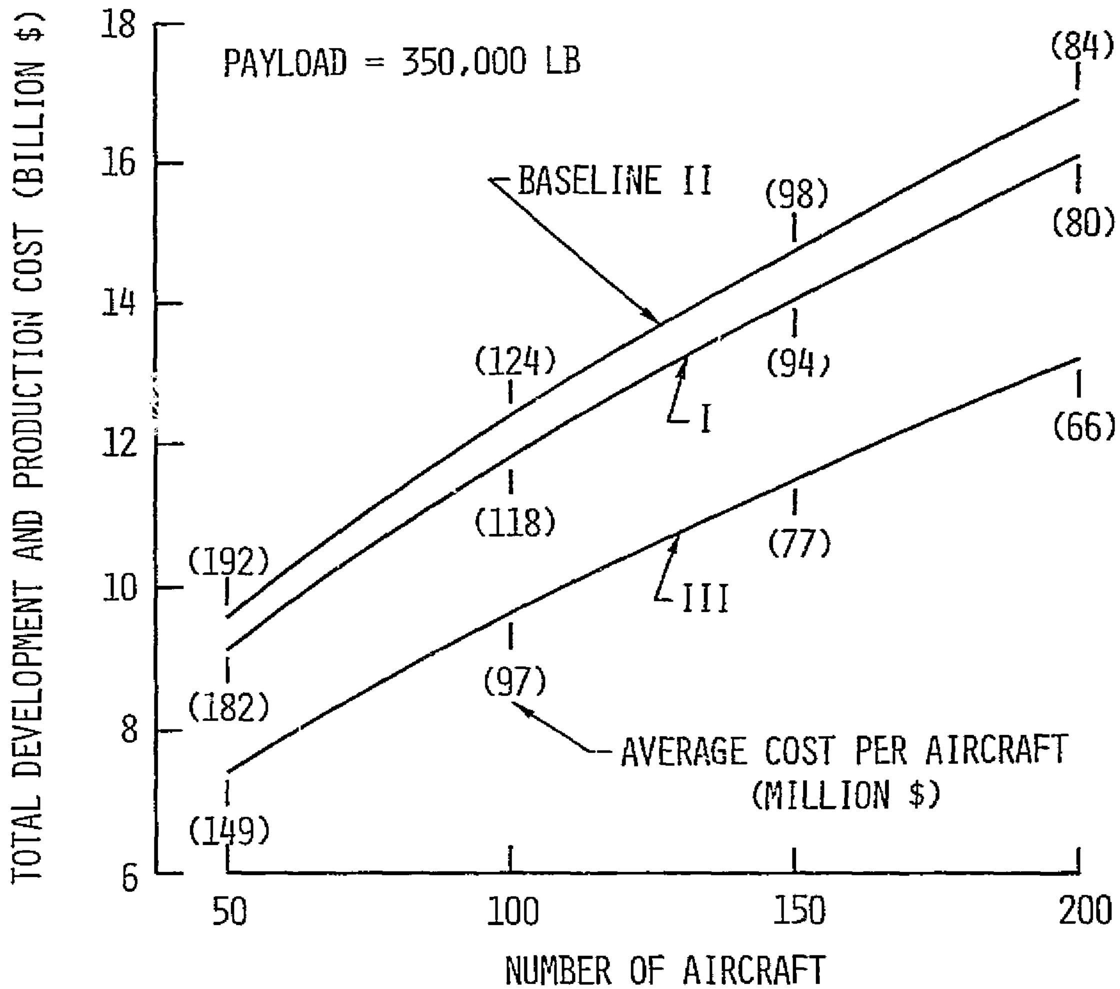
(b) Effect of takeoff field length.

Figure 23.- Concluded.



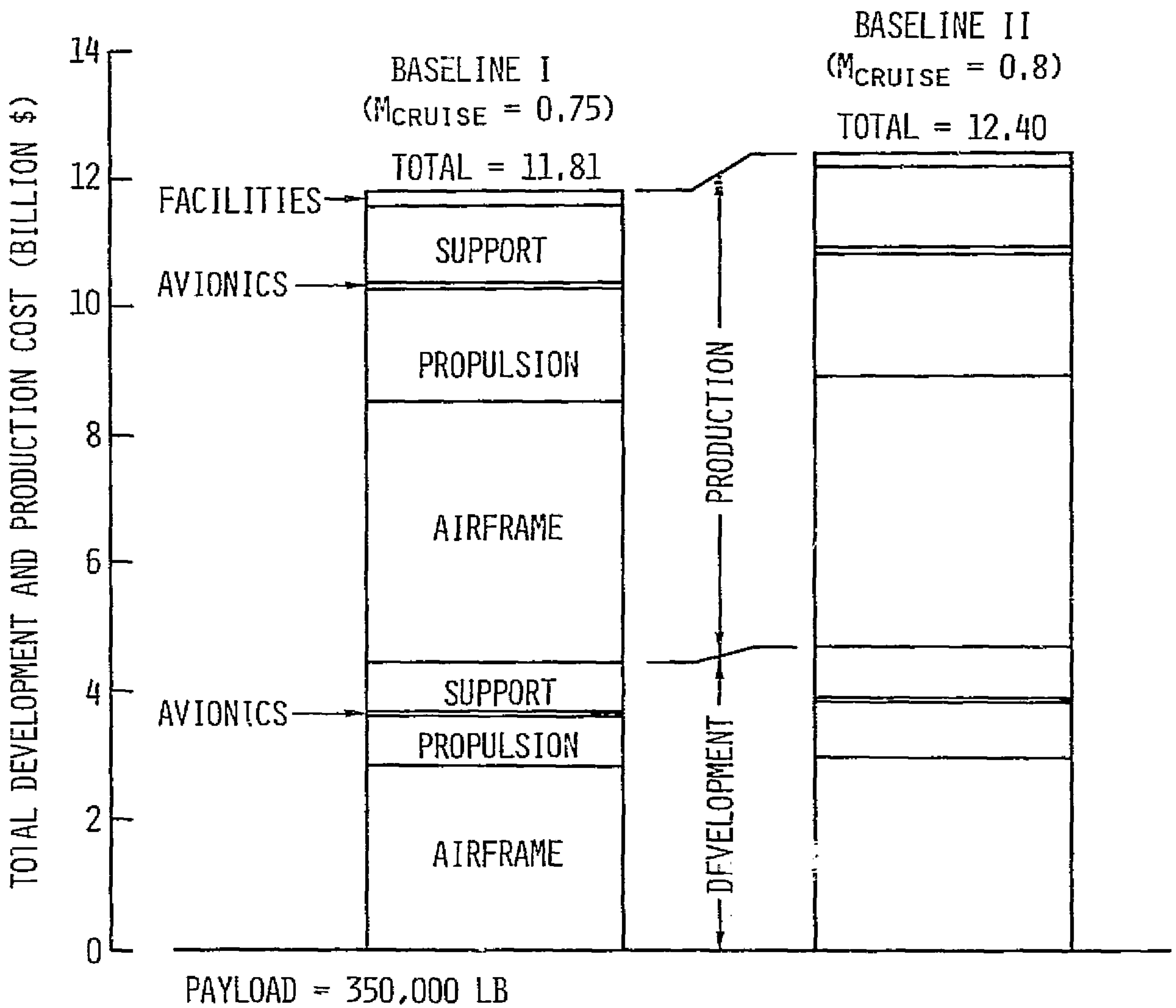
BASELINE	W_g (MILLION LB)	M_{CRUISE}
I	1.97	0.75
II	2.20	0.80
III	1.32	0.75

Figure 24.- Range-payload tradeoff; 8000-ft takeoff field length.



(a) Effect of number of aircraft.

Figure 25.- Development and production costs.



(b) Cost breakdown for 100 aircraft.

Figure 25.- Concluded.

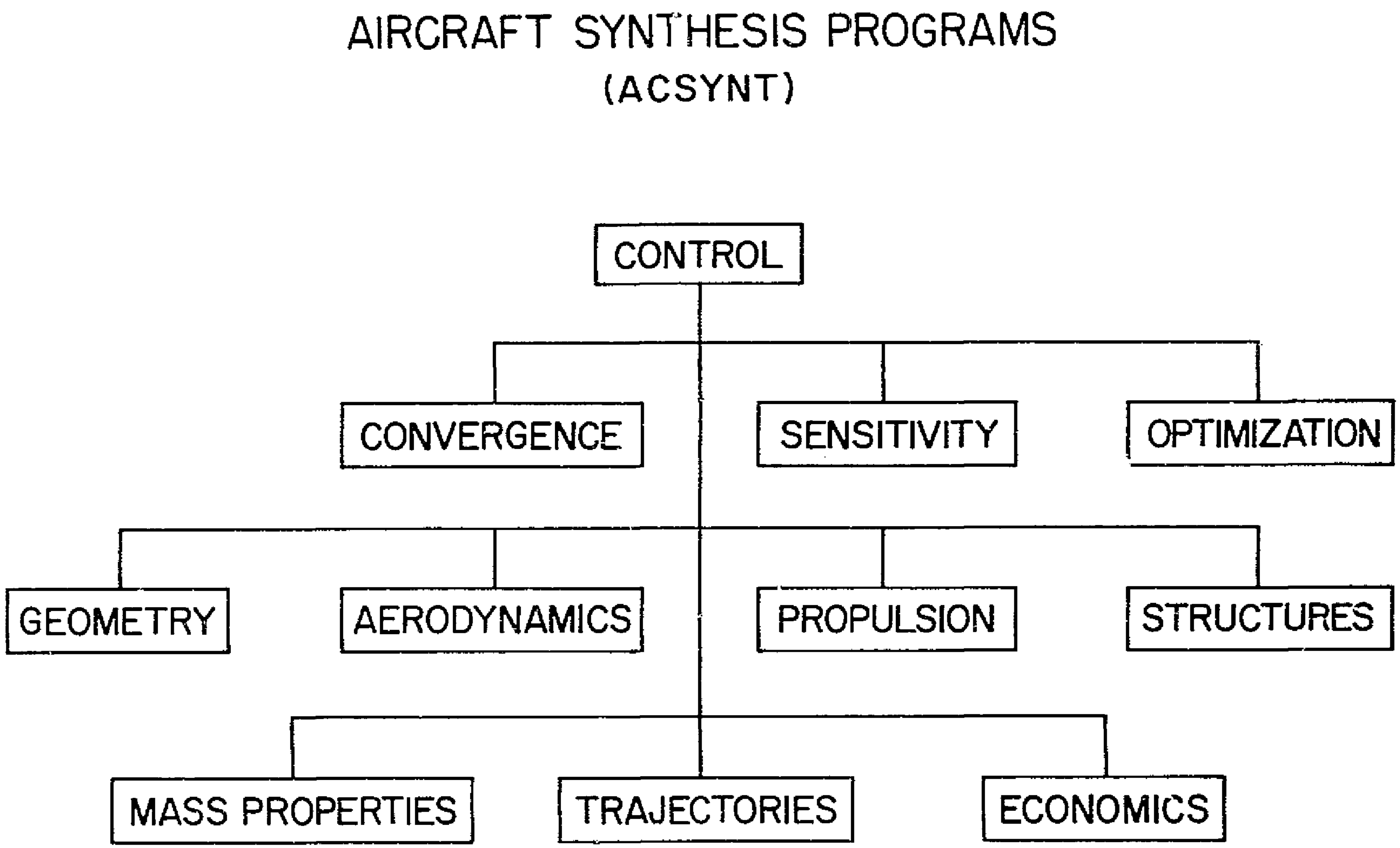


Figure 26.- Aircraft synthesis program disciplines.

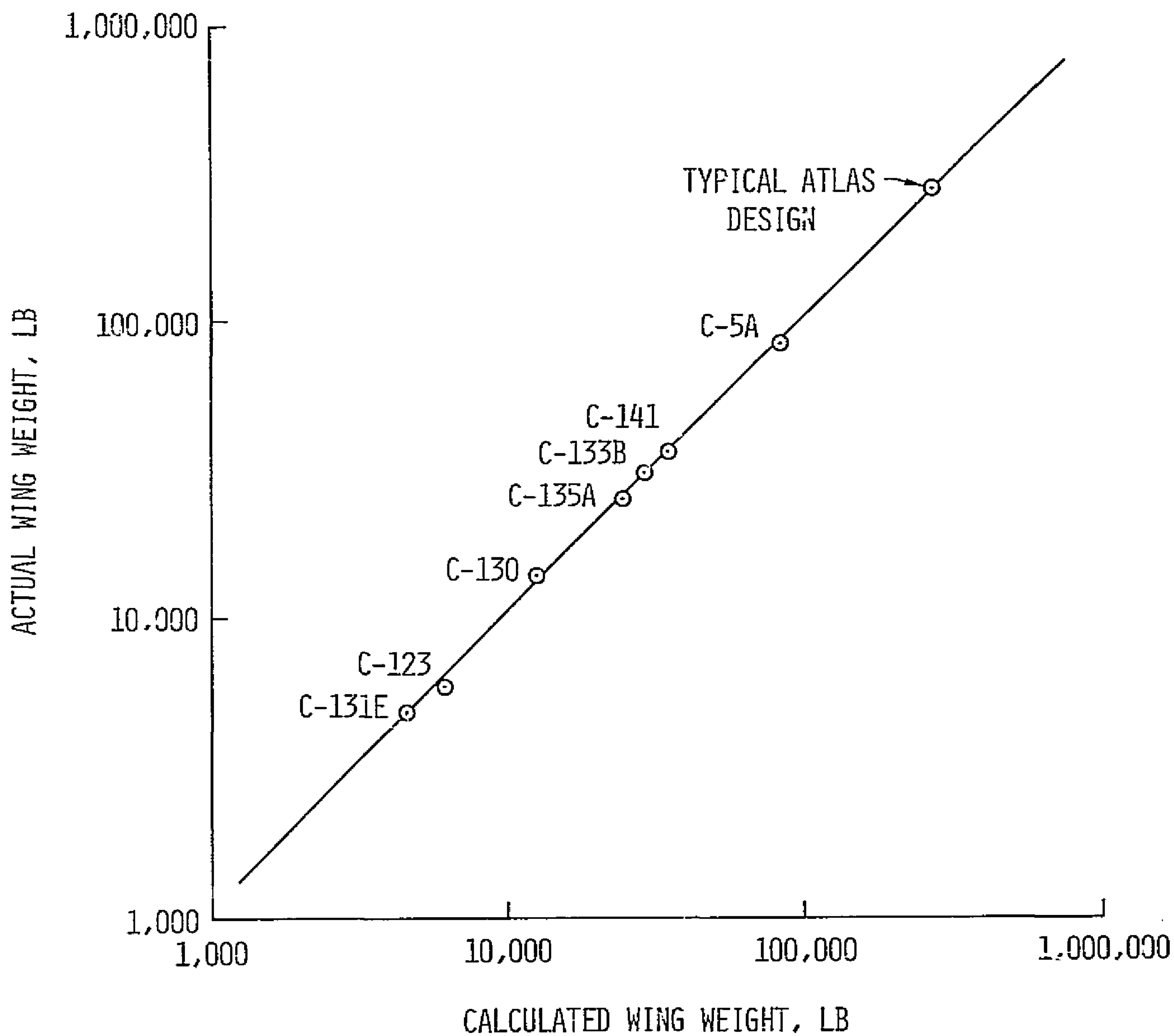


Figure 27.- Correlation of wing weight equation.

