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AIRCRAFT WING WEIGHT BUILD-UP METHODOLOGY WITH MODIFICATION FOR MATERIALS AND CONSTRUCTION TECHNIQUES

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16. Abstract

This study defines an aircraft wing weight estimating method based on a component buildup technique. A simplified analytically derived beam model, modified by a regression analysis, is used to estimate the wing box weight, utilizing a data base of 50 actual airplane wing weights. Factors representing materials and methods of construction were derived and incorporated into the basic wing box equations. Weight penalties to the wing box for fuel, engines, landing gear, stores and fold or pivot are also included. Methods for estimating the weight of additional items (secondary structure, control surfaces) have the option of using details available at the design stage (i.e., wing box area, flap area) or default values based on actual aircraft from the data base.

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PREFACE

This report presents an aircraft wing weight estimating method based on an analytical approach which is sensitive to material and construction techniques. This study was sponsored by the National Aeronautics and Space Administration under contract number NAS2-9805. Mr. Gary C. Hill monitored the study for the AMES Research Center. Work was performed in two phases, between December 1977 and December 1978, and later between April 1979 and September 1980 by the Weight and Mass Properties Control Section of the Grumman Aerospace Corporation.

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LIST OF SYMBOLS

b	Wing Span, ft
b'	Folded Wing Span or Pivot Span for Variable Sweep, ft
B	Weight of Body and Contents, lb
C_1, C_2	Constants
C_R	Wing Root Chord Length, in.
C_T	Wing Tip Chord Length, in.
$\cos \Lambda$	Cosine of Sweep Angle at 40% Chord
F	Allowable Cover Stress, lb/in.^{2}
F_S	Ultimate Developed Shear Stress, lb/in.^{2}
F_W	Total Thrust of Wing Mounted Engines
FDGW	Flight Design Gross Weight, lb
HP_W	Total Horsepower of Wing Mounted Engines
K_{BW}	Aileron, Elevon, Flaperon or Deceleron Balance Weight Factor
K_{CT}	Substructure Carry - Thru Factor
K_{FSCVR}	Cover Fail - Safe Factor
K_{LED}	Leading Edge Device Factor
K_{MG}	Main Landing Gear Factor
K_{MTLCVR}	Cover Material/Construction Factor
K_{MTLSUB}	Substructure Material Factor

K_{ROLL}	Elevon, Flaperon or Deceleron Factor
K_{TS}	Triple Slotted Flap Factor
$K_{TEMPCVR}$	Cover Temperature Effects Factor
K_{WS}	Variable Sweep Factor
LDGW	Landing Design Gross Weight, lb
M	Ultimate Vertical Bending Moment, in. -lb
MCGW	Maximum Clean Gross Weight, lb
MZWFW	Maximum Zero Wing Fuel Weight, lb
n	Ultimate Load Factor (Maneuver) at FDGW
N_B	Number of Spanwise Beams
N_{BOX}	Ultimate Load Factor (Maximum of Maneuver & Gust)
N_{GUST}	Ultimate Load Factor (Gust)
N_{LDGW}	Ultimate Load Factor at LDGW
N_X	Ultimate Axial Running Load in Cover, lb/in.
P	Ultimate Axial Load in Cover, lb
q	Ultimate Shear Flow in Beams, lb/in.
S_{BEAM}	Beam Web Area, ft^2
S_{BOX}	Wing Box Area, ft^2
S_{FLAP}	Trailing Edge Flap Area, ft^2
S_{LED}	Leading Edge Device Area, ft^2
S_{LEF}	Leading Edge Flap Area, ft^2
S_{MGDR}	Main Landing Gear Door Area, ft^2

S_{ROLL}	Aileron, Elevon, Flaperon or Deceleron Area, ft^2
S_{SLAT}	Leading Edge Slat Area, ft^2
S_{SPOIL}	Spoiler Area, ft^2
S_W	Total Wing Area, ft^2
S_{WSB}	Wing Speed Brake Area, ft^2
t	Beam Web Thickness, in.
t_{COVER}	Cover Thickness, in.
TOGW	Take - Off Gross Weight, lb
T_R	Wing Root Thickness, in.
T_T	Wing Tip Thickness, in.
V	Ultimate Vertical Shear Load at Body Attachments, lb
V_L	Limit Speed, knots EAS
V_S	Stall Speed at LDGW, knots
W_{CVR}	Cover Weight, lb
W_{SUB}	Substructure Weight, lb
W_{WFUEL}	Wing Fuel Weight, lb
W_{WING}	Total Wing Weight, lb
$W_{WSTORES}$	Summation of Heaviest Store Weight on all Wing Stations Including Drop Tanks, lb
y	Spanwise Location of Wing Center of Pressure, ft
y'	Location of Wing Center of Pressure Along Structural Axis, ft
ρ	Material Density, lb/in.^{3}
AR	Wing Aspect Ratio

λ	Wing Taper Ratio
(t/c)	Wing Thickness to Chord Ratio

SUMMARY

This study defines an aircraft wing weight estimating method based on a component buildup technique. A simplified analytically derived beam model, modified by a regression analysis, is used to estimate the wing box weight, utilizing a data base of 50 actual airplane wing weights. Factors representing materials and methods of construction were derived and incorporated into the basic wing box equations. Weight penalties to the wing box for fuel, engines, landing gear, stores and fold or pivot are also included. Methods for estimating the weight of additional items (secondary structure, control surfaces) have the option of using details available at the design stage (i.e., wing box area, flap area) or default values based on actual aircraft from the data base.

INTRODUCTION

The objective of this study is to derive a theoretically based, empirically corrected wing weight method and to define and derive weight influence factors for materials and methods of construction and design philosophies. The method will provide correct trends for design tradeoff studies as well as reasonable accuracy. An extensive existing data base of metal wings of various aluminum alloys plus the F-14A and F-15A whose wing boxes are made entirely or partly of titanium were used. A simplified beam model similar to the Grumman "Level II" method was chosen to provide a theoretical basis for the structural analysis. A substantial amount of knowledge on material and construction techniques was accumulated and compiled in a unique data base. While some general information is available in the open literature, the actual details (alloys, stiffener spacings, rib construction, design philosophy) used to derive weight correction factors for the data base aircraft were often obtainable only from the manufacturers. In all, sufficient information was obtained to derive material/construction factors for 22 aircraft of the existing data base of 50 aircraft. Most of these material data were acquired with the assistance of Mr. Gary Hill of the NASA Ames Research Center and Mr. Gerry Seidel of NADC, Johnsville, PA.

IMPACT OF VARIOUS REQUIREMENTS, CRITERIA AND DESIGN CONSIDERATIONS ON WING BOX WEIGHT ESTIMATES

Material Technology

Metals.

Aluminum alloys: Virtually all of the aircraft structures included in the data base are constructed primarily of aluminum. Aluminum alloys are lightweight, corrosion-resistant, and are easily fabricated in a variety of forms. A significant reduction in mechanical properties in environments of about 300°F limits the use of aluminum alloys on aircraft designed for flight above Mach 2.5 and on local areas of severe thermal environment (e.g., engine exhaust).

Although the composition of aluminum alloys have changed with time due to material technology, the basic mechanical properties have not experienced tremendous improvement. Considerations such as stress corrosion resistance and damage tolerance have limited the application of the higher strength alloys (e.g. 7075-T73 type alloy is usually preferred over higher strength 7075-T6 alloy for better stress corrosion resistance).

The most popular alloys for utilization in current aircraft are generally the 7XXX-series. 7050-TXXXX alloys for sheet or plate and 7049-TXX alloys for forgings are likely candidate materials because of resistance to stress and exfoliation corrosion. Their mechanical properties are approximately 90-100% of the 7075 and 7079 alloys previously used in similar applications (e.g., the material in many of the data base wings). Alloy 7475-T7651 has been developed primarily for applications requiring high fracture toughness. Its mechanical properties are also approximately 90% of previously used 7075-T651 alloy, but its fracture toughness far exceeds any other aluminum alloy of comparable strength. Potential advantage exists for use of 7075-T651 in wings designed to meet the damage tolerance criteria defined by specification MIL-A-83444.

Aluminum will still be used extensively in future aircraft structure in the form of more recently developed alloys. These alloys will not provide noticably lighter or stronger structure; however they will provide a structure which is more corrosion-resistant and damage-tolerant. Rather than reduce the weight of structure, they will prevent the weight from increasing above the data base when the requirements of new design criteria are adhered to.

High strength steel: Steel used in airframe structure consists of alloys with a wide range of maximum strengths. The more commonly used are the higher strength alloys. Application includes major attachment fittings, landing gear components, hinge fittings and control surface tracks and linkage. The usual criteria for steel usage are high strength requirement, high temperature environment, or a combination of both. Clearance or usable space limitations may also dictate the use of steel. For example, the F-14 tail support frame required steel structure because of space limitations around the engine. Use of steel for basic wing box structure as in the early 1960s (e.g., F-111 Center Section), was replaced on later designs requiring high strength material (e.g., F-14) by titanium alloys (See Titanium discussions). Since the current and projected use of steel in wing box structure is limited to local fittings, there appears to be no great impact on wing box weight due to steel alloys in the near future.

Titanium Alloys: Titanium alloys used for aircraft structure are relatively lightweight, having a very good strength to weight ratio and are corrosion-resistant. They retain good mechanical properties for prolonged exposure to high temperature of at least 750°F, making titanium a good material choice for high performance aircraft. Due to its high strength, good strength to weight ratio and fatigue resistance, titanium has replaced steel in many structural applications (e.g., F-14 wing box). The major drawback of titanium has been cost, especially where aluminum can meet the requirements within an acceptable weight penalty.

Damage tolerance studies have revealed that certain titanium alloys, although they exhibit high strength and good fatigue characteristics, suffer from rapid crack growth rates. This reduces the structural efficiency of these particular alloys for applications including damage tolerance requirements; however different annealing processes may improve crack growth characteristics minimizing the impact of this criteria (e.g., Beta annealed Ti 6Al-4V alloy may be used in place of mill annealed Ti 6Al-4V and Ti 6Al-6V-2Sn alloys).

Miscellaneous metallics: Although many metals are capable of sustaining wing loads, it appears that metal wing box construction will continue using the three primary materials discussed above. Design properties (Ref. 1) for these materials are shown in Table 1. Other metallics are too specialized or not cost-effective for general use. More development is underway in the area of manufacturing techniques for titanium and this is discussed under Manufacturing Methods. Powdered metallurgy techniques are being developed to form various shapes similar to forgings. These techniques include cold isostatic pressing, hot pressing and hot isostatic pressing.

Advanced composites. - Advanced composite materials offer the best near-term prospect for significantly reducing wing weight. The use of advanced composite materials in first generation applications offers an improvement over historical wing weights of 15-30%. Unlike metals, advanced composites may be tailored to particular applications or requirements for greater structural efficiency. The epoxy-based composites are corrosion and fatigue resistant and may be tailored for good damage tolerance characteristics. The combination of high strength, low density and tailored design accounts for the significant weight savings achieved by utilizing advanced composite design.

The major composite material utilized at this time is Type A graphite/epoxy, due to its high specific shear strength, specific compressive strength, specific stiffness and resistance to crack propogation compared to other materials such as aluminum. Graphite/epoxy hybrid materials are also used with Kevlar, fiberglass and boron, used in combination with the graphite/epoxy. The material in the hybrid depends on the application (i.e., boron for high stiffness). Design properties for various composite materials are shown in Table 2.

Future development which will improve the weight savings potential of advanced composites include:

- High Strain Design - Improved design techniques may allow utilization of the materials maximum strain capability (5000 to 6000 $\frac{\mu\text{-in.}}{\text{in.}}$) instead of the current limits (3000 to 4000 $\frac{\mu\text{-in.}}{\text{in.}}$)
- Post Buckled Strength - Designing to minimize the frequency of buckling the structure at lower load levels, but allowing more buckling at higher less frequent loads is being studied.