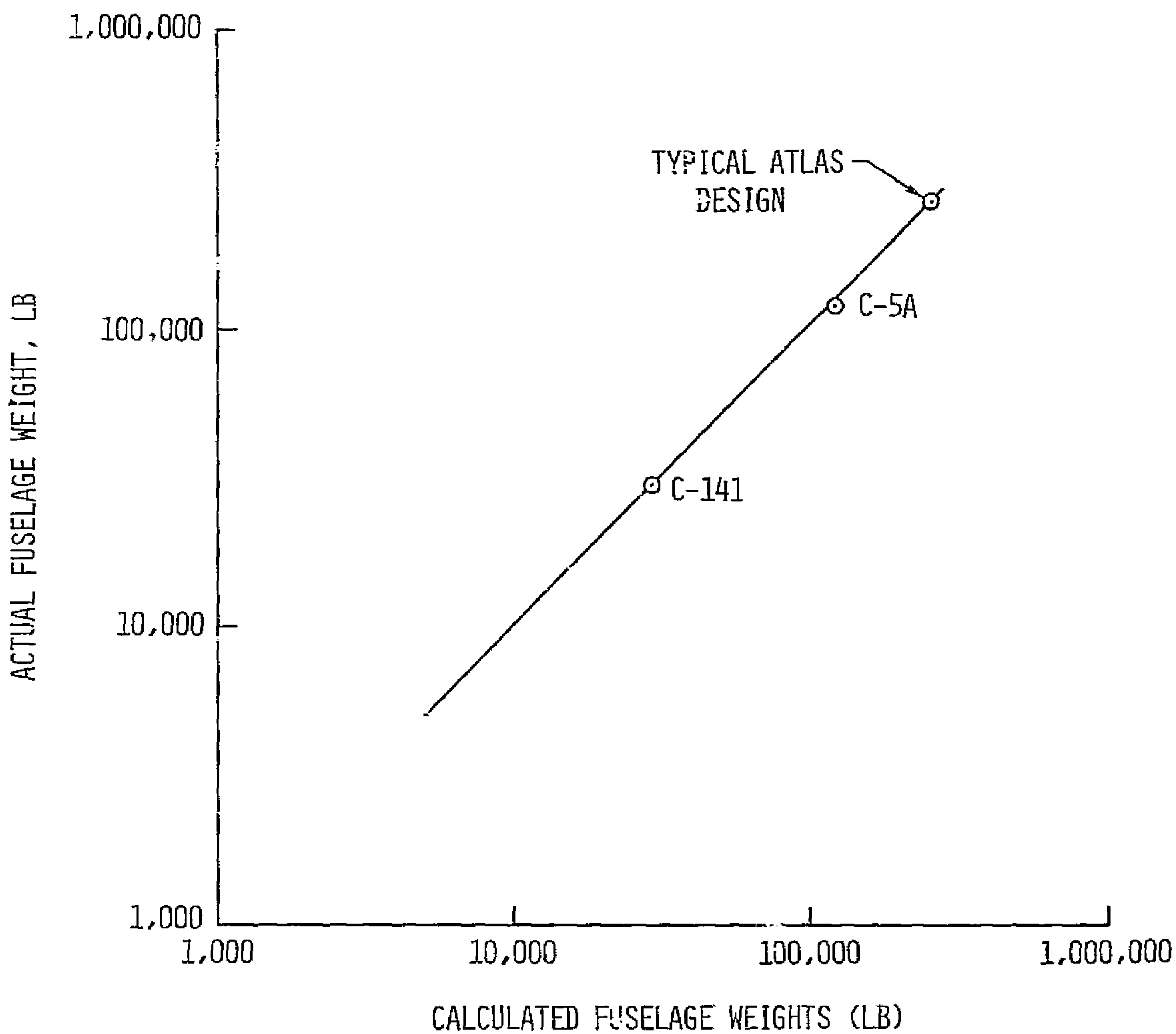


Figure 28.- Correlation of fuselage weight equation.

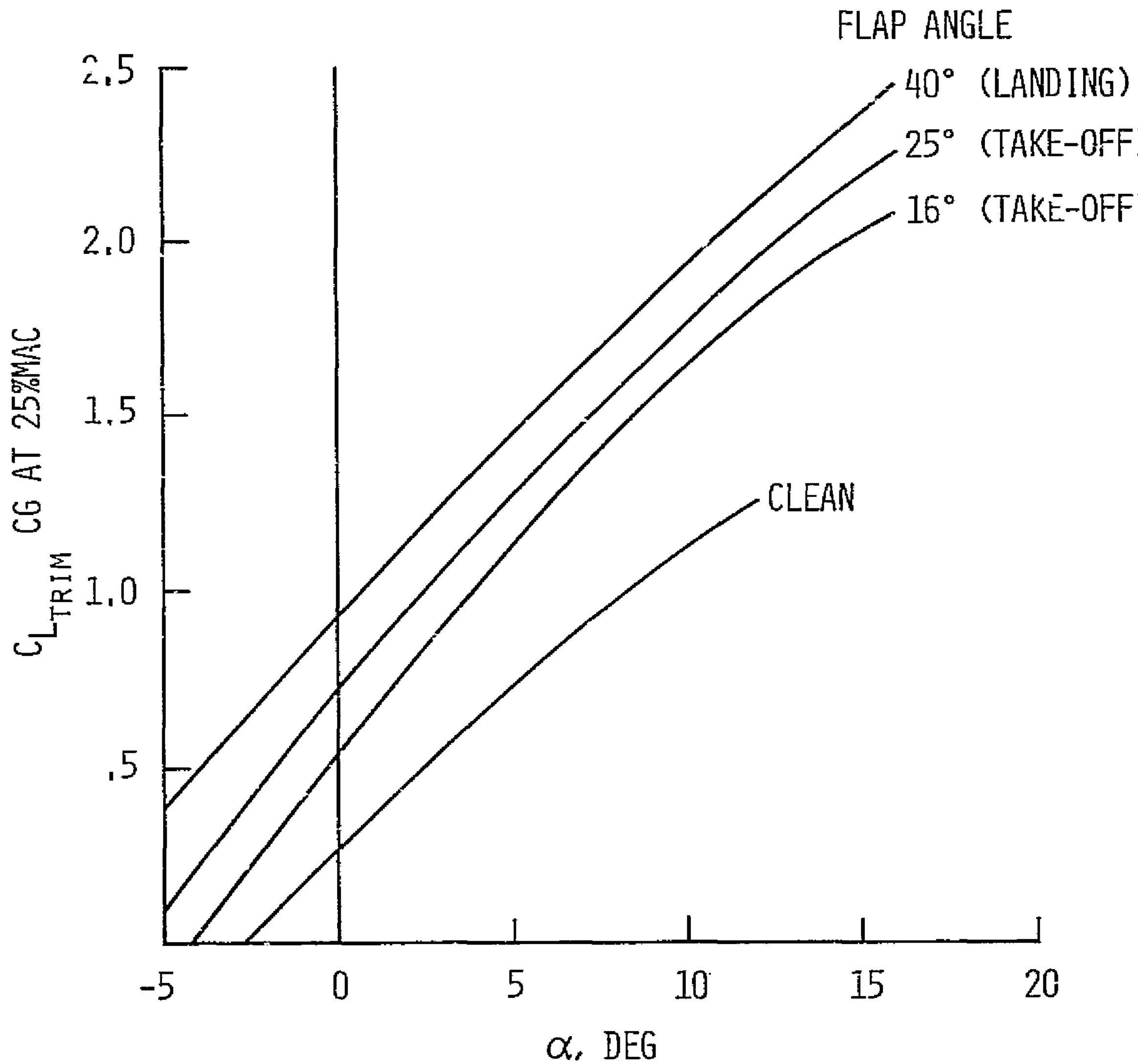


Figure 29.- Takeoff and landing lift characteristics used in study.

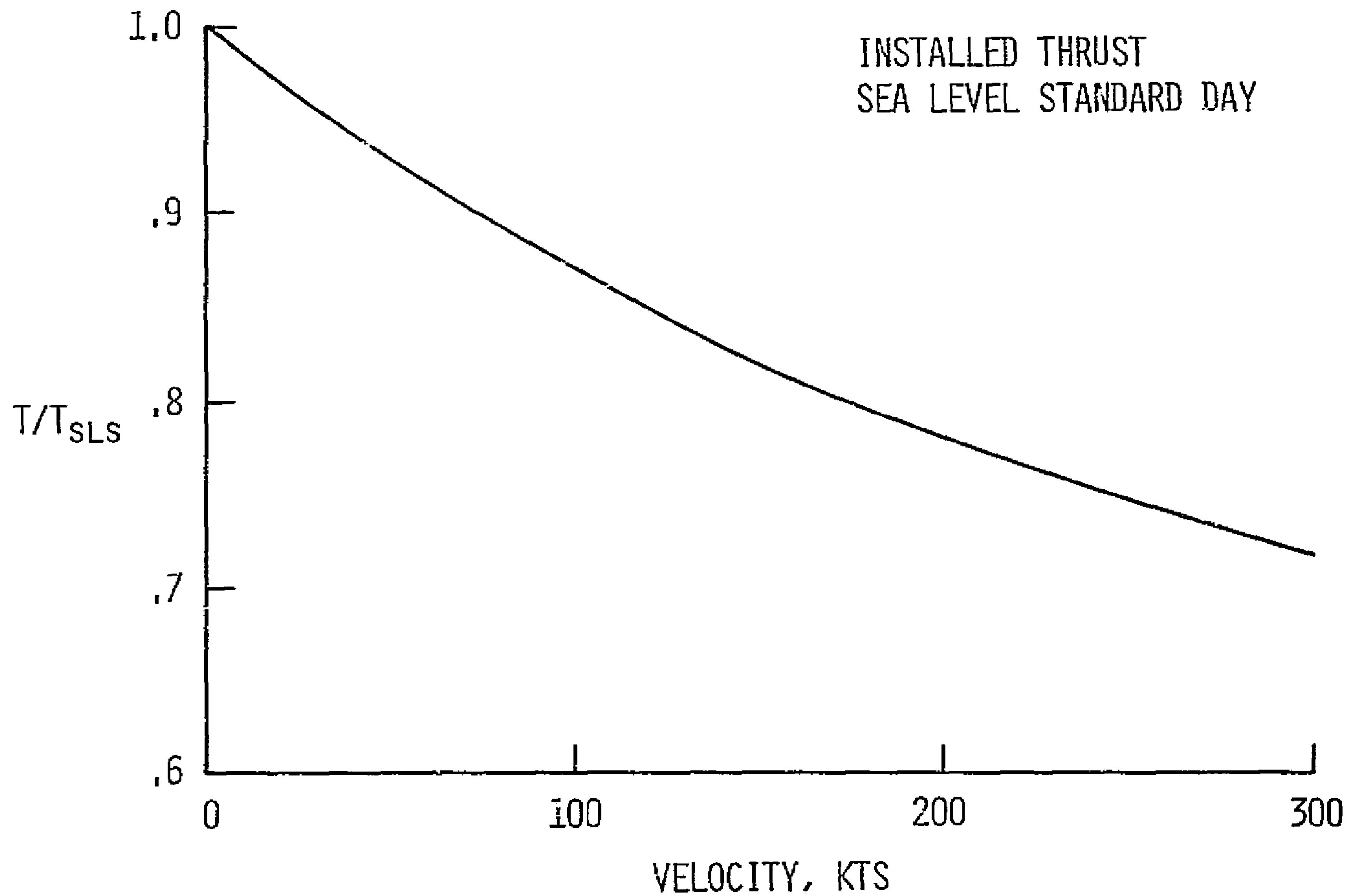


Figure 30.- Thrust lapse rate with takeoff velocity for study engine.

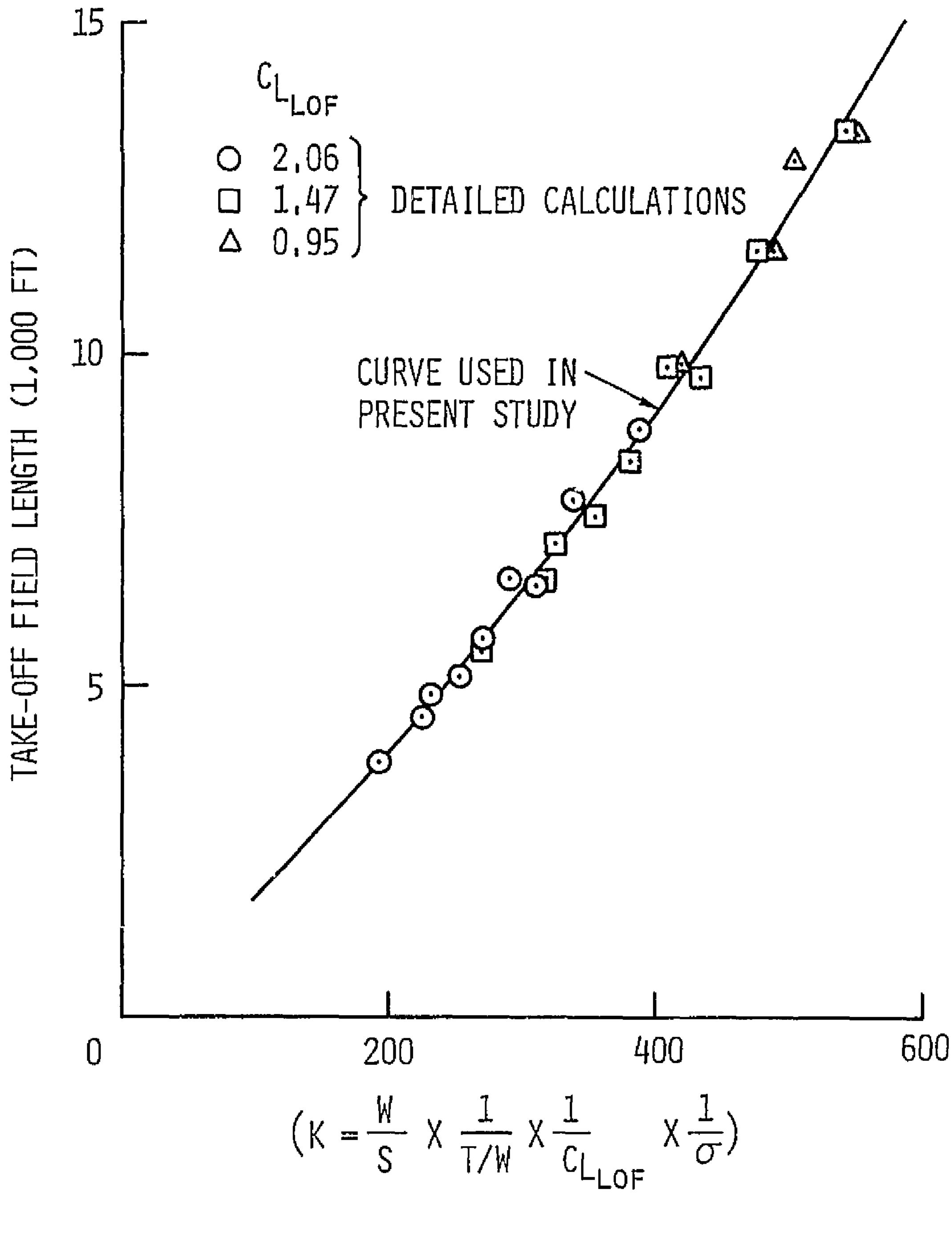
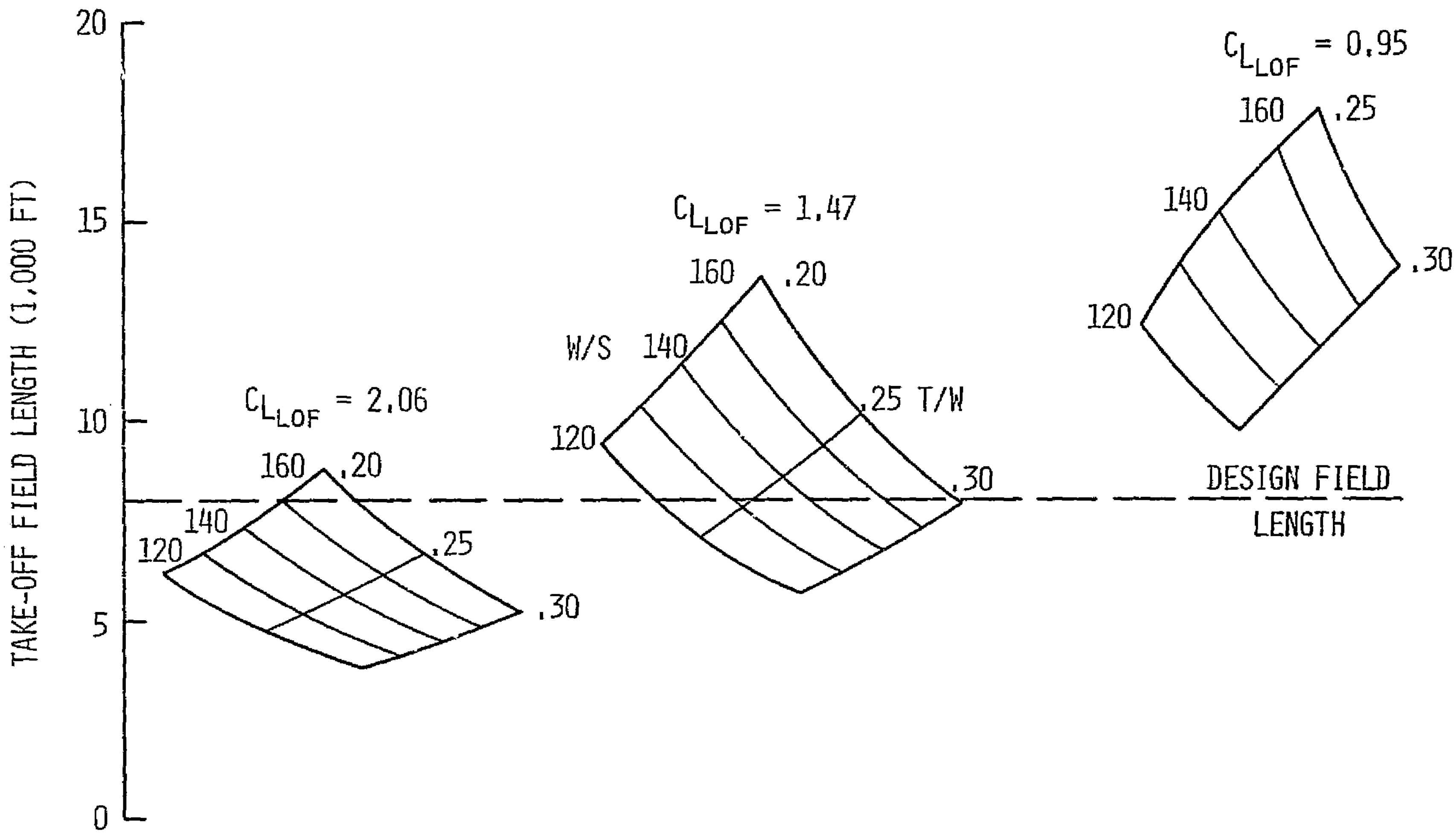
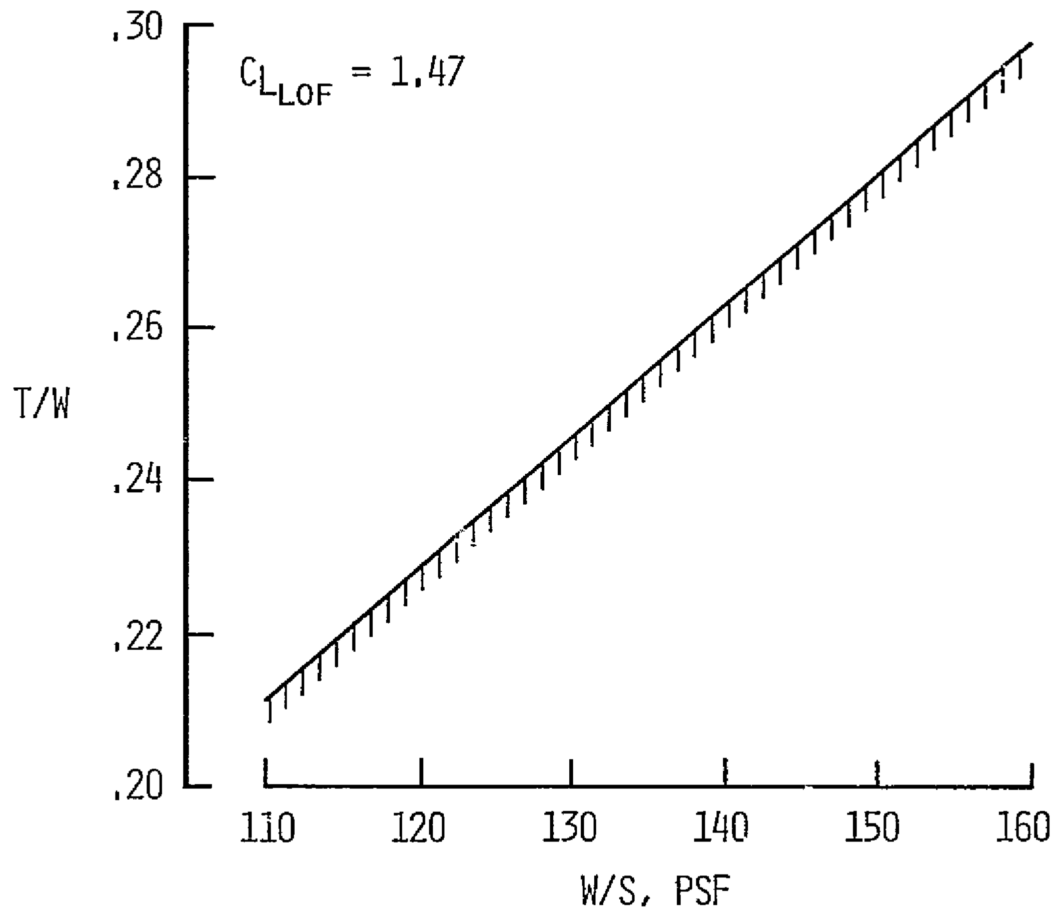


Figure 31.- Generalized form of takeoff field length used in study; all engine operation over a 50-ft obstacle.



(a) Carpet plot for various lift coefficients.

Figure 32.- Takeoff field length used in study; all engine operation over a 50-ft obstacle.



(b) Required thrust-to-weight ratio for 8000-ft takeoff distance.

Figure 32.- Concluded.

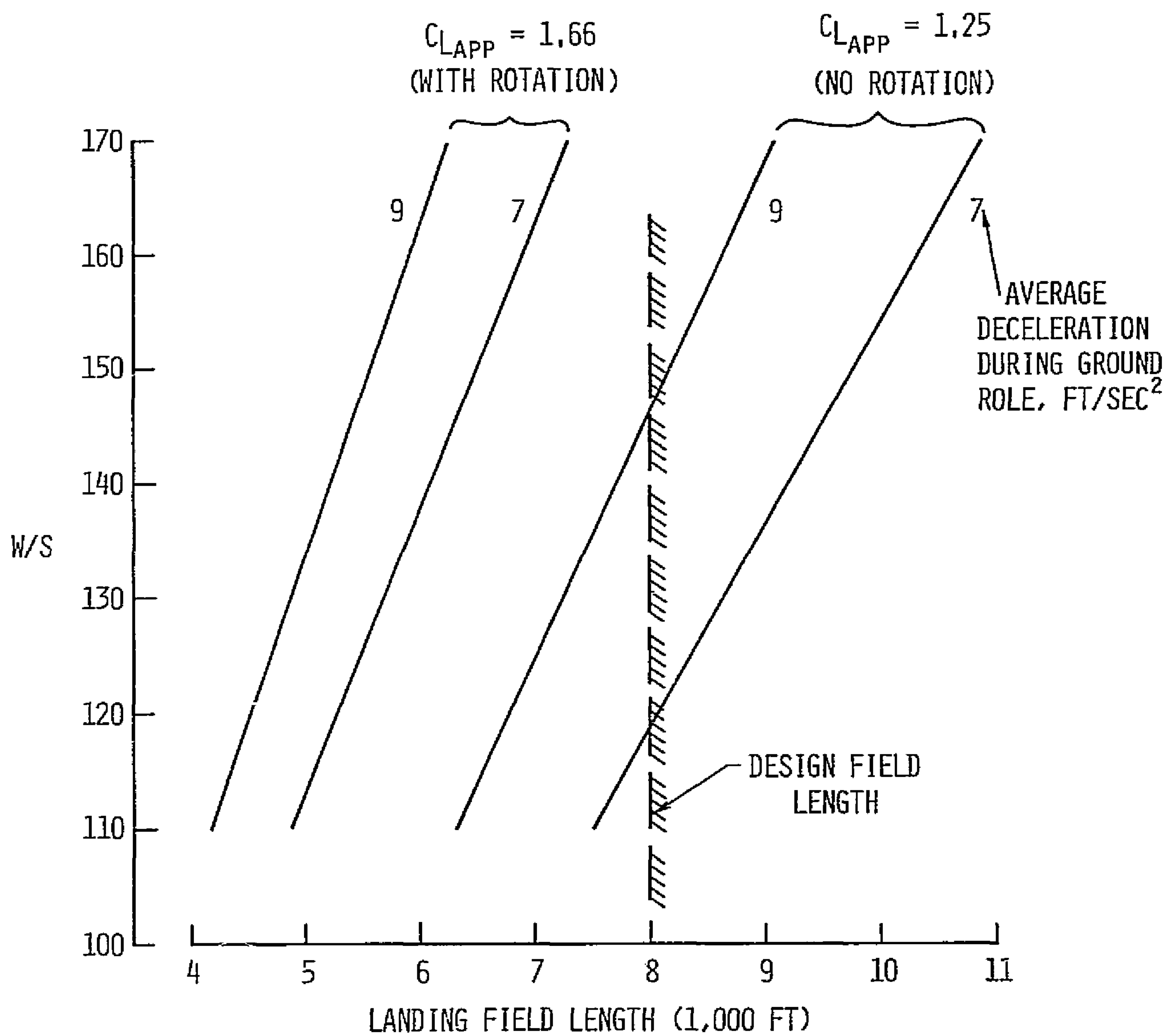


Figure 33.- Landing field length used in study.

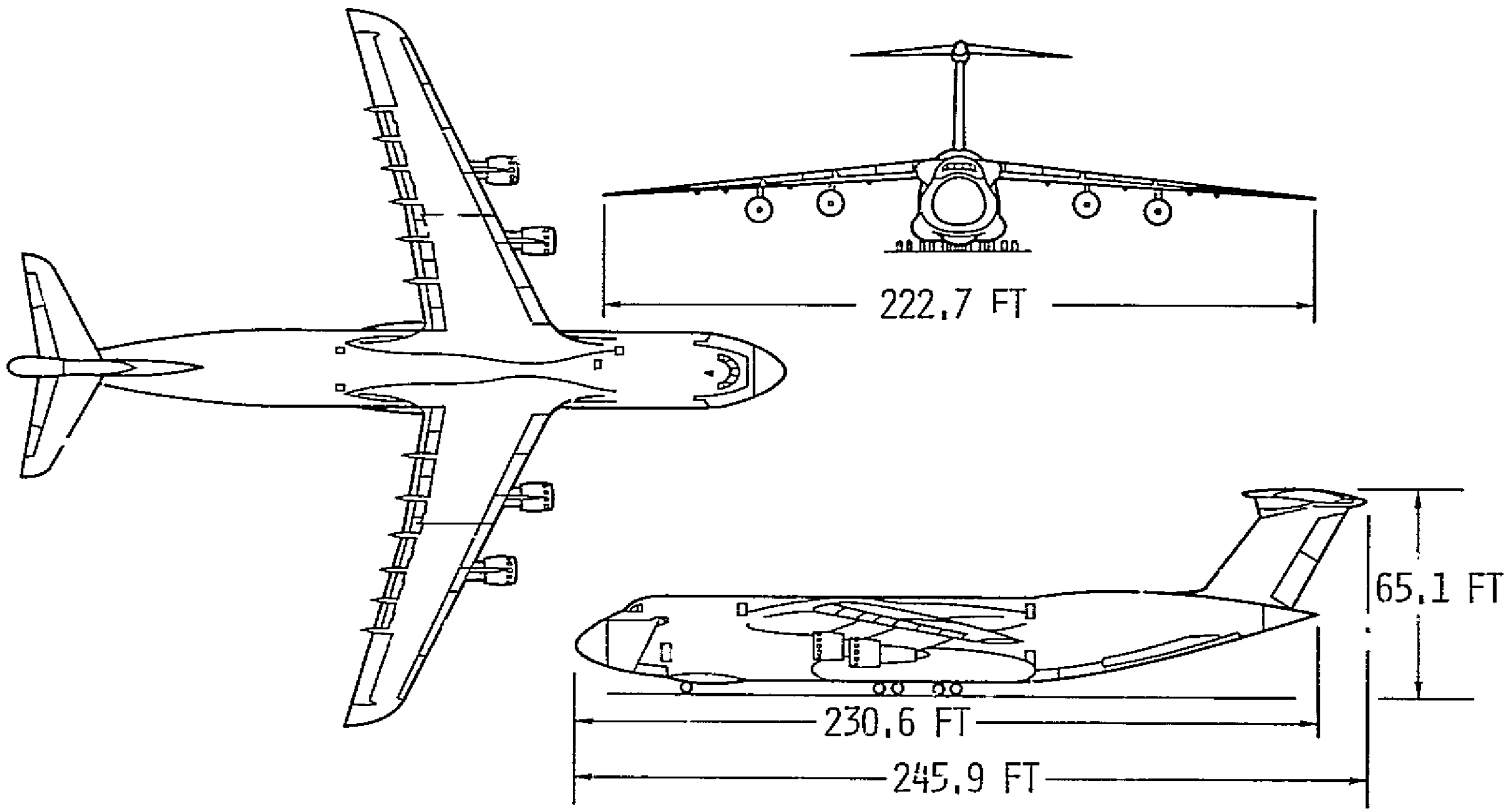
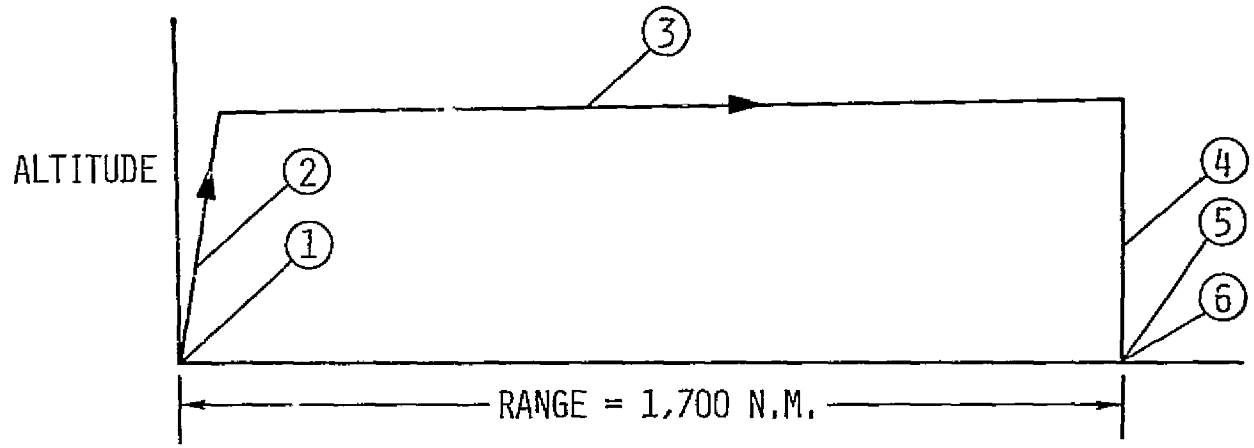


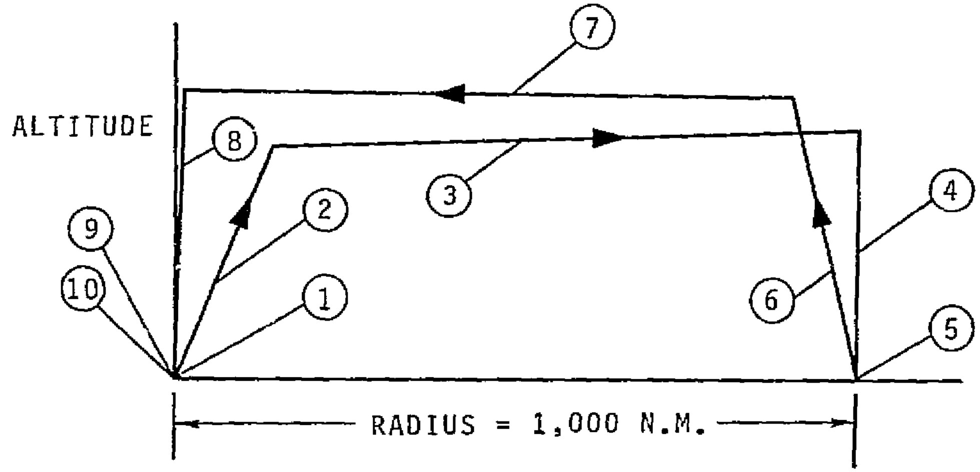
Figure 34.- C-5A general arrangement.