TABLE 1. – DESIGN PROPERTIES OF VARIOUS METALS

Alloy	F_{TU}, psi	F_{TY}, psi	F_{SU}, psi	F_{CY}, psi	$F_{0.7}$, psi	η	E, msi	G, msi	Density, lb/in.^{3}
Aluminum									
7075-T6	78,000	70,000	47,000	69,000	69,800	14.5	10.5	3.9	0.101
7075-T651	76,000	69,000	45,000	68,000	66,600	12.5	10.6	3.9	0.101
7075-T7351	67,000	57,000	38,000	56,000	54,300	18.4	10.6	3.9	0.101
7075-T76511	75,000	65,000	41,000	65,000	65,500	34.0	10.7	4.0	0.100
7079-T651	71,000	65,000	42,000	64,000	65,300	14.5	10.8	3.9	0.099
7178-T651	83,000	73,000	43,000	73,000	75,200	14.0	10.6	3.9	0.102
7475-T7351	70,000	59,000	41,000	58,000	58,500	20.0	10.6	3.9	0.101
2024-T3	64,000	47,000	39,000	39,000	38,000	10.5	10.7	4.0	0.100
2024-T351	62,000	47,000	37,000	39,000	37,800	8.5	10.9	4.0	0.100
2024-T81	67,000	59,000	40,000	59,000	59,700	16.0	10.7	4.0	0.100
2024-T851	67,000	59,000	38,000	59,000	59,600	17.5	10.9	4.0	0.100
2124-T851	66,000	57,000	38,000	57,000	57,400	16.0	10.9	4.0	0.100
Titanium									
6AL-4V Ann.	130,000	120,000	76,000	126,000	128,200	33.0	16.4	6.2	0.160
6AL-6V-2SN Ann.	150,000	140,000	91,000	139,000	148,000	45.0	17.5	6.5	0.164
Steel									
PH15-7MO	190,000	170,000	123,000	179,000	184,100	9.3	30.0	11.0	0.277
D6AC	220,000	190,000	132,000	198,000	201,200	25.0	29.0	11.0	0.283

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TABLE 2. – DESIGN PROPERTIES OF COMPOSITE MATERIALS

PROPERTY			UNIDIRECTIONALLY NON-WOVEN										BI-DIRECTIONAL WOVEN							
			BORON/EPOXY AV5505 3M SP290 UNITARY		BORON/EPOXY AV5505 3M SP290 IN HYBRIDS		GRAPHITE/EPOXY (LHS) 3501/AS		GRAPHITE/EPOXY (LHS) T300 N5208/X934		GRAPHITE/EPOXY (LHS) GY70/HYE-1534		KELVAR '49/CE 3305		GRAPHITE/EPOXY HMF 330C 181 STYLE (T300/X934)		KEVLAR '49/EP 181 STYLE		GLASS/EPOXY 7743	
			RT	350 F	RT	350 F	RT	350° F	RT	350° F	RT	350° F	RT	350° F	RT	350° F	RT	350° F	RT	350° F
NO HOLES	F_1^{tu}	KSI	178	154	165	152	169	152	169	165	90	90	168	112+	68.0	63.2	67.0	45.0	86	70
	F_2^{tu}	KSI	10 5	6	10 5	6	6.0	3	4 J	3.3	2.4	2.4	1.6	1.0+	67.6	61.2	67.0	45.0	9.6	6.0
	F_1^{cu*}	KSI	480	480	480	480	162	137	141	140	90	90	40	23+	65.0	32.0	18.0	12.0	63	50
	F_2^{cu}	KSI	40	12	40 0	12	25	15	20	10+	28	16.8	12.1	6.0+	67.0	32.0	18.0	12.0	10.0	10.0
WITH 5/16 IN DIA HOLES[x]	F_1^{tu} GROSS	KSI	80	80	80 0	80.0	76	76	76	76	21	21	95+	63+	56.0	49.0	53+	36+	40+	30+
	F_2^{tu} GROSS	KSI	5 0	3 0	5.0	3 0	3 2	2 0	2 1	1.7	1.2	1.2	.8	.5	54.0	49.0	53+	36+	4+	3+
	F_1^{cu} GROSS	KSI	185	180	185 0	180 0	76	76	76	76	21	21	30+	18+	62.0	26.0	14+	10+	30+	25+
	F_2^{cu} GROSS	KSI	12 0	5 5	12 0	5 5	12 5	7 5	10 0	6.0	14.0	8 4	6.1	3.0	53.5	26.0	14+	10+	5+	5+
	E_{11}^t	MSI	30 3	28 3	30 3	28 3	18.5	17.7	22.0	21 0	42.0	42.0	11 9	9.2+	10.4	10.4	4.5	4.0	6.1	6.1
	E_{22}^t	MSI	2 8	9	2 8	9	1 6	1 1	1.5	1.0	1.0	.65	.64	.5+	9.1	8.5+	4 5	4.0	2.2	.75
	E_{11}^c	MSI	31 8	30 6	31 8	30.6	18 5	17 7	22.0	21.0	42.0	42.0	12.6	9.2+	11.7	11.0+	4.5	4.0	4.9	4.9
	E_{22}^c	MSI	2 8	9	2 8	9	1 6	1 1	1 5	1.0	1.0	.65	.65	.5+	9.0	8.5+	4 5	4.0	2.2	.75
	G_{12}	MSI	723	232	723	232	65	.20	52	.20	.70	.21	.40	.2+	1.5+	.7+	1 2+	.5+	1.22	.50+
	ν_{12}		25	25	.25	25	25	.25	25	.25	25	.25	.30	.30+	.10	.10	.10	.10	.10	.10
	G_i	KSI	250	134	250	134	104	98	>141	140	>80	>80	>40	>23+	65	32.0	>18	>12	>63	>50
	F_{is} D/t 2	KSI	7 0	4 5	7 0	4 5	7 1	5 1	14 2	6.3	4 3	3.6	10.0	5.0+	7.4	5.0	4.0	2.0+	4.3	2.0
	F_{bru}	KSI	140 0	64 0	140 0	64 0	66.0	48 0	66 0	48.0	66.0	48.0	22 0	12.0+	66.0	48.0	24.0	12.0+	47.0	24.0+
	α_{11}	IN/IN/°F	2 5	2 5	2 5	2 5	25	.25	.30	.30	– .58	–.58	2.0	– 2.0	1.4	2.0	0	0	5.5	3.3
	α_{22}	IN/IN/°F	13 0	22 0	13 0	22 0	15 2	20 0	11 0	19.8	16.5	20.0	32.0	36.0+	1.5	1.6	0	0	5.5	3.3
	t	IN	0051		00525		00525		.00525		0055		.00716		.0140		0100		.0096	
	ρ	LB/IN³	07.		075		055		.055		.061		.048		.055		050		.065	
COST $/LB			200		200		40		45		75		30		55		20		5	
REFERENCE			AFML TR 70 231		AFML TR 72-232		AMFL-TR-72-232 NAS 8 26675-2		CONCEPT HDWARE FUSE 4TH Q LMSC DO 55983		CONCEPT HDWARE STAB, 3RD Q NAS 8 26675-2		AFFDL-TR-73-31 DU PONT		LMSC DO 55983 GAC IRAD		AFFDL-TR-73-31 DU PONT CONCEPT HDWARE FUSE 4TH Q		AFML-TR-70-231 MIL HDBK 17A	

NOTES · INTERLAMINAR SHEAR MODULUS G_i IS CUT OFF ON COMP STRENGTH

* ESTIMATED VALUE

x HOLE DIAMETER CORRECTION FACTOR FOR OTHER SIZES

DIA	3/16	1/4	5/16	3/8	1/2
CF	1.142	1.07	1.0	.95	.87

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- Low Density Materials - New lightweight materials with excellent elastic properties are under development. Kevlar is an example of this type of material which is now available.

Design Considerations

Applied loads.

Flight loads: Maneuver and gust are the primary flight conditions that influence the design of aircraft wings. Fighter-attack aircraft wings are generally designed by specified maneuver loads (e.g., symmetrical or rolling pullout), while the design requirements for transport, patrol, ASW and AEW aircraft wings are a combination of maneuver and gust loads. Maneuver loads result from movement of the controls while gust loads are caused by atmospheric turbulence. The magnitude of the maneuver loads is defined in the aircraft specification in terms of load factors, limit speed, pitching and rolling accelerations. The wing must be designed to meet these criteria at the worst possible points within the airplane flight envelope. Gust loads are determined by one or more of the following methods, depending on the requirements of the customer or certifying agency.

- Simplified gust formula as defined in the applicable specification (e.g., FAR Part 25)
- Discrete gust analysis for a given gust velocity and altitude
- Power spectral density - A statistical analysis of the anticipated gust environment.

The maximum vertical load factor at the aircraft center of gravity (maneuver or gust) has been found to be the most satisfactory parameter for representation of applied load affects on wing box weight.

Ground loads: Three ground load conditions which may influence wing weight are:

- Landing loads for wing mounted landing gear. The landing gear reactions generally add weight locally, particularly in ribs, spars and local attachment fittings. The landing vertical load factor is the best defined parameter for determining the impact of these loads on wing weight.

- Crash load requirements which are defined in the aircraft specification. This includes barricade engagement for carrier base aircraft and fuel containment requirements for wing fuel tanks. These loads are difficult to define in an empirical analysis and are generally contained within the equation constants.
- Negative "g" loads on wing. This includes the effect of large concentrated weights mounted on the wing such as engines and external stores.

Other conditions for ground handling, such as jacking loads do not usually have a substantial impact on total wing weight.

Fatigue loads: The previous discussion of flight and ground loads involved static design loads only. To prevent fatigue failure in wing structure a fatigue analysis must be performed dealing with frequency as well as magnitude of loads. This not only considers the frequency due to aircraft environment, but the affect of dynamic response for flexible wings. Studies of the peated loads spectrum result in a safe working stress which is generally used in the design phase and may be utilized for weight estimates. Weight penalties may be determined by analytical methods using the static allowable stress and the safe working stress to calculate the additional material required for the latter. More extensive fatigue analysis and testing are used as the design progresses to verify the integrity of the structure.

Fail-Safe design: Fail-safe criteria imposed on a design requires that even after failure, the wing will remain intact and sustain flight. Fail-safe structure is required for FAA certification under FAR Part 25, and introduces substantial cover and substructure weight penalties to the wing box. Fail-safe is rarely required for military tactical aircraft, but may be incorporated in designs where it can be accomplished without increasing weight or cost appreciably. Isolating the wing bending (cover) material required for a fail-safe design is difficult to assess from weight statements except for those structural members added explicitly and only for that purpose. The majority of the cover weight increment required for fail-safe is included in the rib and splice pads, splice hardware and increased thicknesses to suppress stress levels. The magnitude of the analysis that would be required did not permit breaking out the penalty analytically.

Identifying fail-safe material in the substructure encounters problems similar to the covers. Members added explicitly for fail-safe on a typical aircraft investigated were of the order of 5% to 7%. The additional hidden fail-safe material occurs in spar caps, rib caps, splice material, and hardware.

For these reasons, a completely empirical approach to determination of a fail-safe wing box weight penalty was selected. The factor determined by this approach is as follows:

Bending Material (cover) fail-safe factor (K_{FSCVR}) = 1.261

This parameter when applied to the substructure, however, proved to be insignificant in the regression analysis and was not retained in the final equation.

Dynamics and aeroelasticity. - Aerodynamic forces resulting from the elastic motions of the wing structure are called aeroelastic phenomena. These include such problems as flutter, buffeting and divergence. Wing weight may be penalized by flutter and divergence, as described below.