1. TAKE-OFF - 5 MIN. NORMAL RATED THRUST FOR GROUND OPERATIONS AND TAKE-OFF AT SEA LEVEL
2. CLIMB - CONSTANT INDICATED AIRSPEED CLIMB AT NORMAL THRUST TO LONG-RANGE CRUISE ALTITUDE
3. CRUISE - ACCELERATE TO AND CRUISE AT OPTIMUM CRUISE-CLIMB ALTITUDE AT 440 KTS (M = 0.75) FOR 1,700 N.M. FROM TAKE-OFF POINT
4. DESCENT - DESCEND TO SEA LEVEL; NO FUEL COST; NO RANGE CREDIT
5. LOITER - 30 MIN. LOITER AT VELOCITY FOR MAX. ENDURANCE AT SEA LEVEL; NO RANGE CREDIT
6. LAND - LAND WITH 5% INITIAL FUEL RESERVE; NO FUEL COST; NO RANGE CREDIT

POINT DESIGN PARAMETER - AIRCRAFT WILL ACHIEVE MACH = 0.75 AT 32,000 FT ALTITUDE AT NORMAL THRUST AT END OF CLIMB 2

Figure 35.- C-5A basic range mission; payload = 265,000 lb.



(1) TAKE-OFF - 10 MIN. NORMAL RATED THRUST FOR GROUND OPERATIONS AND TAKE-OFF AT SEA LEVEL

(2) CLIMB - CONSTANT INDICATED AIRSPEED CLIMB AT NORMAL THRUST TO LONG-RANGE CRUISE ALTITUDE

(3) CRUISE - ACCELERATE TO AND CRUISE AT OPTIMUM CRUISE-CLIMB ALTITUDE AT 440 KTS (MACH = 0.75) FOR 1,000 N.M. FROM TAKE-OFF POINT

(4) DESCENT - DESCENT TO SEA LEVEL AND LAND; NO RANGE CREDIT; NOT FUEL COST. UNLOAD PAYLOAD

(5) TAKE-OFF - WITHOUT REFUELING; NO PAYLOAD

(6) CLIMB - AS IN (2) ON RETURN LEG

(7) CRUISE - RETURN CRUISE AS IN (3)

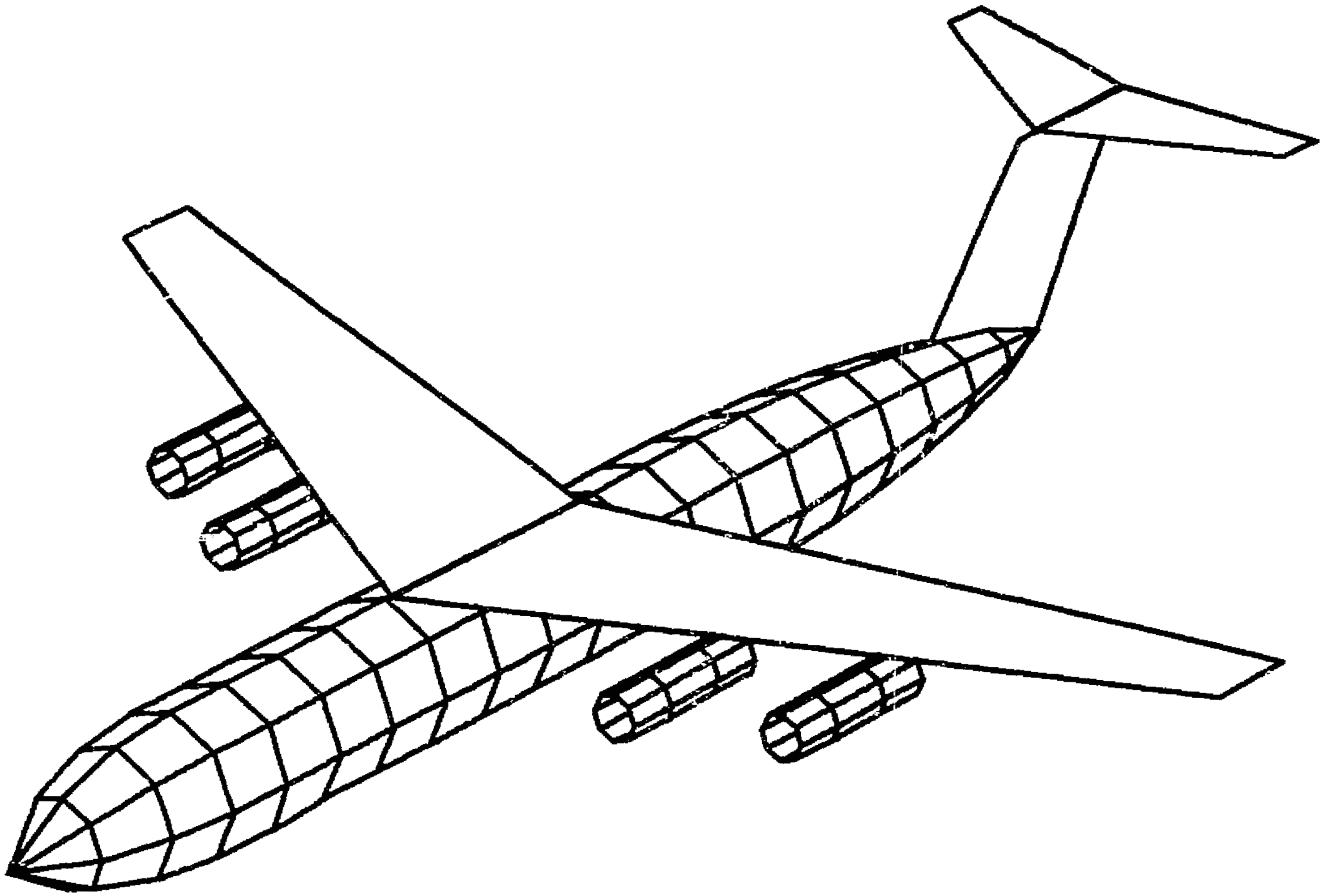
(8) DESCENT - DESCEND TO SEA LEVEL; NO FUEL COST; NO RANGE CREDIT

(9) LOITER - 30 MIN. LOITER AT VELOCITY FOR MAX ENDURANCE AT SEA LEVEL; NO RANGE CREDIT

(10) LAND - LAND WITH 5% INITIAL FUEL RESERVE; NO FUEL COST; NO RANGE CREDIT

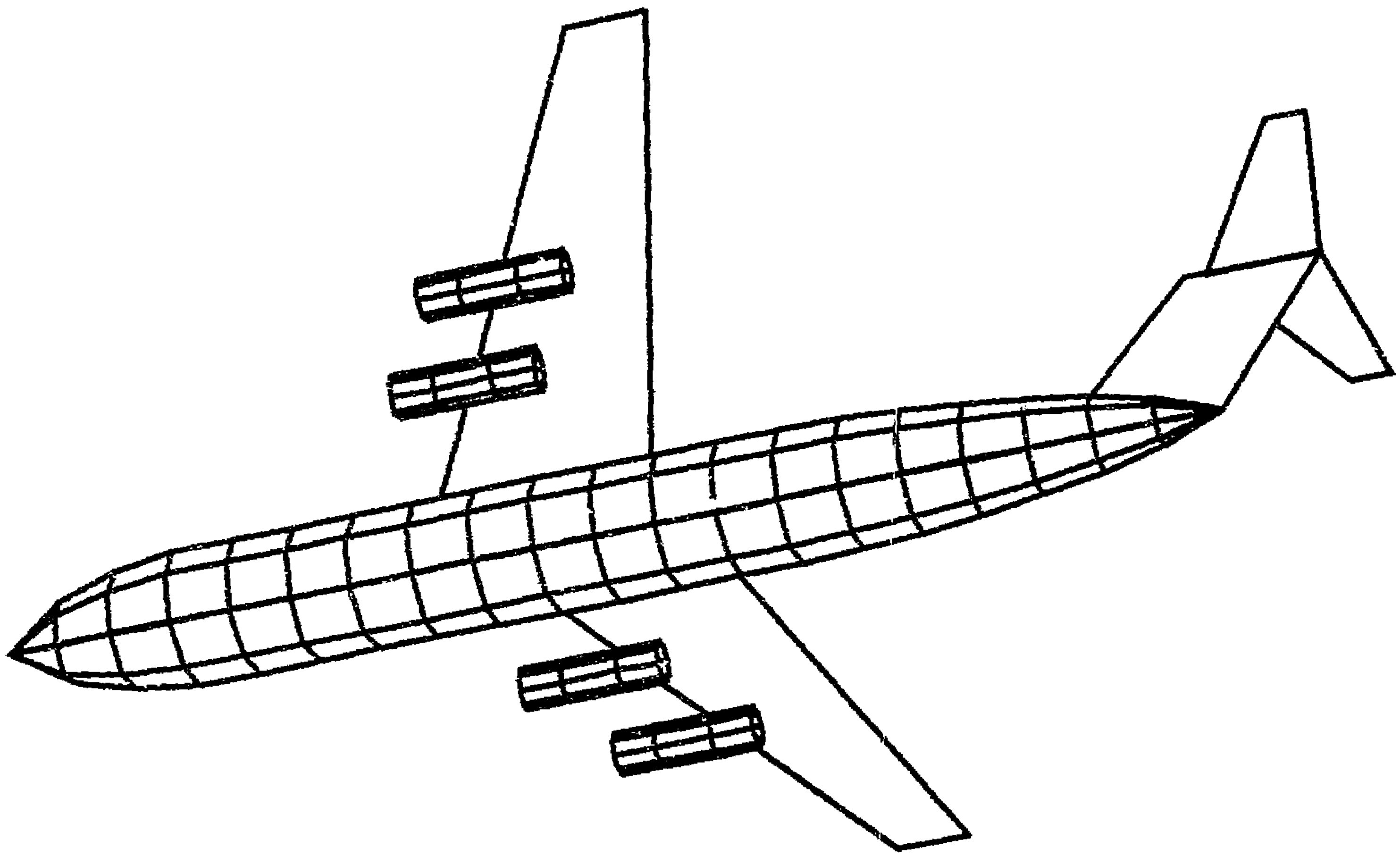
POINT DESIGN PARAMETER - AIRCRAFT WILL ACHIEVE MACH = 0.75 AT 32,000 FT ALTITUDE AT NORMAL THRUST AT END OF CLIMB (2)

Figure 36.- C-5A basic radius mission; payload = 265,[illegible] lb.



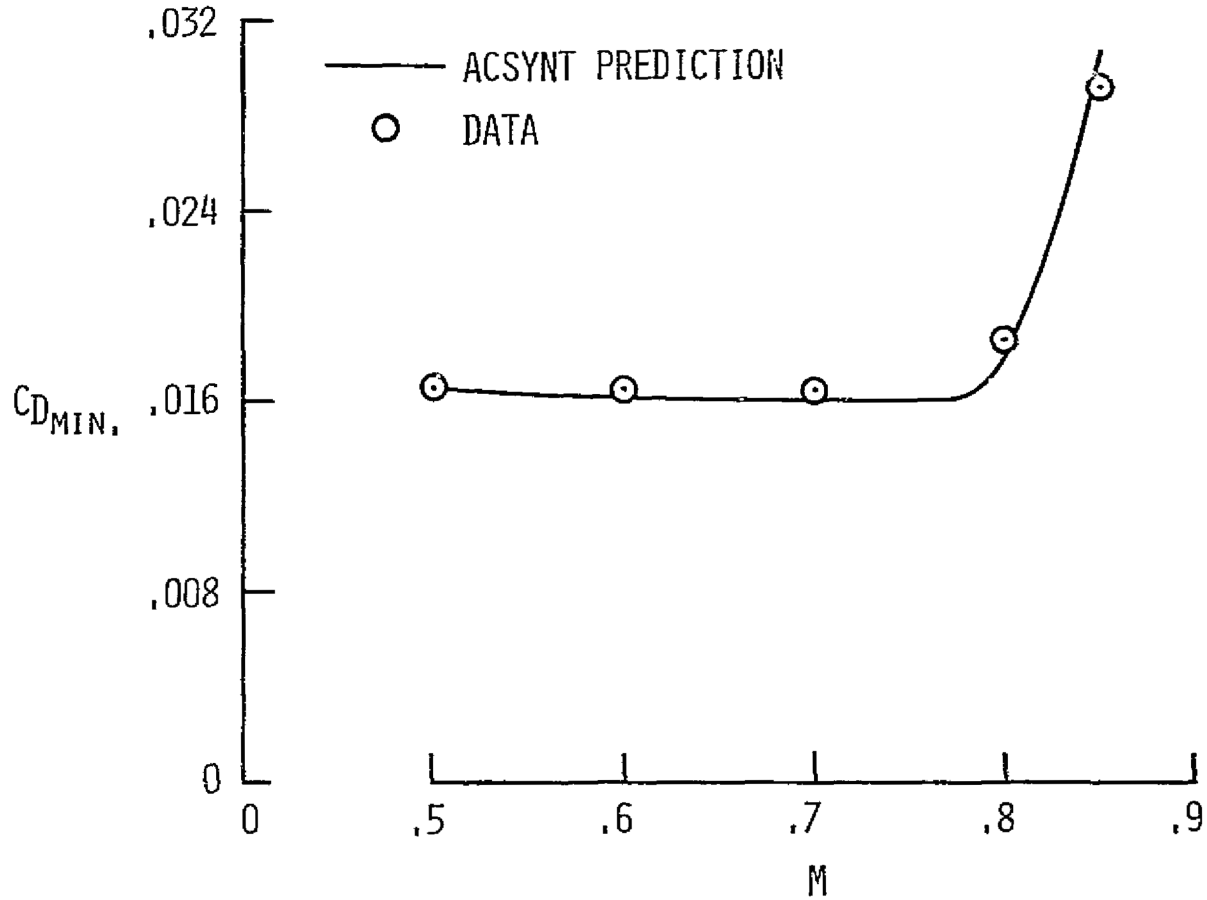
(a) Top view.

Figure 37.- Computer graphics display of ACSYNT-generated C-5A.



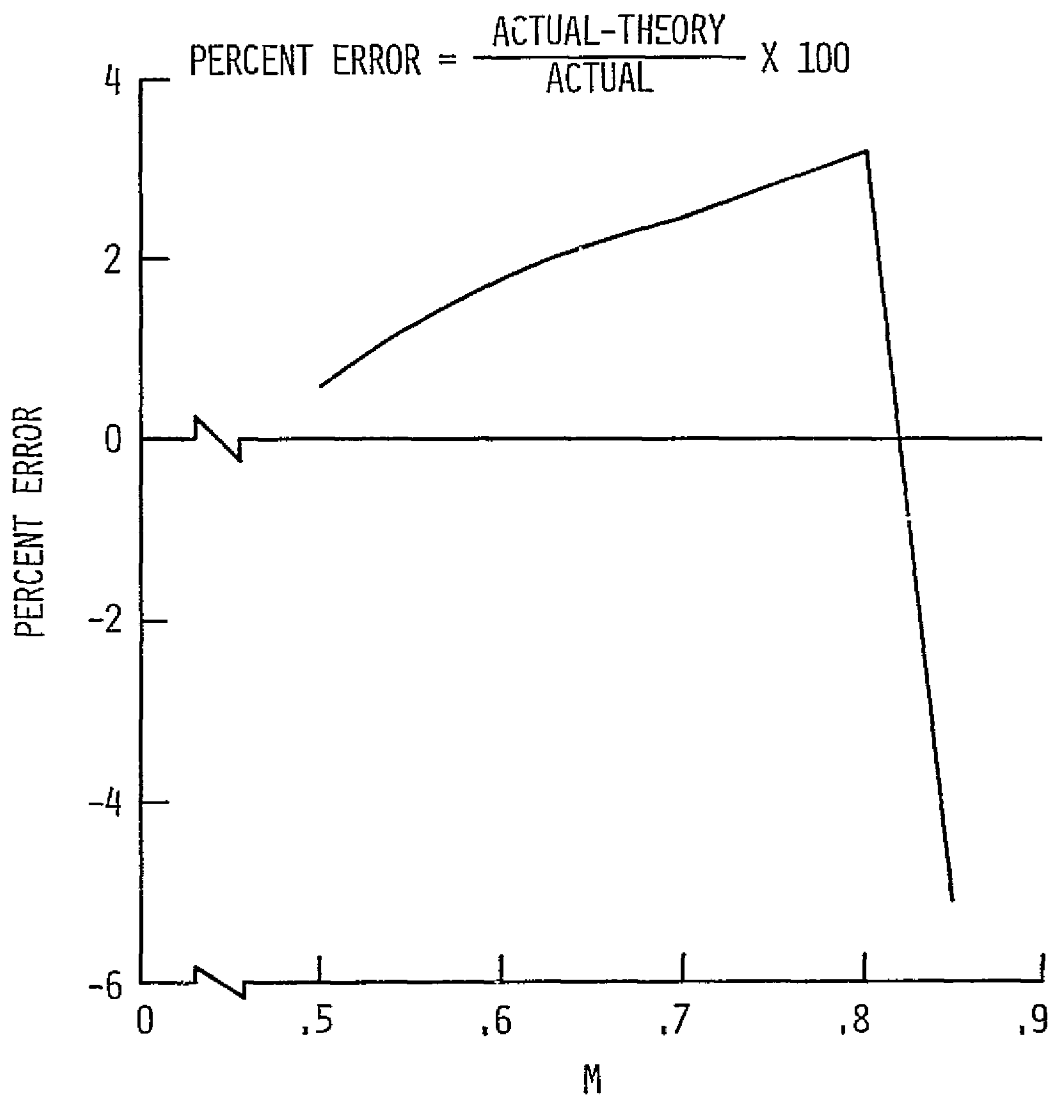
(b) Bottom view.

Figure 37.- Concluded.



(a) Minimum drag coefficient versus Mach number.

Figure 38.- C-5A minimum drag correlation; altitude = 30,000 ft.



(b) Error analysis.

Figure 38.- Concluded.

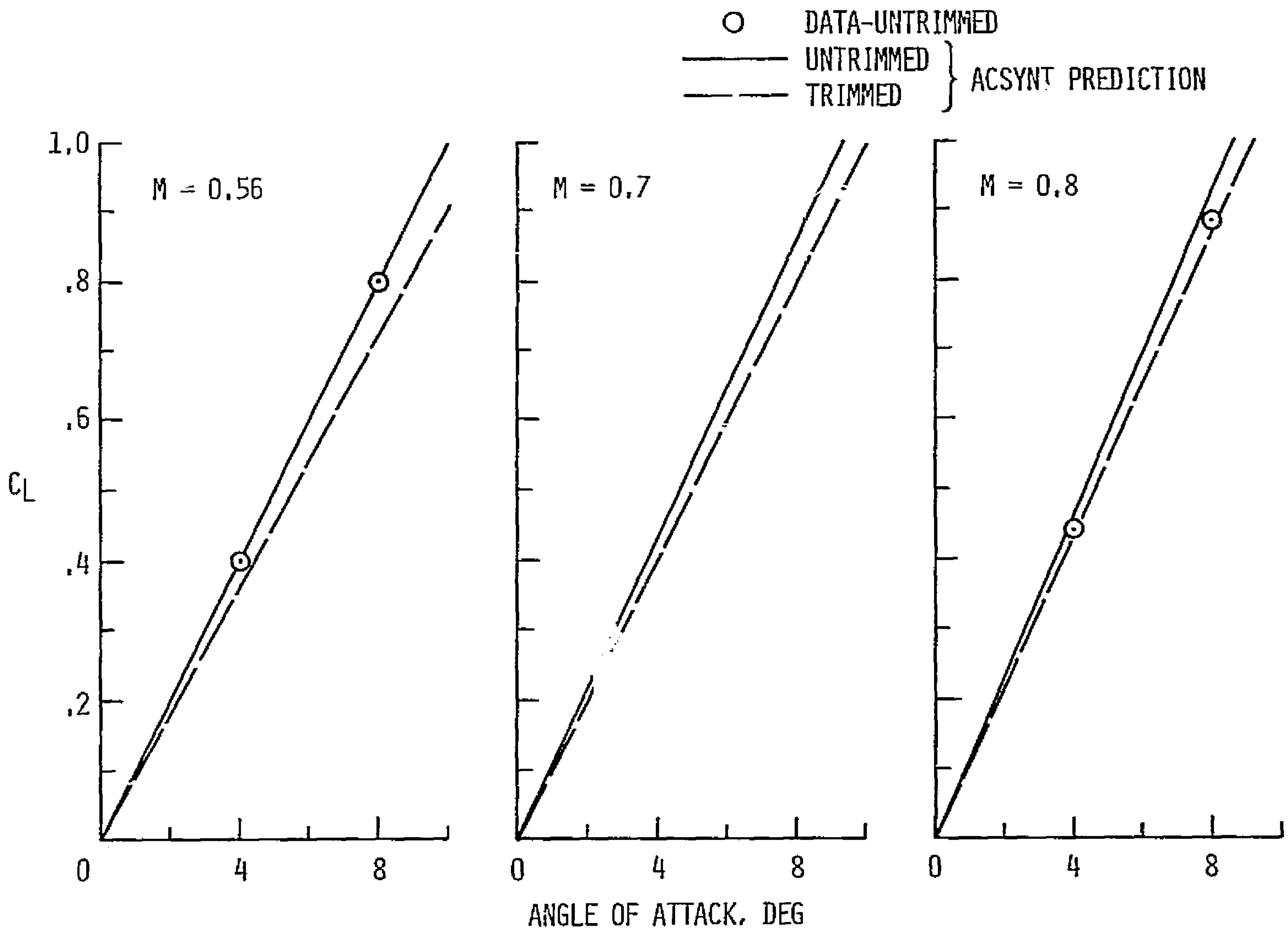


Figure 39.- C-5A lift curve correlation.

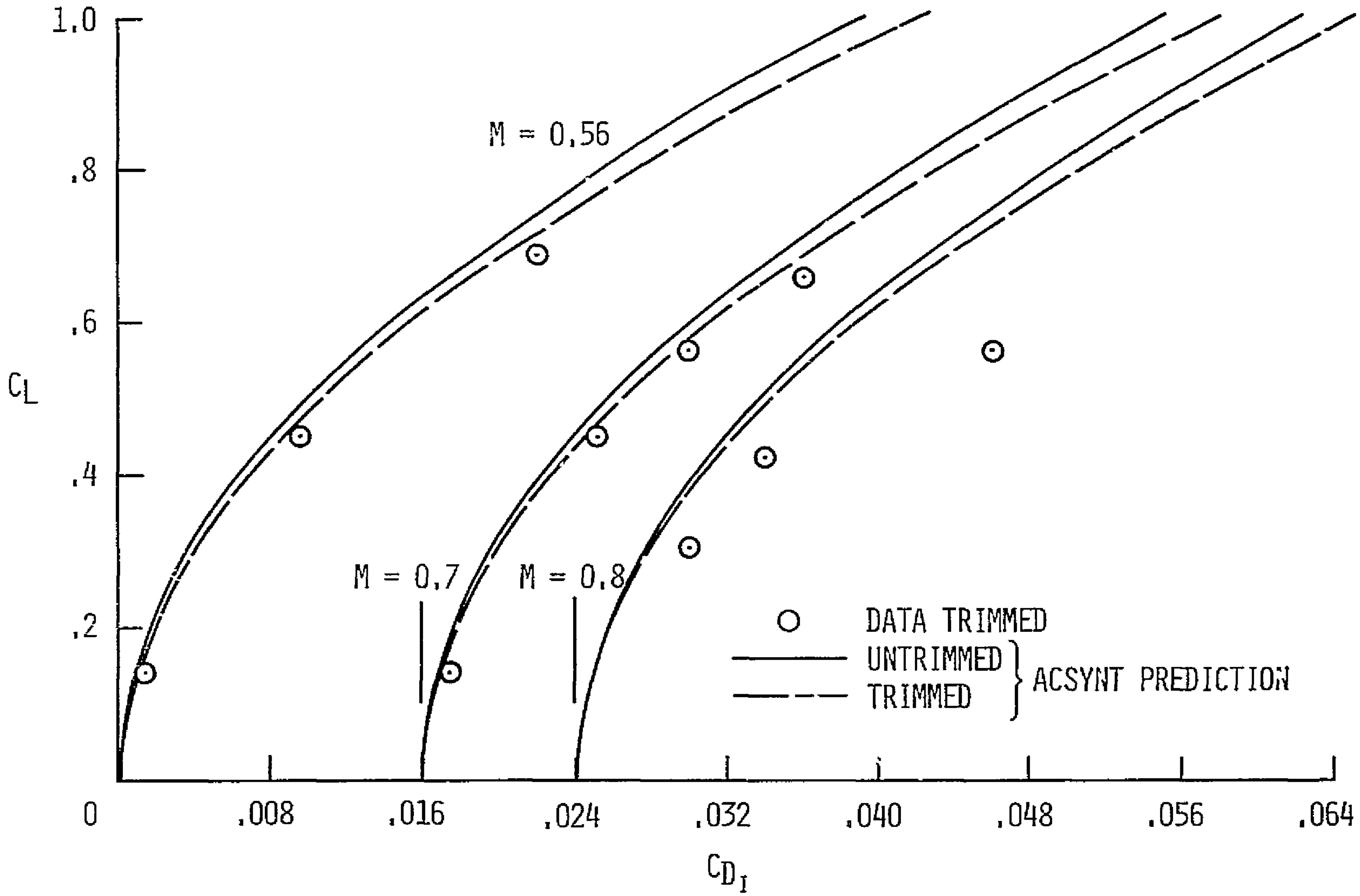


Figure 40.- C-5A induced drag correlation.

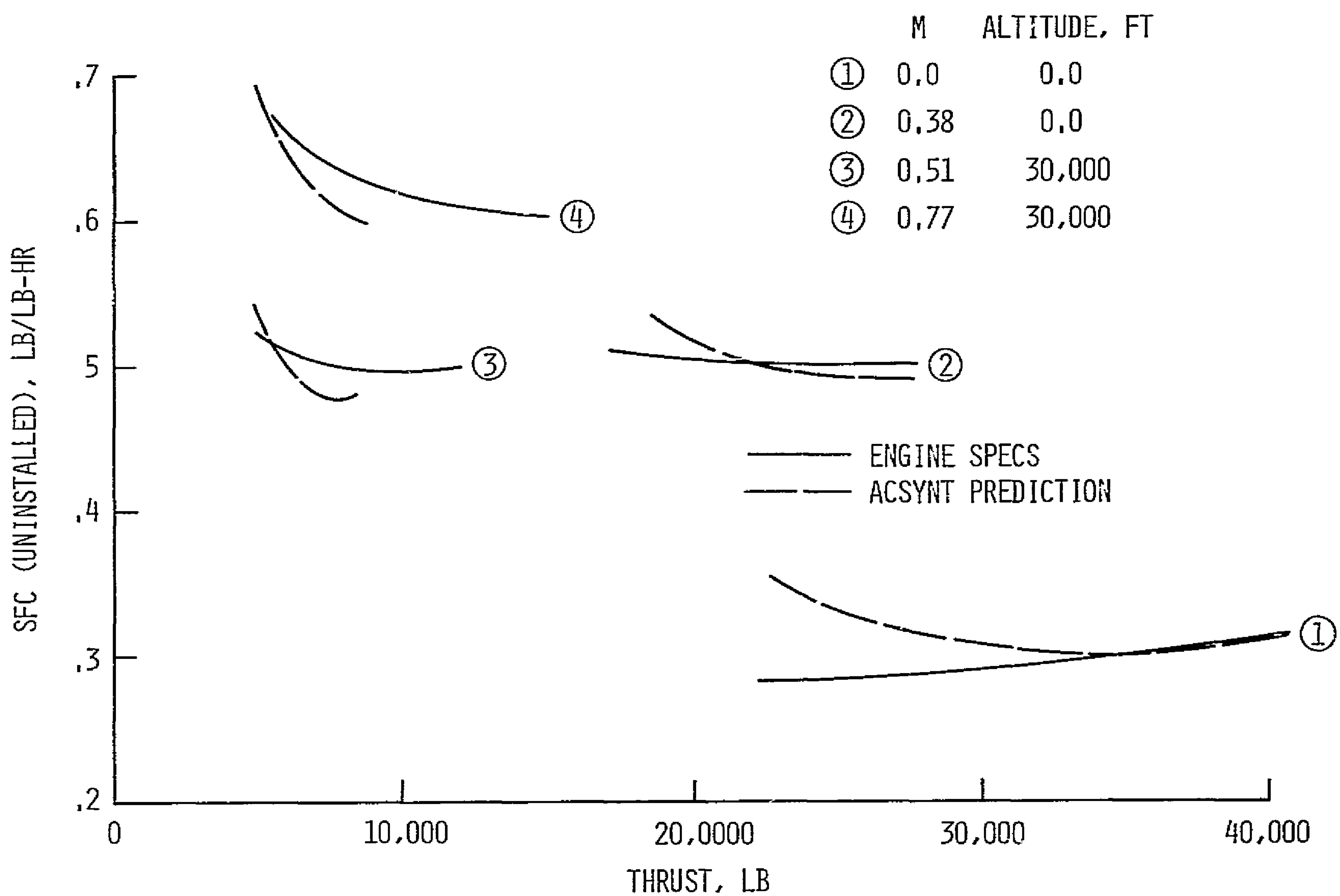


Figure 41.- General Electric TF39-1 engine correlation (C-5A engine).

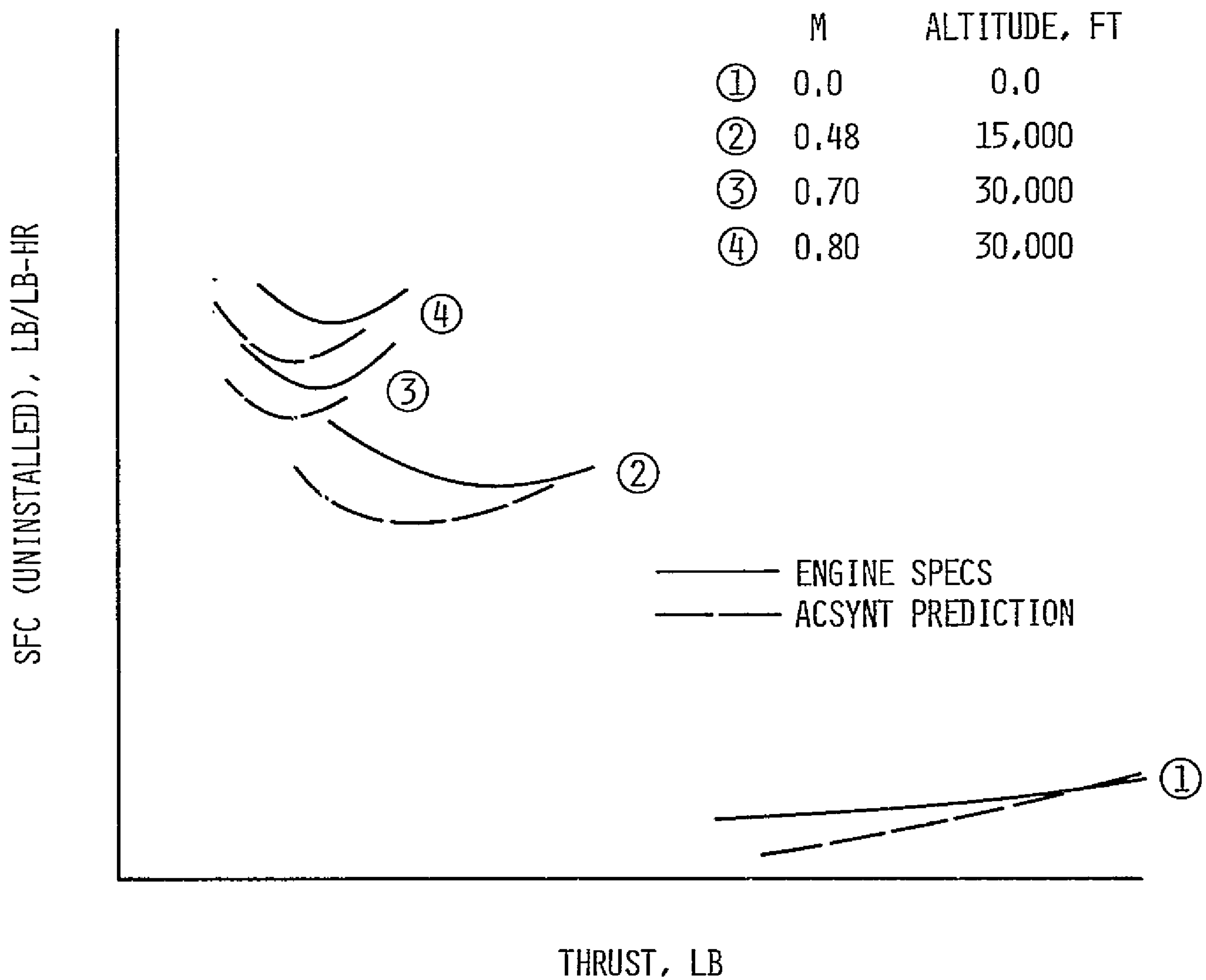


Figure 42.- Pratt and Whitney JT9D-25 engine correlation.