Flutter: When exciting forces acting on the wing produce vibrations which are at or near the natural frequency of the wing, unstable oscillations of the wing take place. These oscillations, which will cause structural failure of the wing, are referred to as flutter. This phenomena is prevented by increasing the torsional stiffness of the wing box, to insure that flutter critical speeds are well above the operating range of aircraft. Flutter penalties are most likely to occur when combining high speed and high aspect ratio. Empirical relationships for stiffness requirements may be used to determine the weight increment above a strength determined design. Flutter penalty for wing weight estimates is a function of such variables as aspect ratio and limit airspeed.

Divergence: Wing box weight may also be influenced by the necessity of limiting wing deflection to a level which will not allow the development of load divergent conditions. Divergence is a major design factor in unique wing designs such as forward swept wing. The deflection characteristics of the wing box must be controlled by proper placement of material in the box covers and beams. Advance composite construction is most adaptable to these criteria since cover and beam layers may be tailored to obtain the desired elastic properties of the structural elements.

Damage tolerance. - Damage tolerance criteria are defined by Military Specification MIL-A-83444, and are intended to improve structural reliability by protecting safety-of-flight structure from effects of flaws, cracks or damage which may occur during production, and/or service. This is a relatively new specification and little data is available on the weight impact of this requirement. Funded studies of application to the F-14 aircraft (Ref. 2) indicate a sizeable penalty for current technology metallics. If available metallics with better crack growth resistance were incorporated in the design and inspection techniques could determine smaller initial flaws, this penalty could be reduced significantly.

Design to cost. - Wing materials, labor and fabrication technology advancements are significant contributors to the weight cost trades in the design to cost process. Advanced composite materials mixed with high strength metals show promising trends in the weight/cost relationship in the 1980s.

The unit production cost advantage inherent in including advanced materials manifests itself from the interplay of material and labor costs as one material is substituted for another, and also from the iterated effects of reduced weight on overall vehicle size and therefore wing size, weight and cost.

The weight/cost relationship is dependent on customer requirements for a particular vehicle. Weight may be critical on high performance aircraft, justifying a low weight/high cost design. If low cost is the goal, then a high weight/low cost design would be justified.

Actual aircraft designs are usually a compromise between cost and weight. The value of a pound for the vehicle being considered will determine when cost/weight compromises must be implemented. The cost/weight compromises associated with wing box design usually involve type of material, type of construction, fabrication techniques and assembly procedures.

Design Concepts

Wing box description. - That part of the wing which transfers net aerodynamic and inertia loads to the fuselage is referred to as the wing box. It is essentially a box beam which resists these applied wing loads by shear, bending and torsion in the box. In addition, the box supports the control surfaces.

leading and trailing edges, secondary structure and other possible wing-mounted items such as landing gear and engines. Figure 1 illustrates the components which make up a wing box.

Wing box structural concepts. - It is always desirable to design structural components of minimum weight. To determine the lightest structural design the optimum configuration of each alternative construction must be evaluated. Only after the minimum weight design has been determined for each candidate concept is it possible to compare the various forms of construction on a common basis. Final decisions are usually based on economic considerations, durability, serviceability, manufacturing familiarity, availability, etc. and not necessarily on minimum weight.

Multi-Spar design: Multi-spar construction defines a wing box having three or more spanwise beams which support the box covers and transfer shear loads spanwise through the box. Chordwise ribs are placed at end closures, points of load introduction and at intermediate positions as required. For closely spaced beams the number of ribs will be minimal. The spar spacing is determined by geometric and packaging requirements in the wing. The covers may be stiffened sheet where the beam spacing is large, or a flat plate when the beams are closely spaced. Multiple spars may be selected to accommodate packaging requirements such as a large landing gear cutout in a wing box, or thin wings having inadequate depth for flanged stiffeners making it more practical to support the covers by beams connecting the covers. Multiple-spar designs are most advantageous where large shear loads are introduced into the wing box such as at wing fold joints or wing/fuselage connections, (e.g., F-106, F-15, F-16).

Multi-Rib design: Multi-rib constructions define a wing box having closely spaced ribs supporting the covers between beams and transferring shear load to the beams. Generally, there are only two or three beams in this configuration unless local requirements dictate otherwise. Multi-rib construction is usually used on deeper wings where there is adequate stiffener clearance between the covers (e.g., E-2A, 747, DC8, 767 wing). Multi-rib design is well adapted to wing boxes also used as fuel tanks, since the ribs serve as fuel tank bulkheads and baffles.

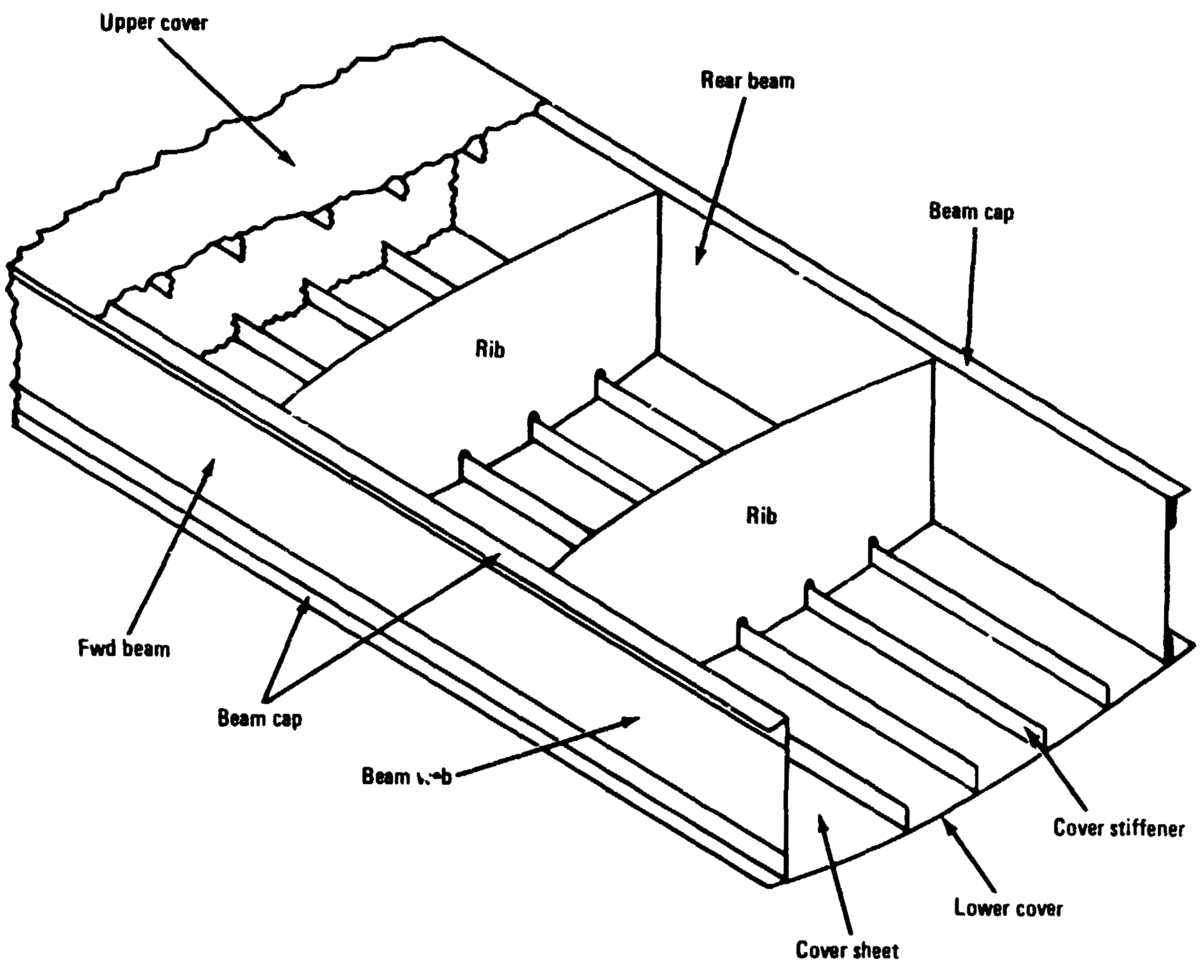

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Figure 1. – Typical wing box showing components.

Stiffened covers must always be used with rib designs. Rib spacing is determined by the column strength required for the stiffeners for compression load in the covers and the ribs are designed to accommodate a combination of local airload, cover crushing loads, fuel pressure loads or local attachment loads.

Full depth honeycomb: Honeycomb construction may be used to replace beams and ribs as cover support (full depth). The full depth concept is particularly useful for very thin wings where assembly space is inadequate for spars or ribs. The major disadvantage of full depth honeycomb is that fuel tank volume is lost from the wing box.

Delta wing design: The structural arrangement for delta planforms are usually a gridwork formed by spars and ribs with rib and spar spacing approximately equal and with covers stiffened in the spanwise direction. The spar locations are dictated by the wing fuselage attachments while the rib spacing is dictated by control surface attachments and a realistic column length for the cover stiffeners. The shuttle wing, F-106, B-58 and SAAB Viggen are good examples of this configuration.

Cover design unstiffened: Unstiffened covers are used with closely spaced spars which provide the only support for the cover material (e.g., F-111 Outer Panel). This arrangement is well adapted to the stiffness critical design of thin wings, since the cover material is totally effective for both torsional stiffness and bending stiffness. The same applies to full depth honeycomb covers; however, the compression strength of the covers is improved since total cover support is provided by the core.

Cover design stiffened: Stiffened covers include the cover sheet and the stiffening elements required for compression stabilization of the cover sheet (or plate). The stiffeners provide stabilization of the sheet for local failure and, in combination with the sheet, provide column strength for the cover (e.g., F-14 Outer Panel). Honeycomb panels are a variation of stiffened covers where two sheets separated by core material provide a stable cover system.

Rib design: Basic rib designs are either truss type or shear web construction. Truss ribs are generally the minimum weight design for thick wings using multirib design (e.g., Shuttle Wing). For thin wings, full shear web ribs are more efficient than trusses, especially when lightning holes are incorporated in the webs. Wing boxes used as fuel tanks requiring sealed compartments and baffles lend themselves well to ribs of the web type design.

Beam design: Basic beam design is very similar to rib design and the comments for ribs apply to beams as well.

Fuel tank considerations: Special considerations must be given to wing boxes used as fuel tanks including rib design mentioned above and fuel pressure loads induced by aircraft maneuvers. Fuel tank sealing, accessability for cleaning and inspection, and control of fuel distribution must also be considered in the wing box design. It is difficult to isolate the total weight penalty for wing fuel since there may e duplicate functions for certain items (e.g., hand holes may be required for wing assembly as well as fuel tank inspection).

Figure 2 illustrates several of the design concepts discussed above.

Construction techniques.

Cover stiffener types: The advantages and disadvantages of common cover stiffeners are listed below.

- Integrally machined stiffeners - Good for fuel tank sealing, but less structurally efficient unless expensive machining processes are used (flanged vs unflanged stiffener).
- Zee stiffened sheet - Easy to manufacture on automated machines and good structural efficiency. Requires sealing of fasteners to sheet for fuel tanks.
- Hat stiffened sheet - Easy to manufacture on automated machines and has good structural efficiency. The additional row of fasteners required over Zee stiffening increases sealing problems and cost of manufacture. The inside of hat stiffeners cannot be inspected easily.
- Y-stiffened sheet - Easy to manufacture on automated machines and has very good structural efficiency. The additional row of fasteners required over hat stiffeners increases cost of assembly. Enclosed area cannot be inspected. Used on F-14 outer panel upper cover.

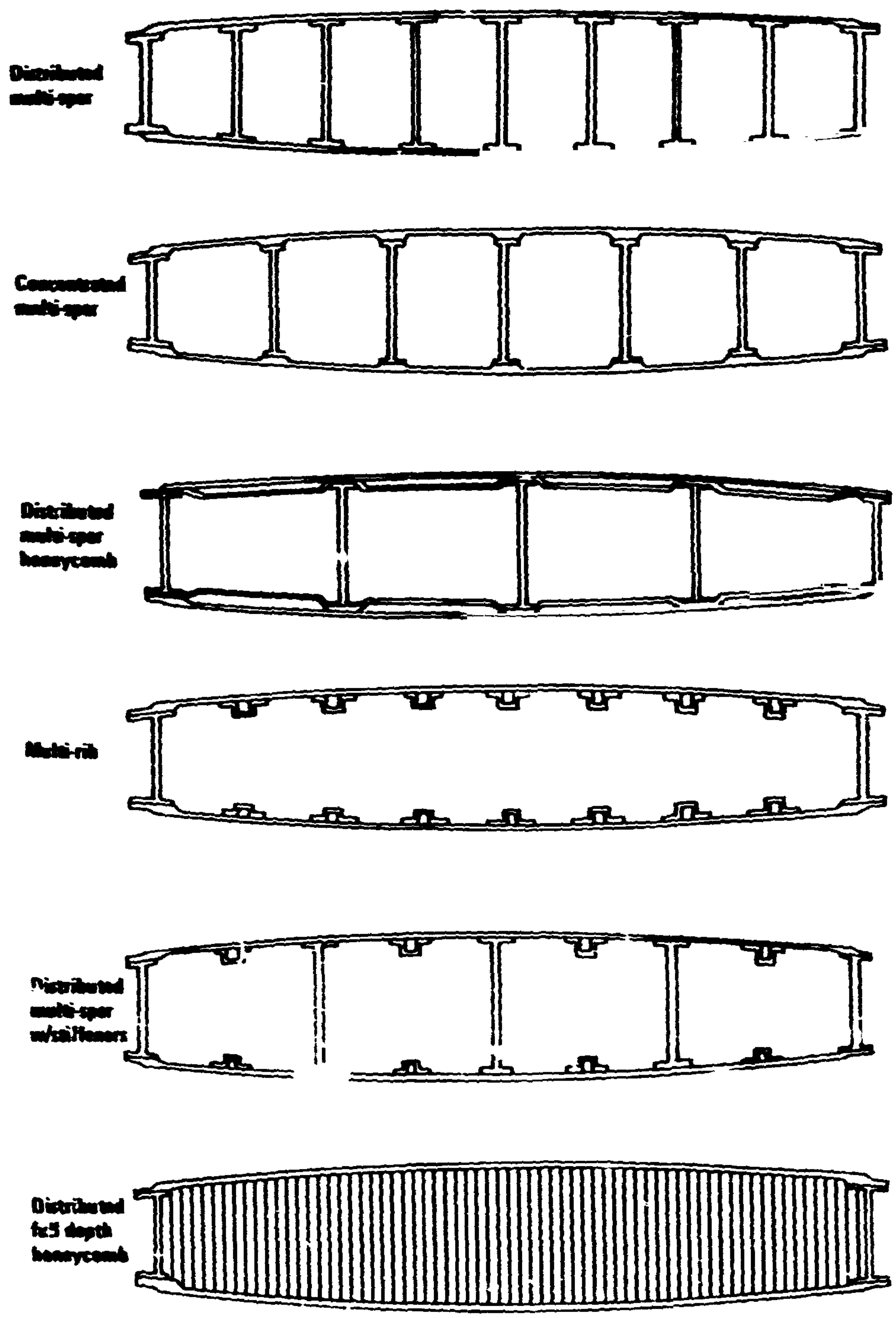

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Figure 2. – Wing box conecpts.

- Honeycomb panels - Good for fuel tank sealing and good structural efficiency for multi-spar designs, where edge material is effective as bending material at panel/spar connection. Expensive to manufacture and difficult to repair.

Other stiffening systems are generally a variation of the types listed above. Figure 3 illustrates the stiffeners discussed above.

Beam and rib construction: Two basic methods of constructing wing box beams and ribs are described below, and illustrated in Figure 4.

- The truss type are made of stable truss members forming cap, post and diagonal components. The caps are usually channel members facilitating connection to the wing covers. The other members may be tubes, channels, cruciforms, or angles depending on load and geometric requirements. Tubes are the most efficient column members for deep trusses where end attachments are not an overpowering weight penalty as may be the case with short members. Trusses are not readily adaptable to the forward and aft beams of the wing box since a closed box is desirable (and necessary in the case of a fuel tank) due to leading and trailing edge functions.

- The web type utilize a full depth web for shear and axial load transfer. Stiffened sheet (integral or separate stiffeners) diagonal tension webs are used extensively since they are a lightweight design which are easily attached to the covers. They are also simple to penetrate for access holes or line runs. Shear resistant designs such as corrugated sheet or honeycomb panels are used in certain applications, particularly for advanced composite design. The honeycomb panels are efficient for fuel tank bulkheads where fuel pressure may be a significant design condition. Wing box fuel tanks dictate some aspects of web type design because of sealing problems. Sealing between caps and covers will establish minimum cap sizes and fastener patterns. Integrally machined web/stiffener combinations eliminate sealing problems in the web itself and minimizes required hardware.

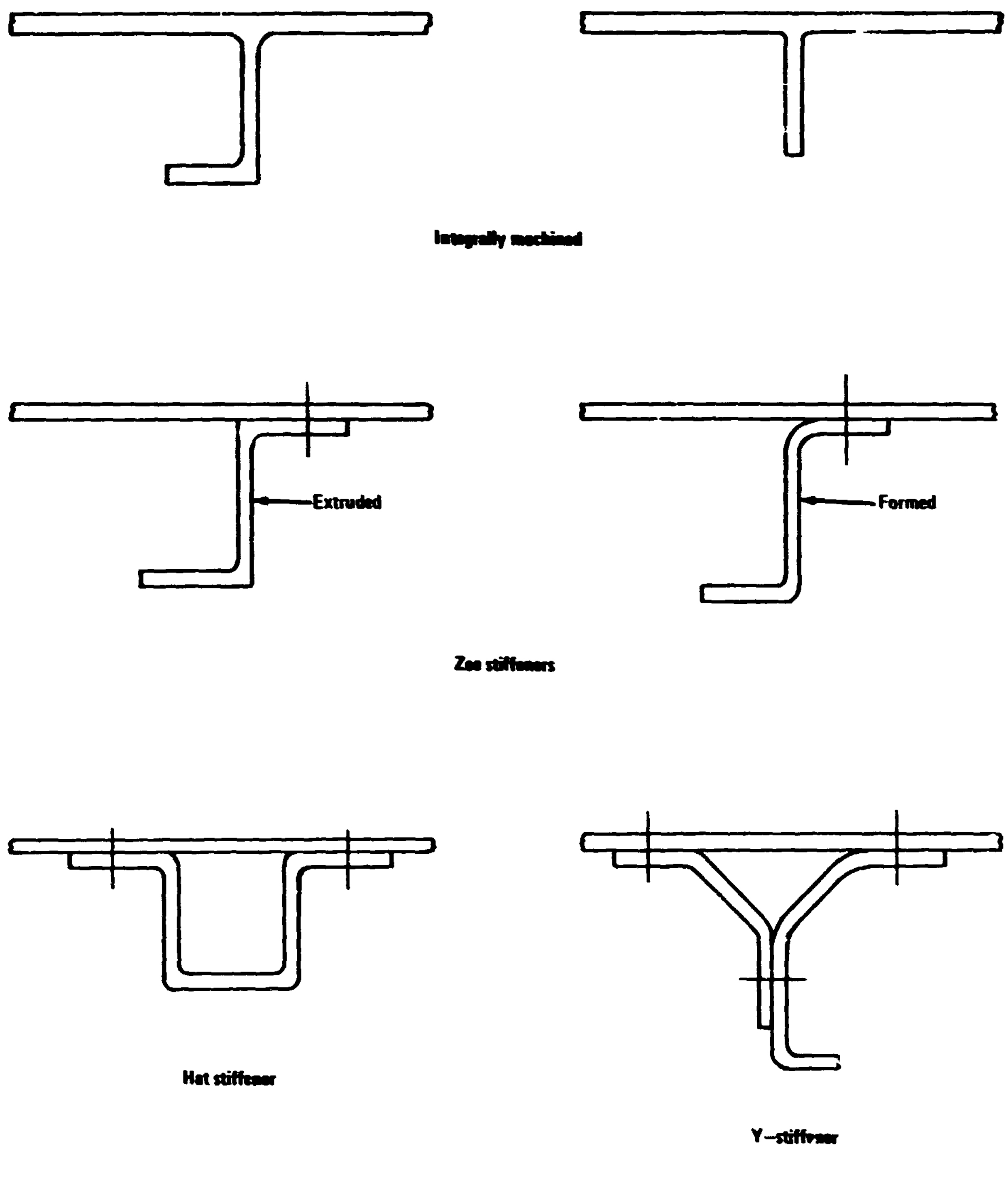
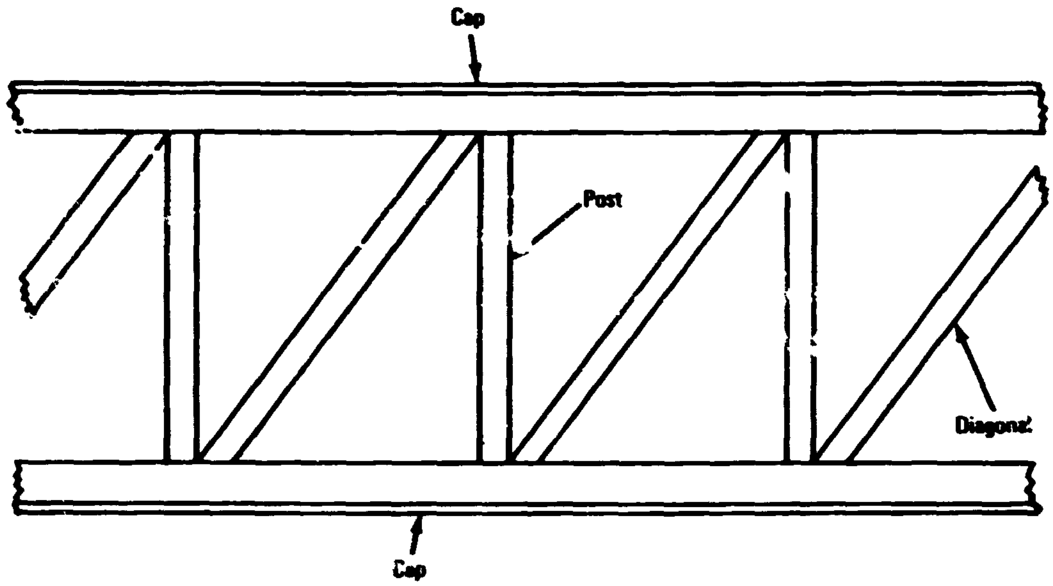
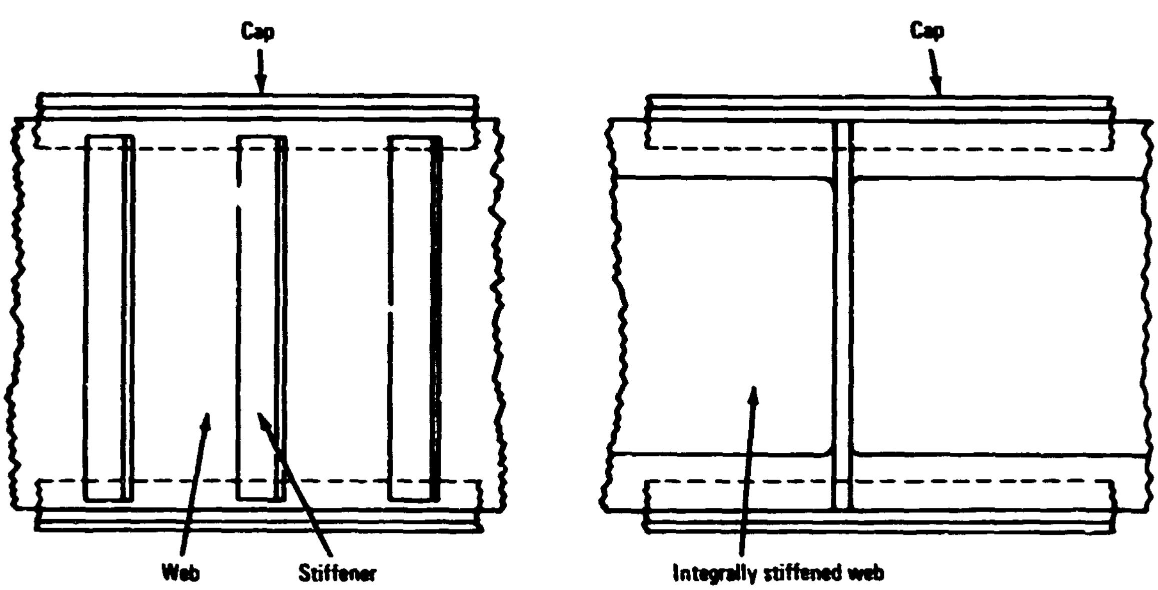


Figure 3. – Typical cover stiffeners.