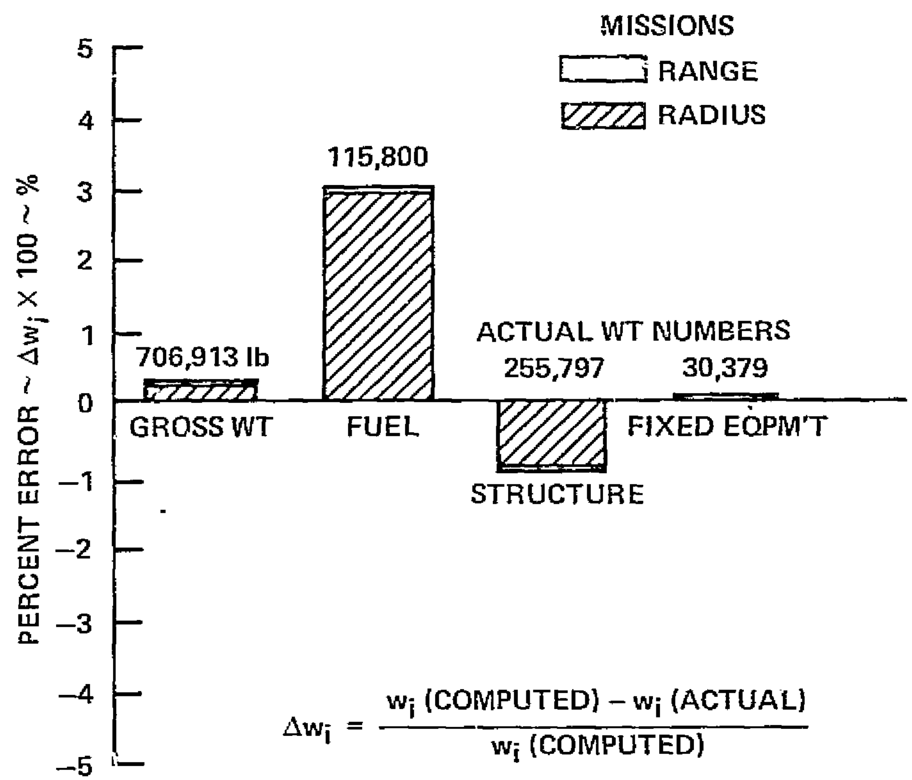


Figure 43.- ACSYNT-simulated C-5A compared to actual C-5A weights.

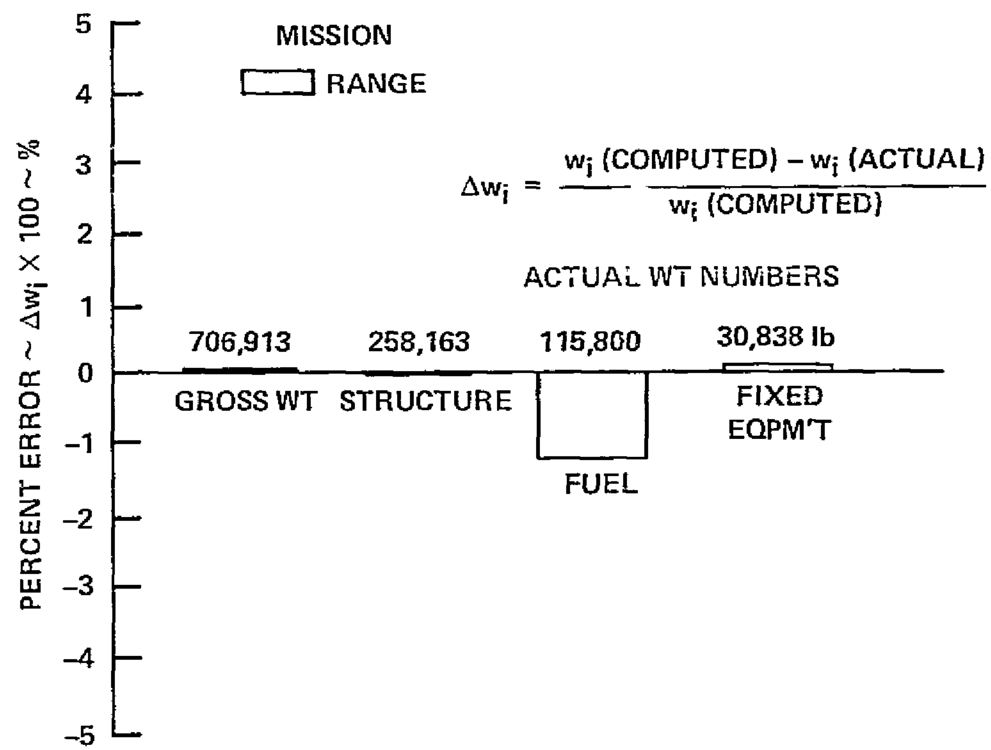
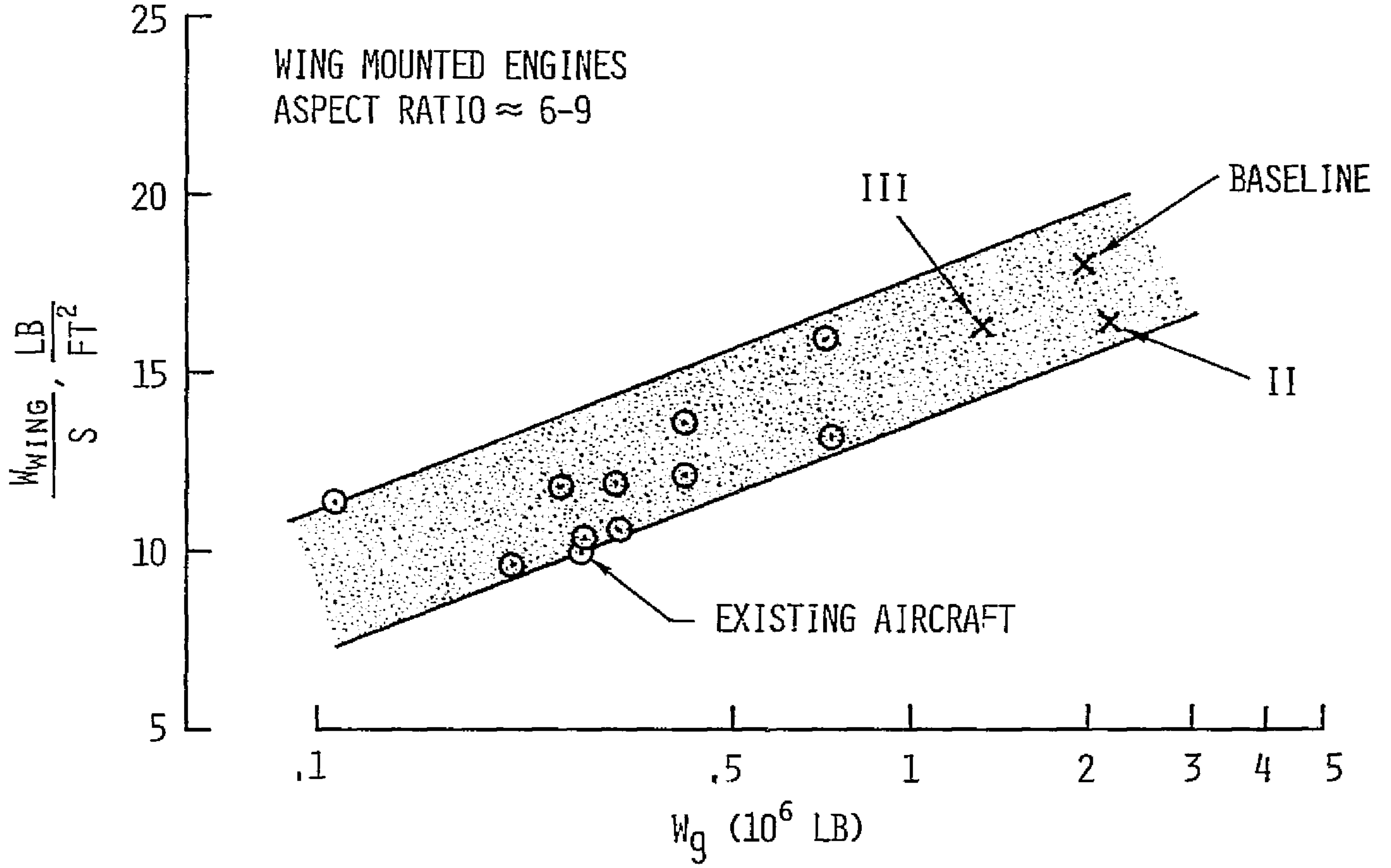
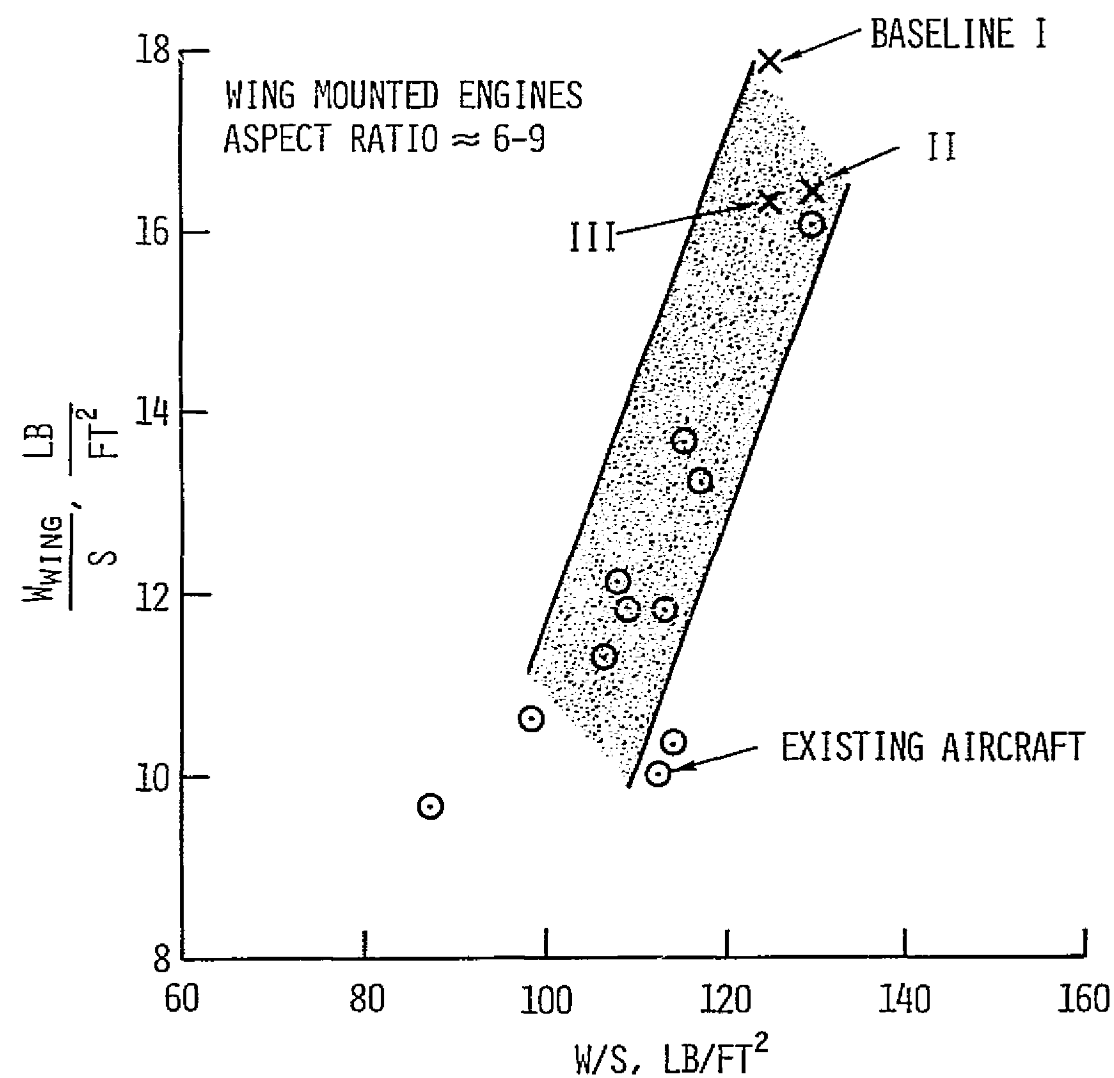


Figure 44.- Optimized ACSYNT aircraft for C-5A basic range mission compared to actual C-5A weights.



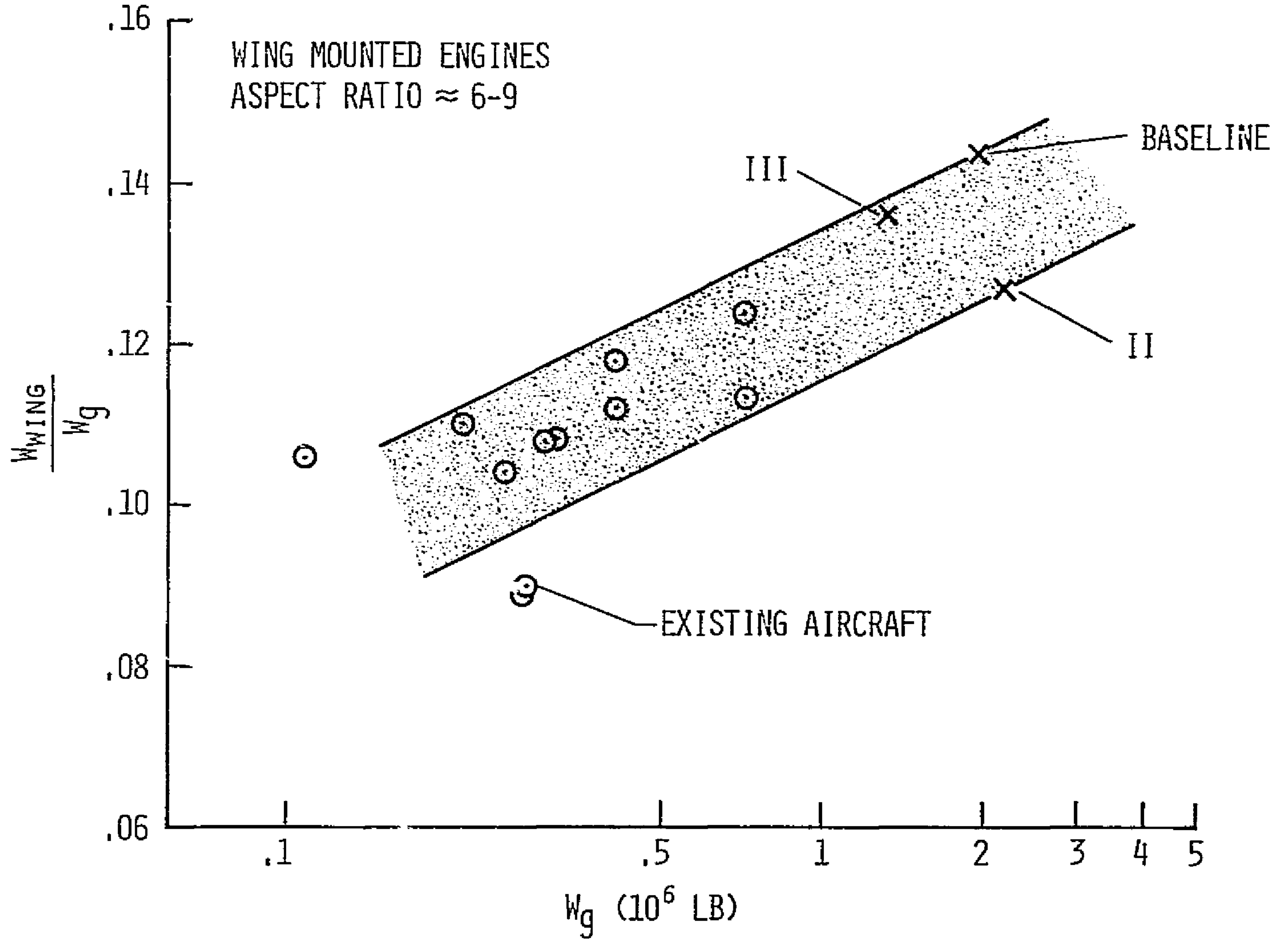
(a) Unit wing weight versus gross weight.