Typical Truss Design



Typical Web Designs

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Figure 4. – Wing box beam & rib constructions.

Fastener systems: Wing boxes are assembled with a combination of fasteners as outlined below.

- Conventional aluminum fasteners - Used where strength allowables are adequate.
- Hi-lock type fasteners - Used where high strength is required or fuel sealing is required. Steel and titanium fasteners available; however titanium fasteners are costly.
- Interference fit fasteners - Used where high strength or fuel sealing is required. Interference fit fasteners provide improved fatigue allowables and fuel sealing without additional hardware such as O rings and washers. They are available in steel and titanium and are well suited to automated installation. Titanium fasteners provide cost effective weight savings when installed with automated devices. The F-14 and Gulfstream III wing boxes utilize interference fit fasteners in parts of the box assembly.
- Blind fasteners - Used where installation of conventional fasteners is not possible due to access problems. Usually avoided if possible, because they have experienced reliability problems in the past.

<u>Wing carry-thru structure.</u>

Continuous wing box: On aircraft configurations with adequate fuselage volume, wing boxes are extended across the fuselage for a continuous box from tip to tip (e.g., A-4, A-[illegible], A-6, A-7, F-4, F-8 and most transport aircraft). This is most efficient from a structural aspect, since symmetrical spanwise bending loads (a major design factor) do not enter the fuselage structure. Wing fuel capacity is much greater with continuous boxes since the section of greatest depth is within the fuselage confines. The fairings and breather joints associated with the wing/fuselage intersection for this type design are relatively lightweight structures; therefore they do not represent a significant weight penalty.

Integral fuselage carry-thru: On small aircraft with fuselage mounted engines (e.g., F-11, F-105, F-15 and F-16), volume for a carry-thru box is often not available. In such a case the wing box must be attached to fuselage frames at several discrete locations. All loads in the exposed wing must be transferred into the fuselage at these attachments. Structurally this is not an efficient load path because:

- Loads must be carried thru the fuselage by frame bending, a less efficient method than a box beam bending
- Shear lag problems in the wing box at the fuselage attachment add a weight penalty to the exposed wing.

There are requirements for breather joints with this design, but the amount of fairing structure will be small.

Special features.

Wing fold: Wing folding is a requirement for storage of many carrier based aircraft. A sizeable weight penalty can result from folding mechanism, hinge and latch fitting and load path discontinuities in the wing box.

Variable geometry: Variable geometry increases the weight of a given configuration, but the penalty to the overall vehicle may be negated by the improved performance. The penalties resulting from sweep actuation mechanism, pivot structure and load path discontinuities must be incorporated into wing weight estimates. These penalties are best evaluated by empirical methods. In the case of the F-14, the penalty for the pivoting wing is partially offset by eliminating the need for a wing fold.

Landing gear, engine and store mounting: Mounting these items on a wing box involves the addition of ribs and/or beams, increased strength of local structure and addition of attachment fittings. In addition landing gear storage frequently requires cutouts in the main box structure (e.g., the F-5 wing). This causes discontinuities in the primary load carrying structure requiring increased strength in the remaining structure. These penalties are best evaluated by empirical methods also.

Winglets: For modifications to existing wings, winglets attached to the wing tips have proven effective in improving wing performance without major structural modification. Since use of winglets is a relatively new development, weight penalties to the wing box presented here are based entirely on the Gulfstream III aircraft, a modification of the Gulfstream II.

Manufacturing Methods

Material preparation.

Formed sheet metal: One of the common methods of manufacturing aircraft wings utilizes flat sheet stabilized by formed sheetmetal members. This method is used extensively on lightly loaded wings which do not serve as fuel tanks (e.g., outer panel of the E-2A). The constant thickness of the sheets and the difficulty in forming thicker material make this technique less practical for more highly loaded wing covers and beams; however lightly loaded ribs of multi-rib design are readily adaptable to this construction.

Extruded sections: Stiffening members of varying shapes may be extuded when thicker members are required. They may be tailored to strength and stiffness requirements by machining operations after extruding. Extruded parts may be manufactured in a variety of shapes and sizes including parts as large as the wing cover planks used on the C-5A. For wing box construction they are limited to aluminum alloys.

Machined parts: Machined parts cover the range from small machined fittings to major sections of wing covers, beams or ribs. Machining operations are used to fabricate parts requiring specific geometric shapes and to remove excess weight. Machined skins for wing covers or beams allow tailoring for strength and stiffness requirements. Stiffeners machined integrally with webs eliminate a sizeable amount of sealing hardware, but are usually less efficient structurally than separate stiffeners attached to machined skins. Major attachment fittings are manufactured by machining operations because of their complex shapes and strength requirements.

Chem-milled parts: Chemical removal of unneeded material is used extensively in wing box design as a weight saving effort. It is used for beam and rib web fabrication where panel thicknesses may be varied over the surface of the web. It is also used for structure where countersunk fastener requirements dictate the depth of the basic sheet, but strength requirements allow thinner panels between rows of fasteners. Chem-milling is a very cost-effective method of minimizing weight and is used on all but the very low cost designs.

Forgings: The process of manufacturing net or near net shapes by forming the metal under pressure is known as forging. Complex shapes may be fabricated at lower cost by eliminating much of the material that otherwise would require removal by machining operations. Ribs or bulkheads with an integral gridwork of stiffeners are particularly adaptable to forging before machining for final thicknesses. Small parts may be forged to the final dimensions as so called no draft forgings. This is primarily a cost saving process and has little impact on weight.

Advanced composite processing/manufacture: Several basic techniques are available for manufacture of organic matrix composites. Among these are press molding, vacuum bag molding and autoclave molding. The basic process is to apply heat and pressure to the starting material (prepreg) to compact the laminae, remove entrapped air and cure the matrix. To date, both in the industry and at Grumman the primary manufacturing process is autoclave molding. Since the process allows molding of several parts (as defined in metal construction) in a one step operation, assembly costs are lower than for comparable sheet metal construction.

Grumman specifications that control the processing of Boron/Epoxy composites, approved by the Navy, are:

- GSS 11200 - Boron/Epoxy Composite Parts Fabrication
- SP-A51-CS-1B - Fabrication, Assembly and Testing of the Horizontal Stabilizer Box Beam

The SP specification is specific for manufacture of the F-14 horizontal stabilizer only. Such special processing specifications are generally required for primary structures of advanced designs using advanced materials.

Several advanced composite developed processing specifications utilized in development programs are:

- SP-G-011 - Processing of Boron/Epoxy Sandwich Structures
- Gr-100A - Processing of LHS Graphite/Epoxy
- H-100A - Processing of Boron/Graphite/Epoxy Hybrids
- T-100 - Processing for Tubular Members Using Advanced Composites

Protection systems: Corrosion protection requires surface treatment of structural parts before assembly in the aircraft. This may be a treatment such as alodine coating or anodizing, or use of clad aluminum on exposed surfaces.

Fuel tank interiors may be treated with additional sealing and protection systems. Protection of the structure is standard procedure on all aircraft and is not to be considered as a penalty over the empirical data base.

Assembly techniques.

Machined assemblies: Assembly of machined parts may utilize mechanical systems or welded connections of steel and titanium parts. Fewer individual parts are involved in machined assemblies thereby reducing assembly hardware weight. Electron beam welding where applicable (such as the F-14 wing center section) provides a strong efficient assembly with a minimum of mechanical fasteners. Handling problems may determine the maximum size of a machined part used in an assembly (e.g., machined wing cover planks must be a reasonable length for handling after machining).

Built-up sheet metal: Sheet metal construction utilizes mechanical fastening systems to assemble individual parts into major components and subassemblies. The numerous parts and associated assembly hardware are weight and cost inefficiencies. Recent development work in sheet metal is aimed at reducing the number of individual parts. Super plastic formed diffusion bonded titanium assemblies are being developed along these lines with cost and weight savings as the major goals. This method forms stiffening elements integral with basic sheet and connects elements by diffusion bonding. The Air Force has funded development studies of this technique, to Grumman and North American (Aft Fuselages) and Boeing and McDonald Douglas (Wing Center Sections), under the title of Built-up Low-cost Advanced Titanium Structures (BLATS).

Honeycomb assemblies: Full depth honeycomb structure is assembled by bonding covers, beams and ribs to a basic core assembly for an efficient structural component. Honeycomb panel structure must be attached to adjacent structure by mechanical fasteners. This can be a considerable weight penalty, especially on smaller panels where a large part of the panel is affected by the fastener patterns.

Advanced composite assemblies: Advanced composite assemblies may be integrated into a few layup and curing processes, thus eliminating many of the separate assembly steps required for other materials. This enhances the weight saving benefits of composites by reducing required assembly hardware. This benefit is considered in the total advanced composite weight savings utilized for weight estimates.

Weight/cost trades. - The best manufacturing methods for a particular wing design can only be determined by conducting cost versus weight trade studies. The so called "value of a pound" for the vehicle in question must be established as a guide for these studies. Compromises must be accepted to keep the cost and weight within reasonable constraints. The value of a pound may be high on vehicles required to meet high performance standards or particular missions such as the space shuttle. For early weight estimates. factors may be used to reflect the relative importance of cost and weight and applied to the weight estimating relationships.

Recent experience with new designs indicate that low cost does not necessarily mean higher weight. For example, a manufacturing procedure which uses fewer individual parts will probably be the lightest practical design. Many companies have design to cost manuals which serve as a guide in selecting the most cost/weight effective methods of manufacturing aircraft components. These manuals are usually of a proprietary nature, and therefore could not be included in this study.

THEORETICAL WING WEIGHT EQUATIONS

Many wing weight equations have been proposed. Some have been strictly theoretical, based on simple beam theory or more elaborate models, sometimes modified by experience or other factors for a particular case (for example, Ref. 3, 4, 5). Others have been almost entirely empirical, relying on a regression analysis of parameters known or assumed to be important (Ref. 6). The synthesis of these approaches has yielded the most useful wing estimating equations (Ref. 7, 8) for preliminary design studies. These methods rely on a rational, though certainly simplified, model for (at least) bending material and determination of constants, coefficients and exponents by a regression or similar analysis to include non-theoretical influences on the box beam weight. Such influences as non-optimum weight, minimum gages and secondary loads, and other design requirements can be accounted for by such empirical adjustments to a theoretical equation. It is considered desirable to have single weight estimation methodology for all types of aircraft. Identification of factors that separate "fighter," "commercial," "general aviation," and the like are usually a means of grouping design philosophies, methods of construction, etc, without identifying them explicitly. This study completely avoids this approach and attempts to identify the underlying physical discriminators so that the same equation can be used to include variable sweep high performance fighters, utility light aircraft, and the spectrum of aircraft in between.

Derivation of Theoretical Equation for Wing Box Cover (Bending) Material Weight

A straightforward beam model was selected as a basis for the empirically corrected wing weight equations. This approach is not preferred in order to produce the most accurate wing weight prediction equation but instead to provide a theoretical basis for improvement by regression analysis. It is intentionally restricted to an elementary format to keep the method compatible with the preliminary design phase. The derivation of the wing box cover (bending) material weight is shown in Figure 5.

DEFINE THE CENTER OF PRESSURE FOR A UNIFORMLY LOADED WING AS THE CENTROID OF AREA

$$y = \frac{(2b/2)\,(C_R + 2C_T)}{3\,(C_R + C_T)} \qquad (1)$$

CALCULATE THE ACTUAL CENTER OF PRESSURE LOCATION ALONG THE STRUCTURAL AXIS (40% CHORD)

$$y' = \frac{2b}{\cos\Lambda}\,\frac{(C_R + 2C_T)}{(C_R + C_T)} \qquad (2)$$

THE AIRLOAD ON WING IS

$$V = (B)n \qquad (3)$$

THE BENDING MOMENT AT THE ROOT IS

$$M = \frac{V}{2} \times y' = \frac{2b}{\cos\Lambda}\,\frac{(C_R + 2C_T)}{(C_R + C_T)}\,\frac{(B)n}{2} \qquad (4)$$

BECAUSE THIS IS INTENDED AS A BASIS FOR AN EMPIRICAL METHOD, AN ARBITRARY (BUT REASONABLE) SPAN STATION MAY BE SELECTED FOR ANALYSIS. A STATION 2/3 IN FROM TIP CAN BE TAKEN AS REPRESENTATIVE FOR SCALING, RESULTING IN A BENDING MOVEMENT OF APPROXIMATELY 1/2 THE ROOT BENDING MOMENT BASED ON A PARABOLIC BENDING MOMENT CURVE

$$M_{2/3} = \frac{b}{2\cos\Lambda}\,\frac{(C_R + 2C_T)}{(C_R + C_T)}\,(B)n \qquad (5)$$

THE COVER LOAD 2/3 IN FROM TIP IS

$$P_{2/3} = M_{2/3}/T_{2/3} = \frac{3b}{2\cos\Lambda}\,\frac{(C_R + 2C_T)}{(C_R + C_T)}\,\frac{(B)n}{(2T_R + T_T)} \qquad (6)$$

THE RUNNING LOAD 2/3 IN FROM TIP IS

$$N_{x2/3} = P_{2/3}/C_{2/3} = \frac{3b}{2\cos^2\Lambda}\,\frac{(C_R + 2C_T)}{(C_R + C_T)}\,\frac{(B)n}{(2T_R + T_T)}\,\frac{3}{(2C_R + C_T)} \qquad (7)$$

ASSUME THAT THE BOX WIDTH IS 1/2 THE CHORD

CORRECT THE AVG CHORD $\left(\frac{2C_R + C_T}{3}\right)$ TO AN APPROXIMATE BOX WIDTH $\left(\frac{2C_R + C_T}{3 \times 2}\right)$

$$N_{x2/3} = \frac{3b}{2\cos^2\Lambda}\,\frac{(C_R + 2C_T)}{(C_R + C_T)}\,\frac{(B)n}{(2T_R + T_T)}\,\frac{6}{(2C_R + C_T)} \qquad (8)$$

THE BENDING (COVER) WEIGHT IS

$$W_{CVR} = \rho\, t_{COVER}\, S_{BOX} \qquad \text{WHERE } S_{BOX} = S_w/2$$

$$t_c = N_{x2/3}/F$$

$$W_{CVR} = \rho\, N_{x2/3}/F\, S_w$$

$$W_{CVR} = \frac{N_{x2/3}}{F/\rho}\, S_w$$

$$W_{CVR} = \frac{9}{2}\,\frac{b}{\cos^2\Lambda}\,\frac{(C_R + 2C_T)}{(C_R + C_T)}\,\frac{(B)n}{(2T_R + T_T)}\,\frac{S_w}{(2C_R + C_T)}\,\frac{1}{(F/\rho)} \qquad (9)$$

FOR THE DATA BASE, IT WAS ASSUMED THAT F/ρ DID NOT VARY WIDELY AND COULD ACCORDINGLY BE INCORPORATED INTO A CONSTANT

$$W_{CVR} = C_1\left[\frac{b}{\cos^2\Lambda}\,\frac{(C_R + 2C_T)}{(C_R + C_T)}\,\frac{(B)n}{(2T_R + T_T)}\,\frac{S_w}{(2C_R + C_T)}\right] \qquad (10)$$

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Figure 5. – Derivation of wing box cover (bending) material weight.

Derivation of Theoretical Equation for Wing Box Substructure (Shear) Material Weight

The derivation of a simple expression for substructure (shear) material weight is presented in Figure 6. This relationship is based only on a representation of shear loading in beams; no attempt is made at this stage to account for ribs. Because the correlation will be done using weights taken from actual weight reports where airload designed ribs are not ordinarily differentiated from load distribution and closure ribs, a theoretical expression based on airloads can not be expected to yield a good correlation. Therefore, a separate expression for ribs is not included in this report.

Linear Regression Theory

The statistical correlations were obtained by utilizing the "least squares" method of linear regression. The method of least squares develops a criterion that says the regression line should minimize the sum of the squares of the difference between the actual and calculated points.

If the equation were $y_{CALC.} = a + bx$, the attempt would be to minimize $\Sigma\,(y_{ACT.} - y_{CALC.})^2$. The solution for a and b is provided by the following set of "normal" equations:

$$\Sigma y_{ACT.} = na + b\Sigma x$$

$$\Sigma xy = a\Sigma x + b\Sigma x^2$$

where n is the number of data points. The problem with the least squares approach occurs when the dependant variables in the data set vary over a large range. For example, if the dependant variable were TOGW and the data varied from 100 to 1 million pounds, the least squares approach would attempt to minimize the latter at the expense of the former. The solution is to divide each item by its respective dependant variable such that $\Sigma\,(1 - y_{CALC.}/y_{ACT.})^2$ is really minimized. The x and y terms in the normal equations would be appropriately modified. The normal equations can be expanded to include more independeant variables. The equations for the case of two independant variables ($y_{CALC.} = a + bx + cx$) are:

$$\Sigma y_{ACT.} = na + b\Sigma x + c\Sigma z$$

$$\Sigma xy = a\Sigma x + b\Sigma x^2 + c\Sigma xz$$

$$\Sigma zy = a\Sigma z + b\Sigma xz + c\Sigma z$$

THE AIRLOAD ON WING IS:

$$V = (B)\, n \qquad (3)$$

THE SHEAR AT A POINT 1/2 OF THE WAY FROM THE TIP TO THE ROOT IS APPROXIMATELY

$$V_{1/2} = 0.5\, (B/2) n \qquad (11)$$

THE SHEAR FLOW 1/2 IN FROM TIP IS:

$$q_{1/2} = \frac{V_{1/2}}{(T_R + T_T)\, N_B} \qquad (12)$$

$$q_{1/2} = \frac{0.5(B)\, n}{(T_R + T_T)\, N_B} \qquad (13)$$

THE SHEAR MATERIAL WEIGHT IS

$$W_{SUB} = \rho\, t\, S_{BEAM}$$

$$t = q_{1/2}/F_s$$

$$W_{SUB} = \rho\, q_{1/2}/F_s\, S_{BEAM}$$

$$W_{SUB} = \left(\frac{1}{(F_s/\rho)}\right)\left[\frac{0.5(B)\, n}{(T_R + T_T)\, N_B}\right]\left[\frac{(T_R + T_T)\, b\, N_B}{2}\right] \qquad (14)$$

$$W_{SUB} = \left(\frac{1}{(F_s/\rho)}\right)[0.25(B)\, n\, b] \qquad (15)$$

AS IN THE CASE OF THE BENDING MATERIAL, THE FACTOR F_s/ρ IS ASSUMED NOT TO VARY WIDELY AND IS INCORPORATED INTO A CONSTANT

$$W_{SUB} = C_2\, [(B)\, n\, b] \qquad (16)$$

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Figure 6. – Derivation of wing box substructure (shear) material weight.

Another form of equation used is $y_{CALC.} = a\,x^b\,z^c$. In order to use the method of least squares, the form of the equation is modified to more closely match the "linear" equation. This is done by taking the natural logarithm of both sides of the equation:

$$\ln y_{CALC.} = \ln a + b \ln x + c \ln z$$

The normal equations can now be used to solve for ln a, b, and c. Note that in this case $(\ln y_{ACT.} - \ln y_{CALC.})^2$ is minimized which does not create the same problem as would have occured if $(y_{ACT.} - y_{CALC.})^2$ were minimized in the previous example.

<u>Statistical Correlation of Wing Box Weight</u>

The actual weights used in the regression analysis were arrived at by taking the actual weights from weight reports and subtracting out recognized penalties (i.e., fuel, engine, stores, landing gear, fold, and sweep). Weight penalties not available as coded or implicit structural increments in the weight reports were calculated by Grumman's own methods. The cover weights comprised items actually coded to covers (skin, stiffeners, beam caps, jsf-joints, splices, and fasteners), and the substructure weights, items actually coded to beams and ribs (beam webs, beam caps if integral with webs, beam jsf, ribs, bulkheads, chordwise stiffeners, and rib jsf). Total box beam weights are the summation of the actual cover plus substructure weights less their respective penalties. Wing box design data is tabulated in Appendix A.

The main objective in deriving the equations for the regression analysis was to adhere as closely as possible to the analytical approach derived in Figures 5 and 6. The emphasis was on using parameters to improve the theory rather than improve the "fit" of the regression analysis.

The theory for cover (bending) weight derived in Figure 5 Equation (10),

$$W_{CVR} = C_1 \left[\frac{b \quad (C_R + 2C_T) \quad B\,n\,S_W}{\cos^2 \Lambda \; (C_R + C_T)\;(2T_R + T_T)\;(2C_R + C_T)} \right]$$

was correlated in the regression analysis along with various other parameters in an attempt to account for minimum gage, non-optimum factor and combined bending and shear. The addition of an area (S_{BOX}) term to compliment the theory was based on:

- S_{BOX} better reflects the internal load distribution on the entire wing span (i.e., the derivation is for only one spanwise location)
- S_{BOX} better reflects cover weights if influenced by minimum gage.

The following equation for cover weight was derived:

$$W_{CVR} = 0.081223 \left[\frac{b \quad (C_R + 2\,C_T) \quad B\,n\,S_W}{\cos^2 \Lambda (C_R + C_T)\ (2T_R + T_T)\ (2C_R + C_T)} \right]^{0.5479} \left[S_{BOX} \right]^{0.4897} \tag{17}$$

and resulted in a percent standard deviation of 19.5%

The theory for substructure (shear) weight derived in Figure 6, Equation (16),

$$W_{SUB} = C_2 \left[B\,n\,b \right]$$

was correlated in the regression analysis along with various other parameters in an attempt to account for minimum gages, non-optimum factors and secondary loads in the substructure.