Figure 45.- Substantiation of calculated wing weights.



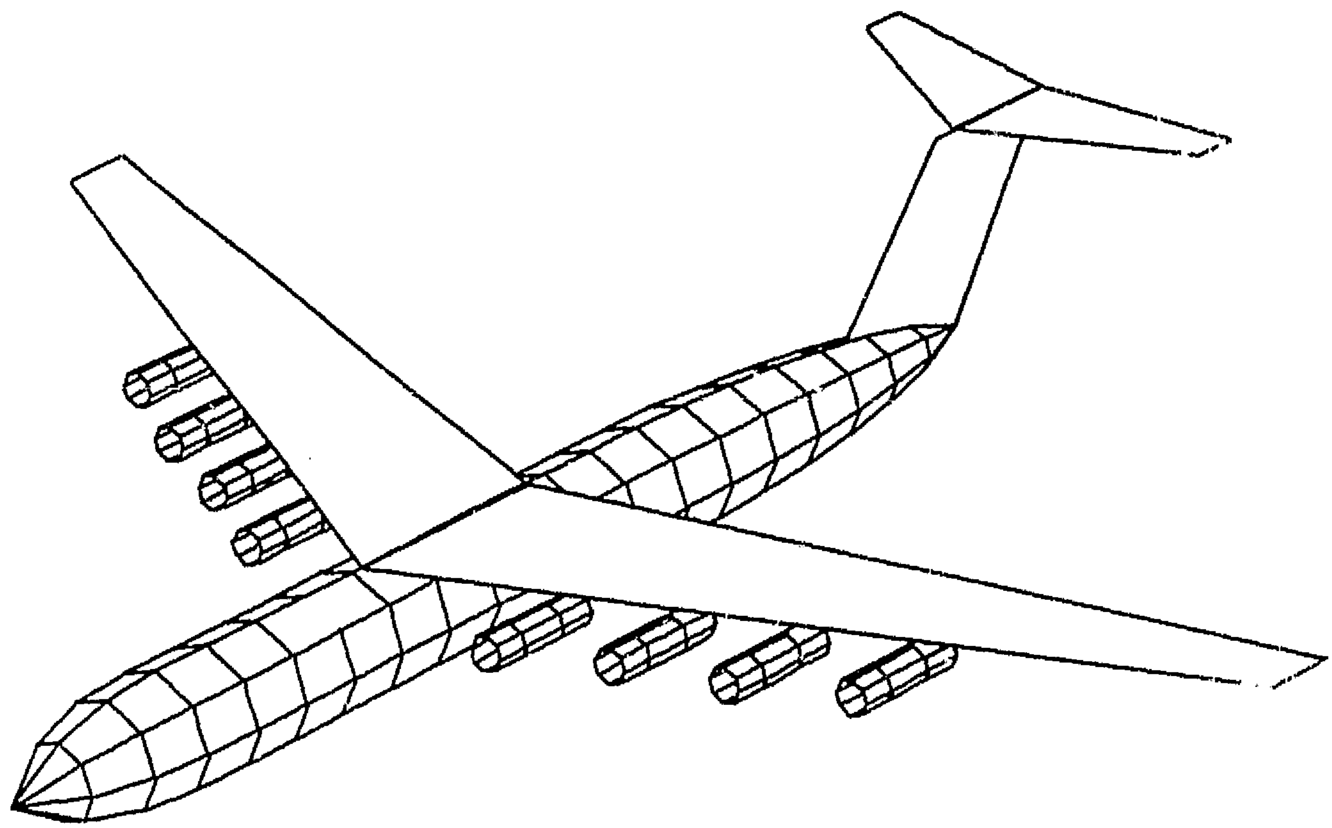
(b) Unit wing weight versus wing loading.

Figure 45.- Continued.



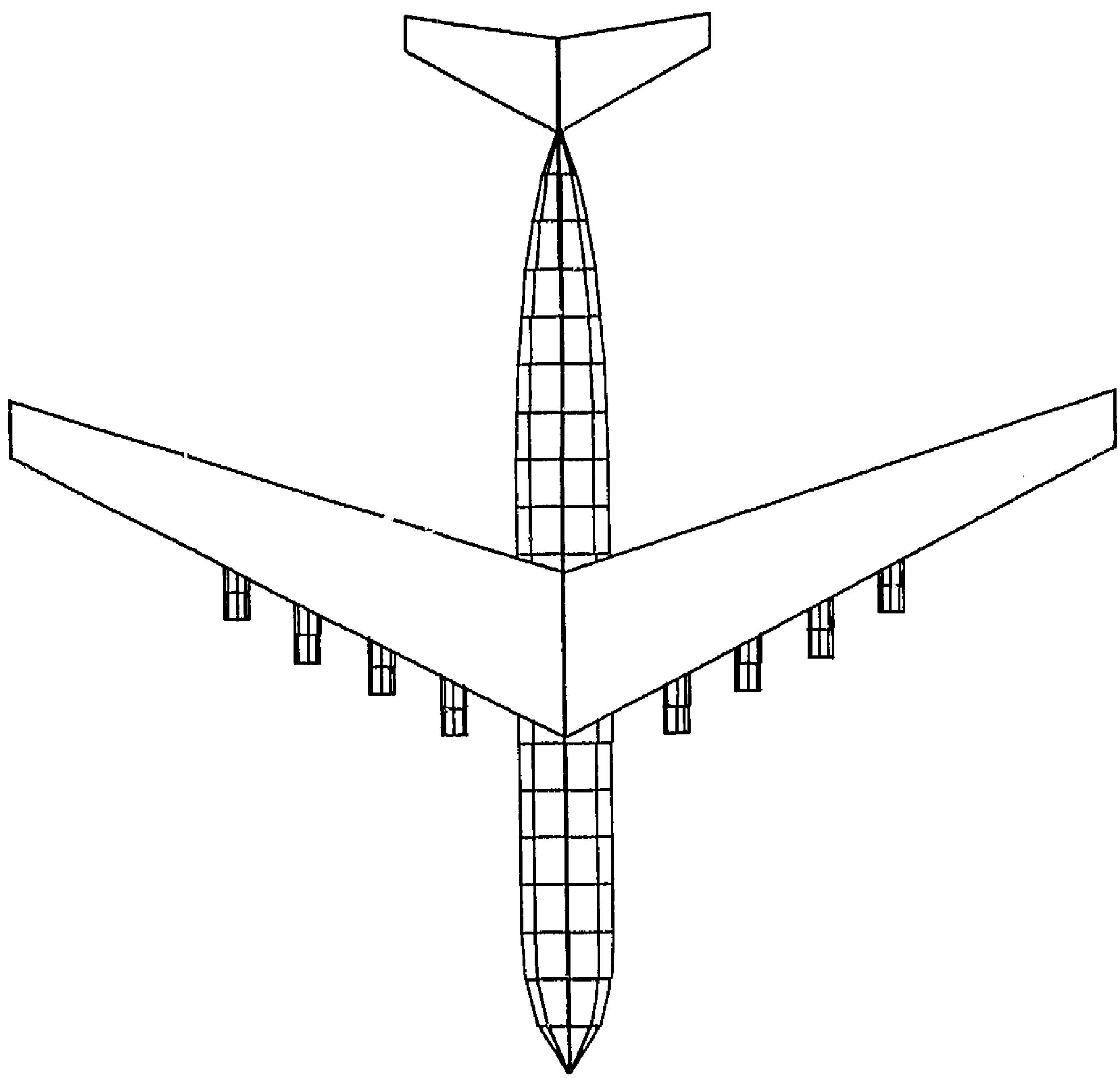
(c) Wing weight fractions versus gross weight.

Figure 45.- Concluded.



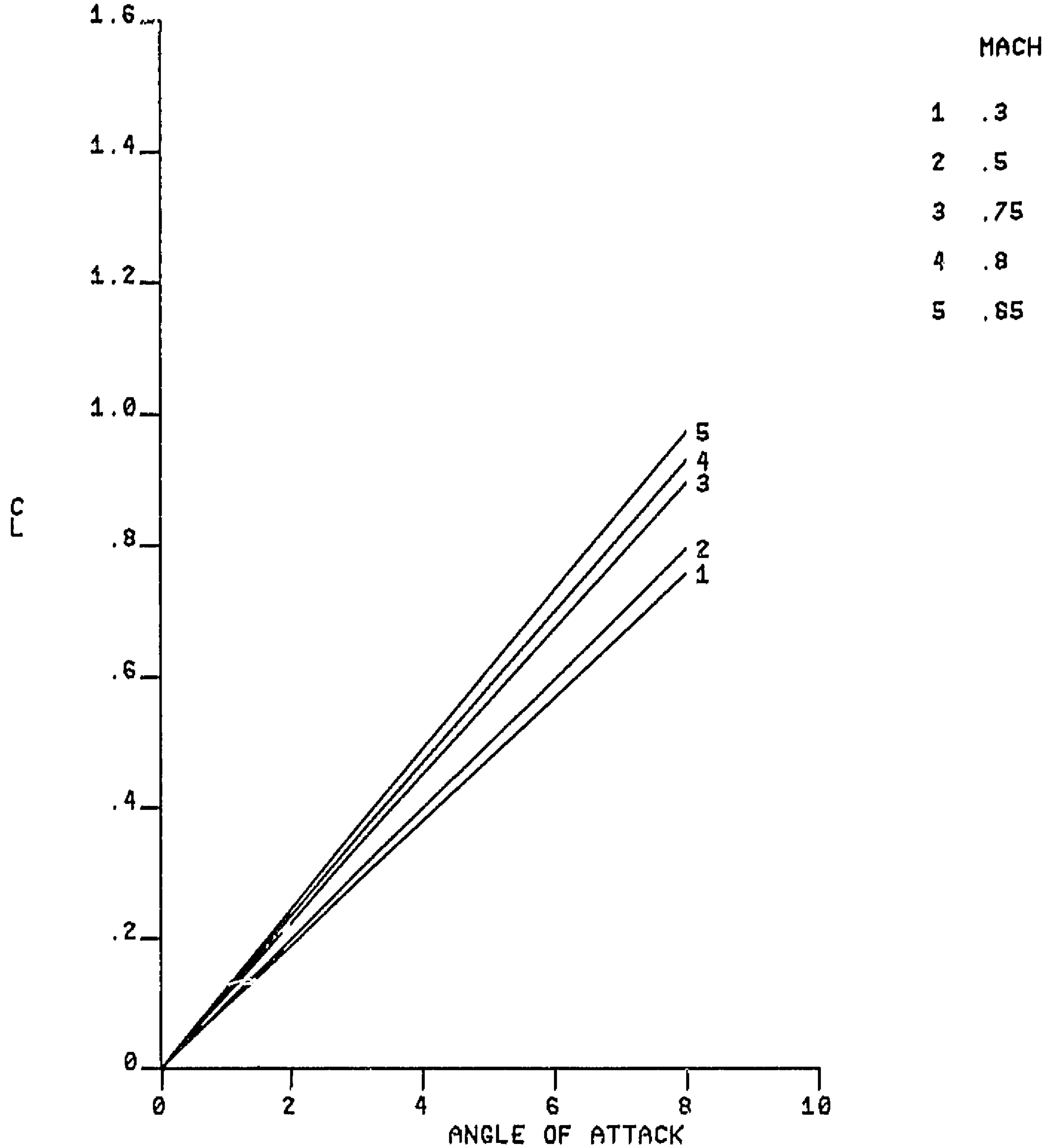
(a) Three-quarter top view.

Figure 46.- Computer graphics display of results for baseline I configuration.



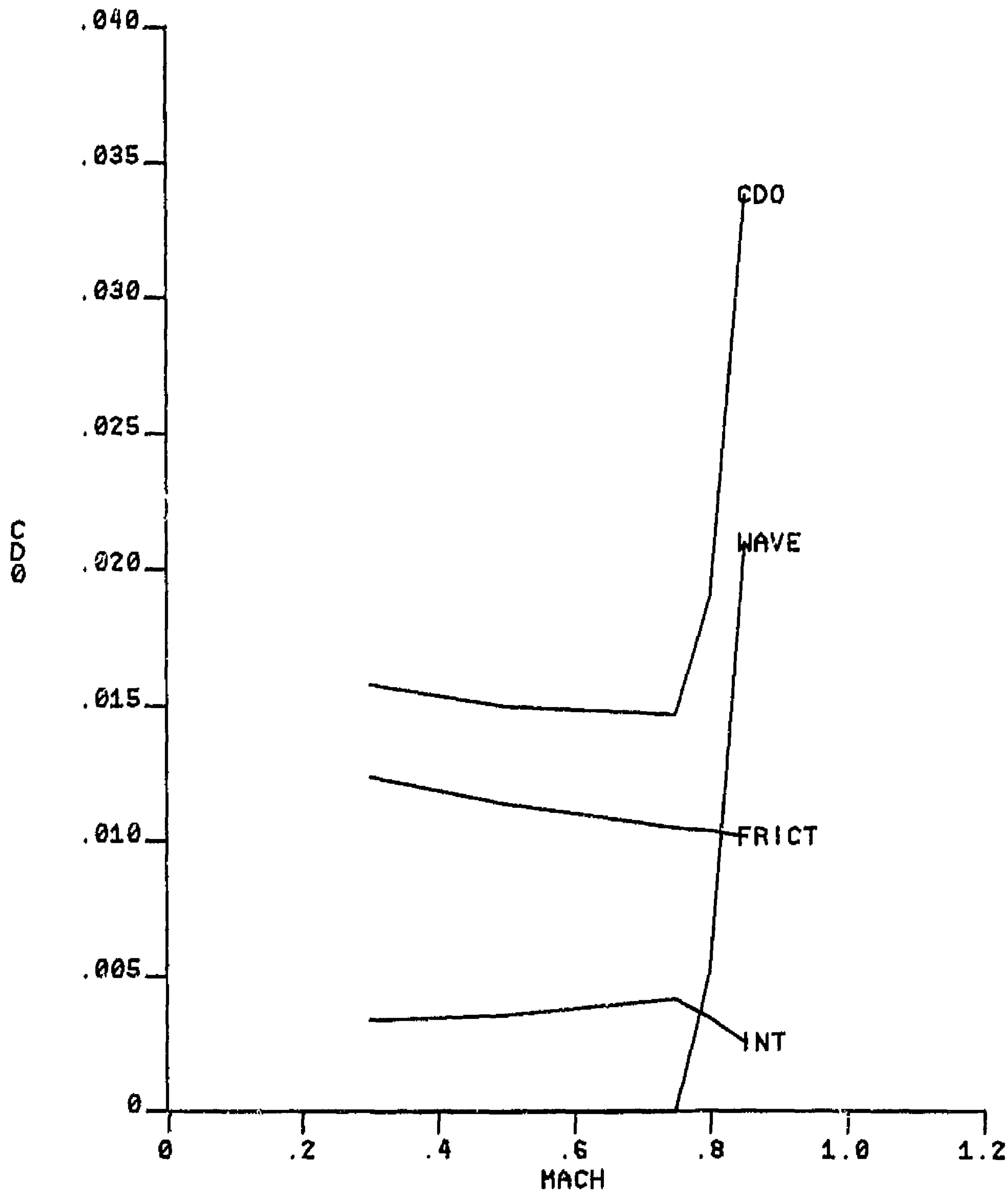
(b) Plan view.