In the regression analysis the addition of a chord term (or S_W in lieu of b), in the theoretical equation, greatly improved the accuracy. Although the emphasis is on improving the theory rather than the "fit" of the regression analysis, we felt justified in adding this parameter since as explained previously, the theory for the substructure was derived for beams only with the ribs being ignored. Adding the chord term would seem to help in accounting for the rib weights. The addition of a volume (S_{BOX} (T_R+T_T)) term better reflects the number of beams and ribs as well as beam & rib weights when influenced by minimum gage.

The following equation for substructure was derived:

$$W_{SUB} = 0.00636 \left[B\,n\,S_W \right]^{0.5614} \left[S_{BOX} (T_R + T_T) \right]^{0.144} \tag{18}$$

and resulted in a percent standard deviation of 30.6%.

The standard deviation for the basic box (covers and substructure combined) was 17.4%.

Correlation plots of actual versus calculated for cover weight, substructure weight and basic box weight, are shown in Figures 7, 8 and 9 respectively.

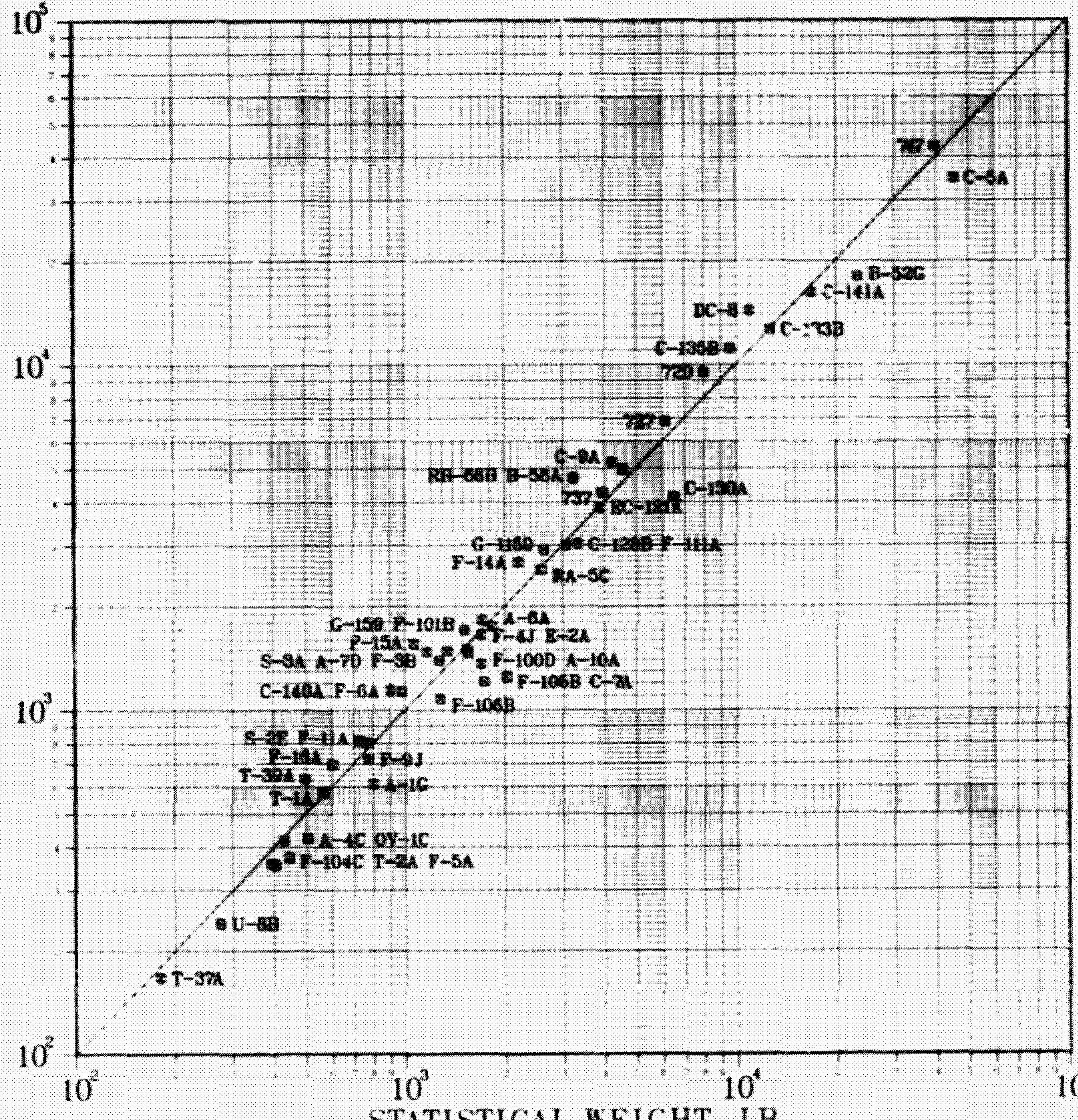

Figure 7. Correlation of basic box cover weight estimate (original equation).

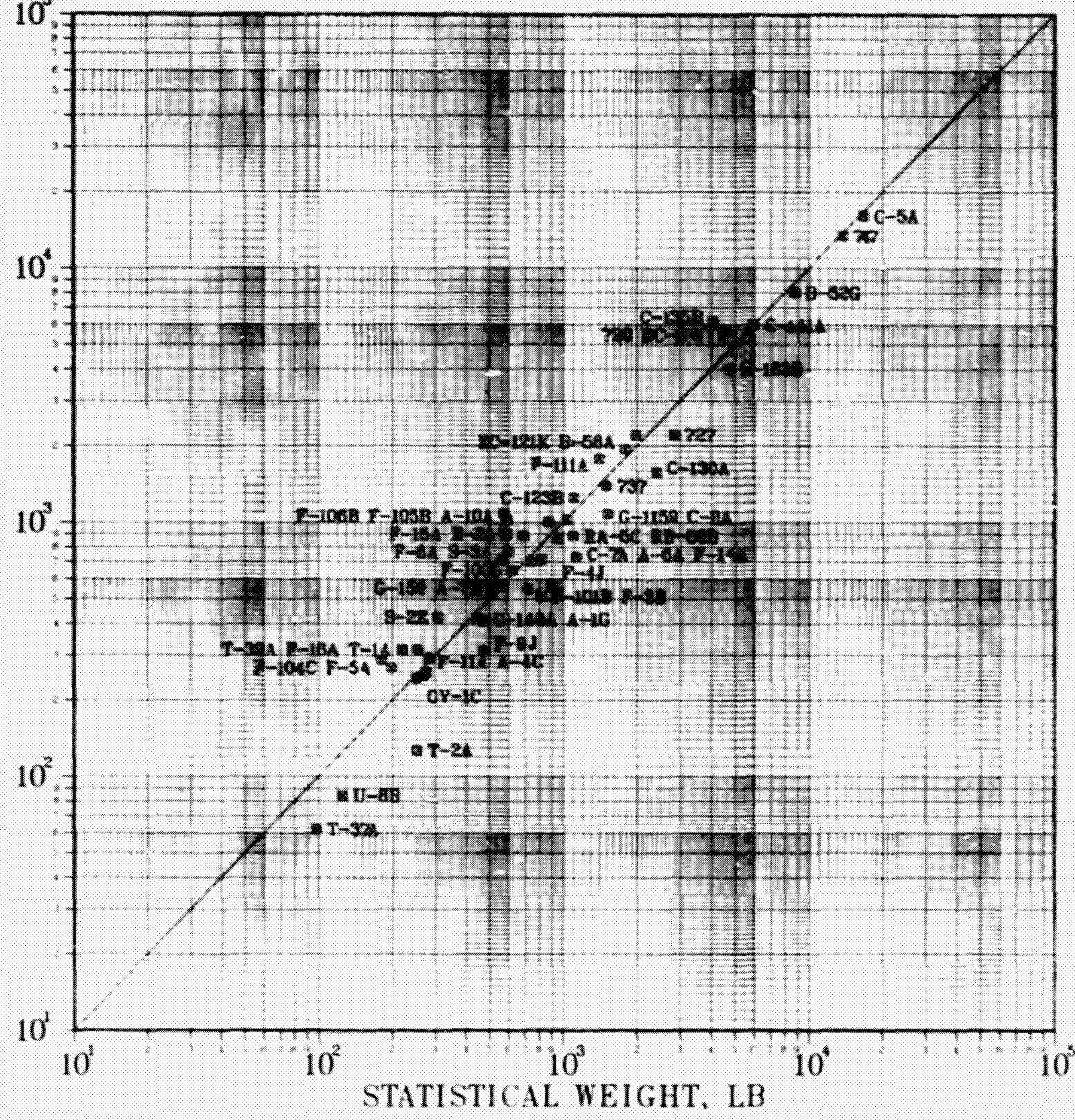

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Figure 8. Correlation of basic box substructure weight estimate (original equation).

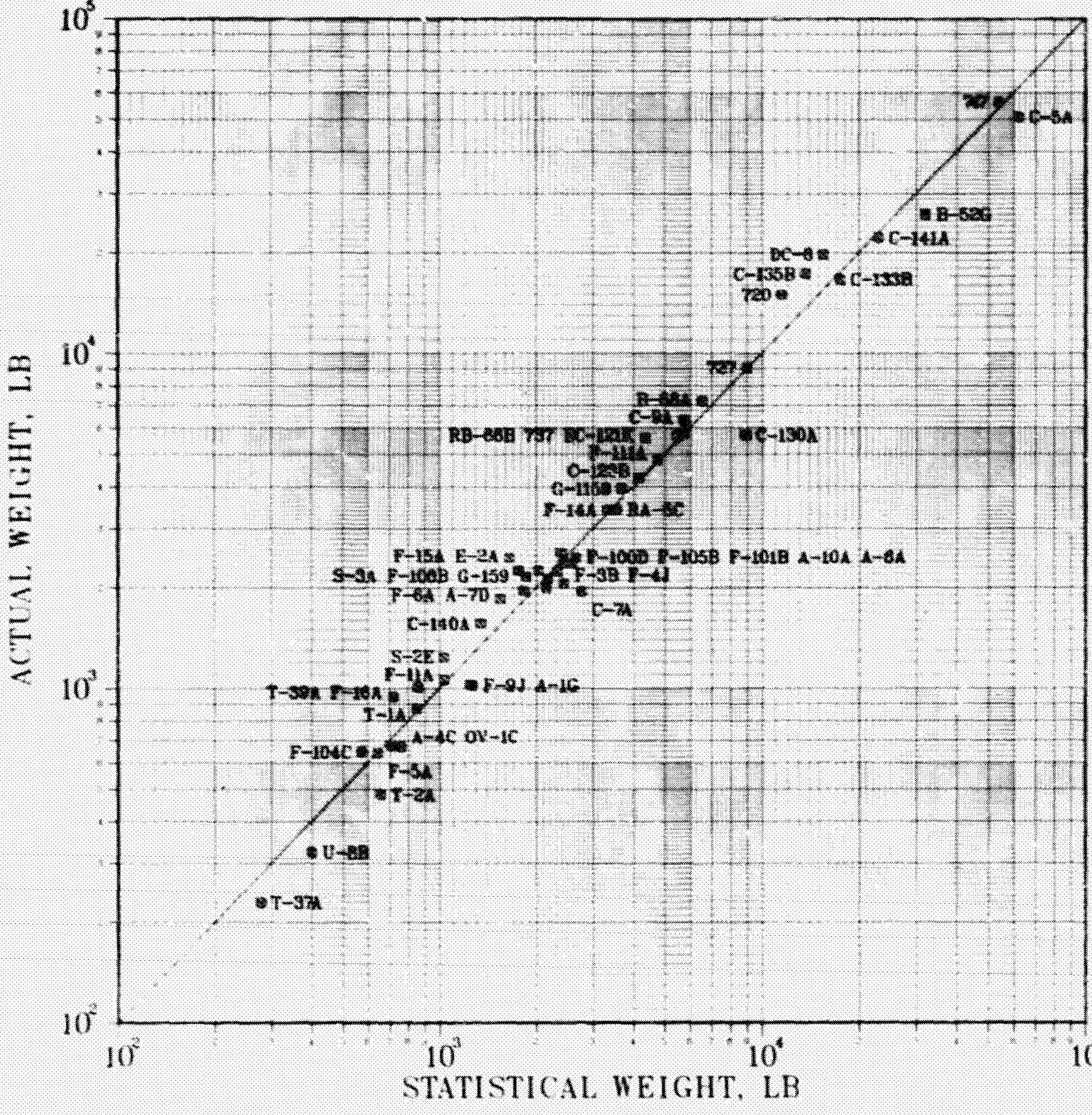

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Figure 9. Correlation of total basic box weight estimate (original equation).