Figure 46.- Continued.



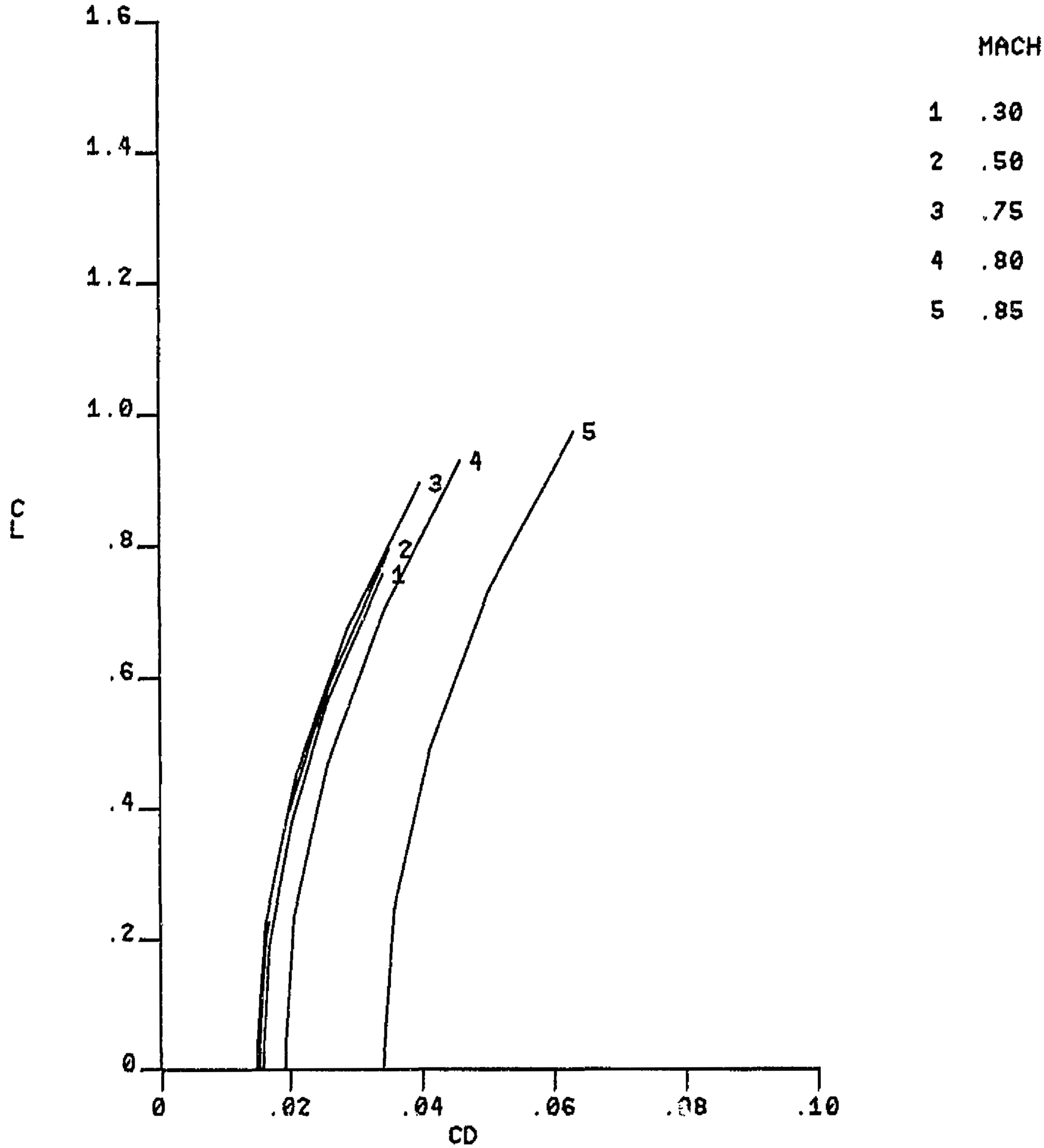
(c) Lift curves.

Figure 46.- Continued.



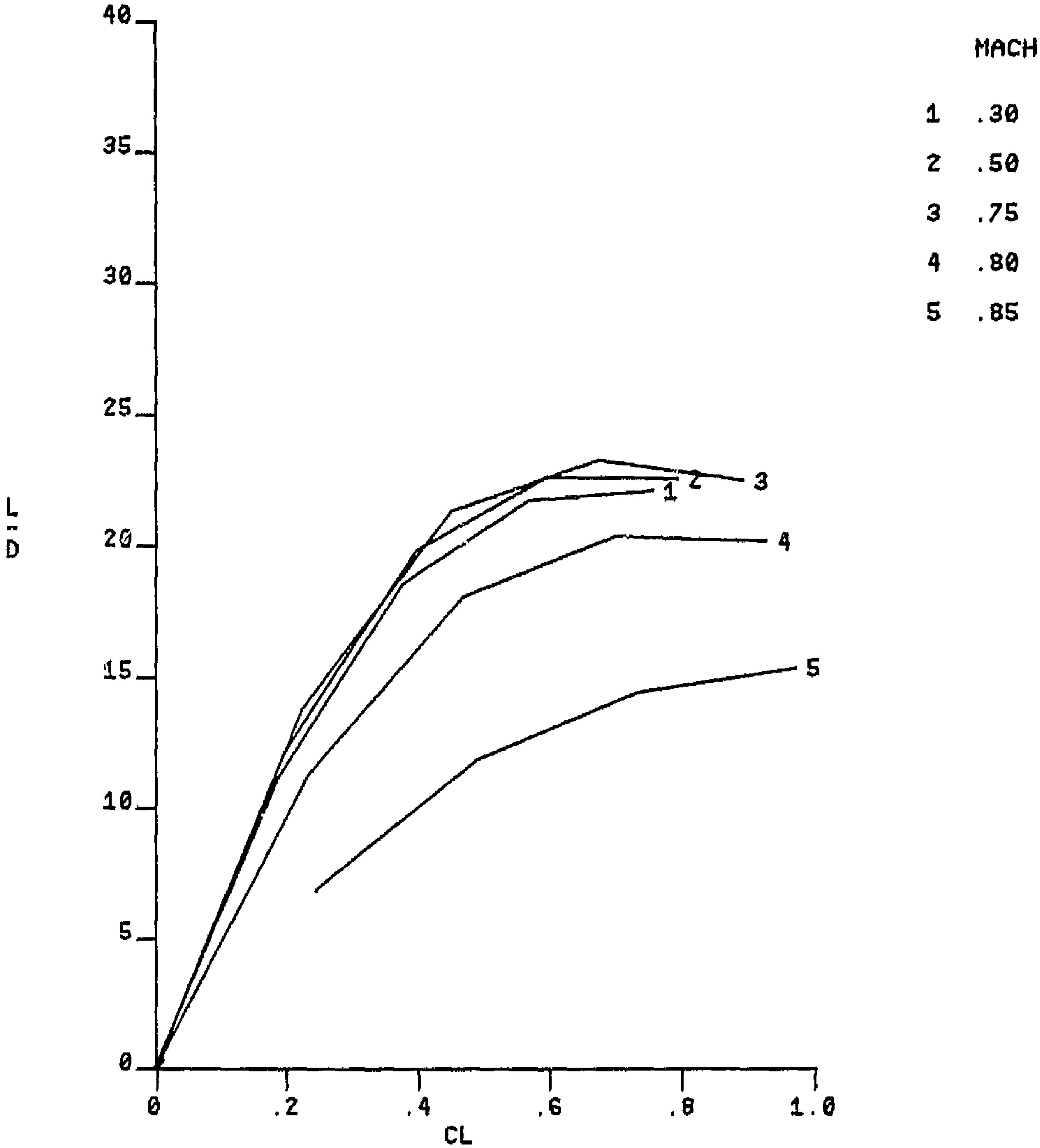
(d) Zero-lift drag buildups.

Figure 46.- Continued.



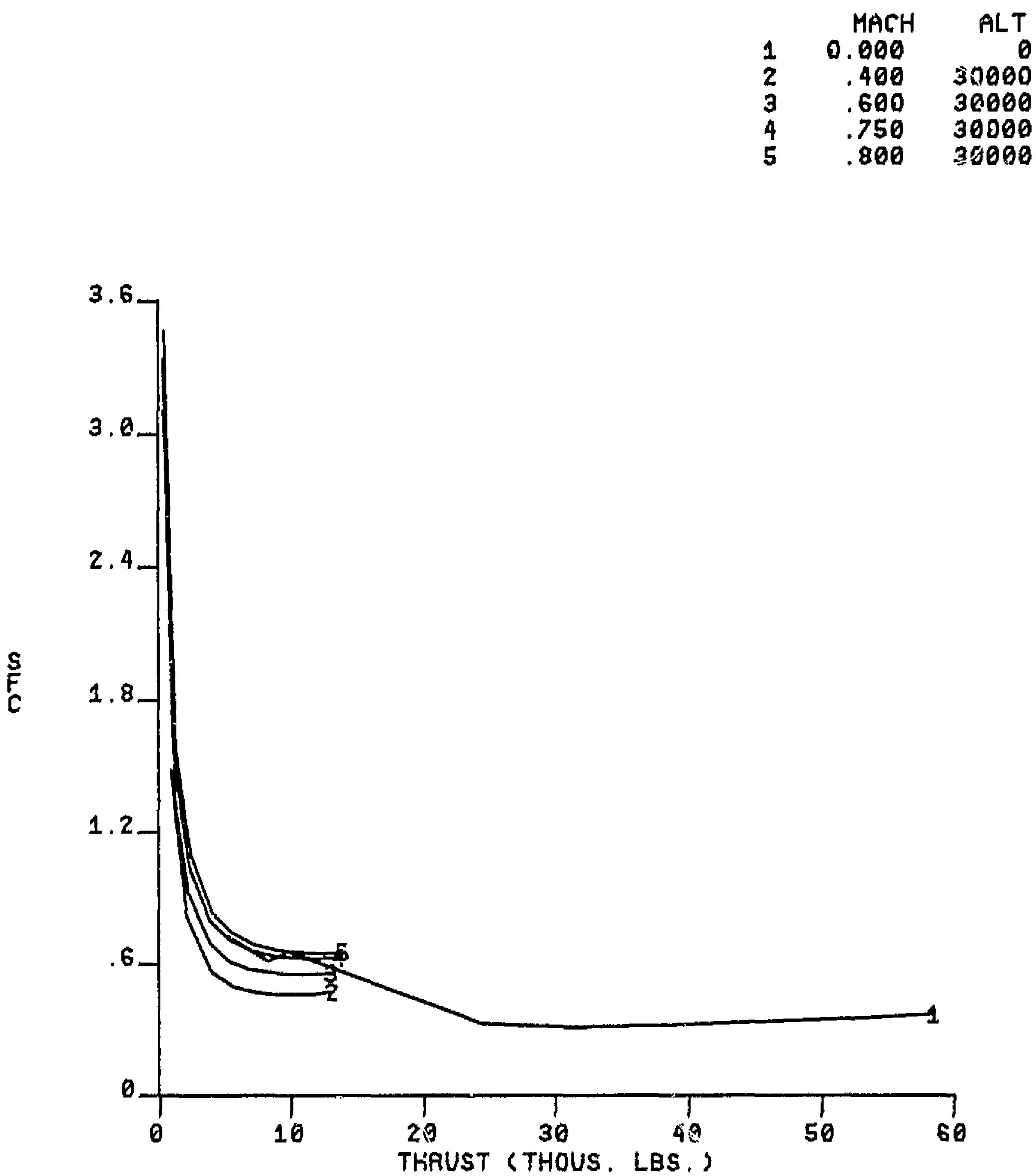
(e) Drag polars.

Figure 46.- Continued.



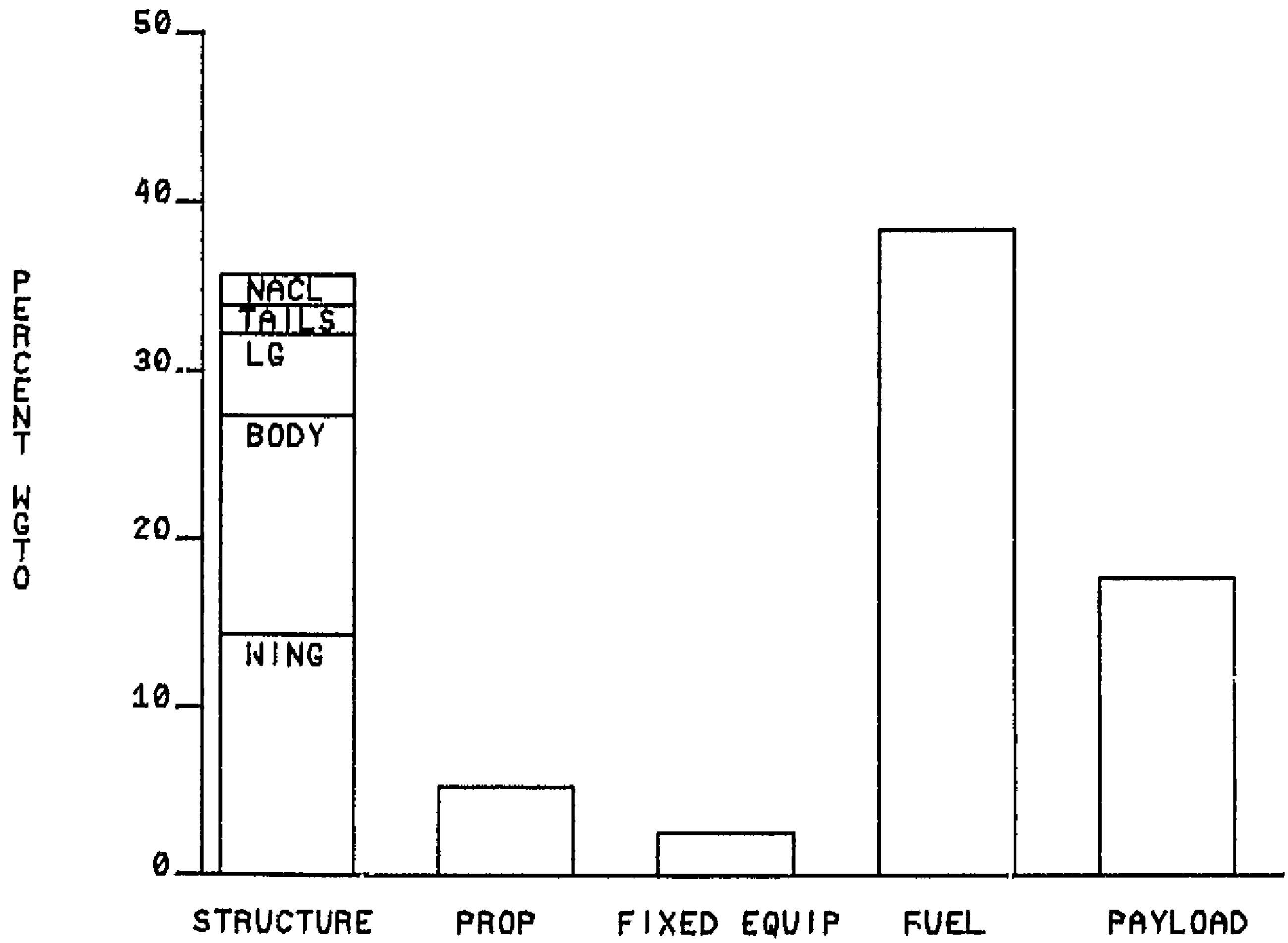
(f) Lift/drag ratios.

Figure 46.- Continued.



(g) Throttled engine characteristics.

Figure 46.- Continued.



(h) Component weight fractions.

Figure 46.- Concluded.