DEVELOPMENT AND INTEGRATION OF FACTORS

The primary objective of this study is to obtain and integrate correction factors for the empirical equation that reflect the influence of various materials, types of construction and broad design philosophies. The equations developed in the previous section will be enhanced by incorporating these factors.

Wing Box Cover Weight

The cover weight obtained from the equation derived in the section on Theoretical Wing Weight Equations is influenced only by external loading. The effects of material, construction and design will be incorporated with additional factors while maintaining the basic theoretical approach.

Fail-Safe design. - A completely empirical approach was selected to determine a fail-safe factor (Refer to the paragraph on Applied Loads). Ten (C-9A, C-135B, C-140A, DC-8, 720, 727, 737, 747, G-159, and G-1159) of the fifty data base aircraft were assumed to have a fail-safe weight penalty. The factor determined by this approach (see Equation (19)) is as follows:

$$\text{Fail-safe factor } (K_{FSCVR}) = 1.261$$

For the various combinations of parameters that were screened during the study, K_{FSCVR} varied between 1.24 and 1.30.

Flutter. - Flutter penalties are most likely to occur when combining high speed and high aspect ratio (Refer to Dynamics and aeroelasticity). The effects of flutter on cover weight is represented by inclusion of the parameter limit airspeed (V_L) into Equation (19).

Carry-Thru design. - A carry-thru factor (K_{CT}) is used to denote whether the wing box continues through the fuselage or attaches to the side-of-body. This parameter was used in the regression but was found to be insignificant and was not retained in the final equation for cover weight. The implications are that there is no additional weight penalty to the covers for wings with no carry-thru. However, even though there is no discrete weight penalty due to K_{CT}, the fact that exposed values are used for B, b, S_W, C_R and S_{BOX} in the regression and are contained in the cover equation means there could be an inherent weight penalty.

Materials and constructions. - Several methods for obtaining material/construction factors were investigated. The emphasis was placed on developing factors that would be an extension of the simple analytical approach used in deriving the cover (bending) material weight in Theoretical Wing Weight Equations. This approach is outlined below.

1. Gather data, (i.e., type of alloy, stiffener spacing, rib spacing, beam spacing, construction type and design philosophy) for the data base airplanes. Complete details were obtained for 22 of the 50 airplanes and partial data was acquired for 7 airplanes; this data is tabulated in Appendix B.

2. Develop material/construction factors for one type of alloy in aluminum, titanium and steel and also for advanced composite (graphite-epoxy). This was accomplished by using a wing multi-station analysis computer program (Ref 9.) on a representative wing (A-6A) and varying required parameters. Factors were obtained for load factor (N_{BOX}) versus construction type/rib or spar spacing for the upper and lower covers. Of 65 wing box elements (i.e., wing outer panel, wing center section and wing substructure) for which we were able to identify the alloy used, 49 were 7XXX series aluminum. Accordingly, when faced with the selection of a "reference material", we chose 7075-T6 (room temperature) 'Z' stiffened with a rib spacing of 12 in. Appendix C shows the selection as having a factor of 1.000; all other factors shown in Appendix C were then computed relative to the baseline.

3. Develop algorithms that would allow factors to be obtained for other alloys of aluminum, titanium and steel. Obtaining factors through use of the multi-station analysis for every alloy would be a monumental task and would also not allow for future alloys to be considered. An alternate approach would be to develop algorithms for these factors as a function of material properties (i.e., compressive yield stress (F_{CY}), ultimate tensile stress (F_{TU}) and density). Though this appeared a worthwhile step, time did not permit this to be pursued as part of this study.

4. Inclusion of this material/construction factor (K_{MTLCVR}) in the regression analysis, to normalize the data base to 7075-T6 aluminum. This factor was considered to be a key parameter but could not be included in the regression analysis as data was only available on 22 of the 50 data base airplanes. A summary of materials and constructions for the 22 airplanes is shown in Table 3.

Temperature effects. - A factor ($K_{TEMPCVR}$) to account for the effects of temperature was generated utilizing a wing multiple station analysis program and the factors are shown in Appendix D for various temperatures and materials.

Wing Box Substructure Weight

The substructure weight is defined only partially by the first order theoretical equations derived in Figure 6. The parameters added to the equation in Statistical Correlation of Wing Box Weight account for secondary effects. The effects of material and design will be added by incorporating the following factors.

Fail-Safe design. - A completely empirical approach was taken, as in the covers, to determine a factor for fail-safe. The factor was determined, through the regression analysis, to be insignificant and was not retained in Equation (20).

Carry-Thru design. - The carry-thru factor (K_{CT}) obtained for the substructure is applied only to an exposed wing. The use of exposed wing area in the equation compensates for reduced box area. The factor, however, is required to account for the effects of cover loads at the side of body being transferred into the spars which connect directly to fuselage frames. This results in a significant substructure weight increment above the substructure weight for a wing with a straight through wing box.

Materials & constructions. - The lack of material information available on substructure did not allow a detailed method to be pursued in this study. The factor (K_{MTLSUB}) will only distinguish between an aluminum and titanium substructure based on the following:

	Aluminum	Titanium
Shear Allowable (F_S)	24,200 psi	50,000 psi
Density (ρ)	0.101 pci	0.164 pci

TABLE 3. – MATERIAL TYPE AND CONSTRUCTION

AIRCRAFT	CENTER SECTION				OUTER PANEL			
	UPPER		LOWER		UPPER		LOWER	
	CONSTRUCTION	MATERIAL	CONSTRUCTION	MATERIAL	CONSTRUCTION	MATERIAL	CONSTRUCTION	MATERIAL
2 A-4C	'Z'	7075-T6	'Z'	7075-T6	'Z'	7075-T6	'Z'	7075-T6
3 A-6A	INTEG	7079-T651	INTEG	7079-T651	INTEG	7075-T651	INTEG	7075-T651
4 A-7D	THICK SKIN	7079-T651	THICK SKIN	7075-T651	THICK SKIN	7079-T651	THICK SKIN	7075-T651
5 A-10A	INTEG	7075-T651	INTEG	2024 T351	'Z'	7075-T6	'Z'	2024-T3
22 727	'Z'	7178-T651	'Z'	2024 T351	'Z'	7178-T651	'Z'	2024-T351
23 737	'Z'	7075-T6	'Z'	2024-T351	'Z'	7075-T6	'Z'	2024-T351
24 747	'Z'	7075-T6	'Z'	2024-T351	'Z'	7075-T6	'Z'	2024-T351
25 G-159	INTEG	7075-T6	INTEG	7075-T6	INTEG	7075-T6	INTEG	7075-T6
23 G-1159	INTEG	7075-T651	INTEG	2024-T351	INTEG	7075-T651	INTEG	2024-T351
27 E-2A	INTEG	7075-T651	INTEG	7075-T651	'Z'	7075-T6	'Z'	7075-T6
29 F-4J	THICK SKIN	7075-T651	THICK SKIN	7075-T651	THICK SKIN	7178-T651	THICK SKIN	7178-T651
33 F-11A	CODED TO BODY	-	-	–	THICK SKIN	755-T6	THICK SKIN	755-T6
34 F-14A	INTEG	6-4	INTEG	6-4	'Y'	6-6-2	'Z'	6-4
35 F-15A	CODED TO BODY	–	–	–	INTEG	2024-T851	INTEG	6-4
36 F-16A	CODED TO BODY	–	–	–	THICK SKIN	2124-T851	THICK SKIN	7475-T73
40 F-105B	CODED TO BODY	–	–	–	THICK SKIN	7075-T6	THICK SKIN	7075-T6
41 F-106B	CODED TO BODY	–	–	–	THICK SKIN	7075-T6	THICK SKIN	7075-T6
42 F-111A	THICK SKIN	D6AC	THICK SKIN	D6AC	THICK SKIN	2024-T851	THICK SKIN	2024-T851
43 S-2E	'HAT'	7075-T6	'HAT'	7075-T6	'HAT'	7075-T6	'HAT'	2014-T3
44 S-3A	INTEG	7075-T7651	INTEG	7075-T7651	INTEG	7075-T7651	INTEG	7075-T7651
47 T-37A	CODED TO BODY	–	–	–	'HAT'	2024-T3	'HAT'	2024-T3
50 OV-1C	'Z'	7075-T6	'Z'	7075-T6	'Z'	7075-T6	'Z'	7075-T6

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	Aluminum	Titanium
F_S/ρ	0.240×10^6	0.305×10^6
Factor (K_{MTLSUB})	1.000	0.787

Other Philosophical Considerations

Other "philosophical" influences on the wing box weight may be enumerated, but sufficient definition within the existing data base simply could not be found. Damage tolerance is possibly a subset of the fail-safe factor but certain identification of enough wing boxes with this characteristic and a detailed understanding of the actual design impact of each makes specific identification of a factor an exercise in guesswork. Design-to-Cost considerations are reflected at a more detailed level by exact identification of materials and methods of construction in the material factors.

Modification of Previous Equations

The equations previously developed in the section on Theoretical Wing Weight Equations were now modified with the factors discussed above and a new regression analysis performed. The material and temperature factors are included as straight multiplying factors, all other factors were derived empirically. Both the original equations and the refitted equations are shown below:

$$W_{CVR} = 0.081223 \left[\frac{b \quad (C_R + 2\,C_T) \quad B\, n\, S_W}{\cos^2 \Lambda\, (C_R + C_T)\, (2T_R + T_T)\, (2C_R + C_T)} \right]^{0.5479} \left[S_{BOX} \right]^{0.4897}$$

[original: Equation (17) repeated]

$$W_{CVR} = 0.039041 \left[\frac{b \quad (C_R + 2C_T) \quad B\, n\, S_W}{\cos^2 \Lambda\, (C_R + C_T)\, (2T_R = T_T)(2C_R + C_T)} \right]^{0.5074} \left[S_{BOX} \right]^{0.5279}$$

$$\left[V_L \right]^{0.1634} K_{FSCVR}\; K_{MTLCVR}\; K_{TEMPCVR}$$

[new: Equation (19)]

Although it was not a requirement of the study, the standard deviation of the new equation has improved:

Standard deviation of the original equation = 19.5%

Standard deviation of the new equation = 17.0%

$$W_{SUB} = 0.00636 \left[B\ n\ S_W\right]^{0.5614} \left[S_{BOX}(T_R + T_T)\right]^{0.144}$$

[original: Equation (18) repeated]

$$W_{SUB} = 0.004147 \left[B\ n\ S_W\right]^{0.5598} \left[S_{BOX}(T_R + T_T)\right]^{0.1877} \left[K_{CT}\right]^{0.518} K_{MTLSUB}$$

[new: Equation (20)]

Standard deviation of the original equation = 30.6%

Standard deviation of the new equation = 28.2%

The standard deviation for the basic box improved from 17.4% to 14.5%. Correlation plots for cover weight, substructure weight and basic box weight are shown in Figures 10, 11 and 12, respectively.

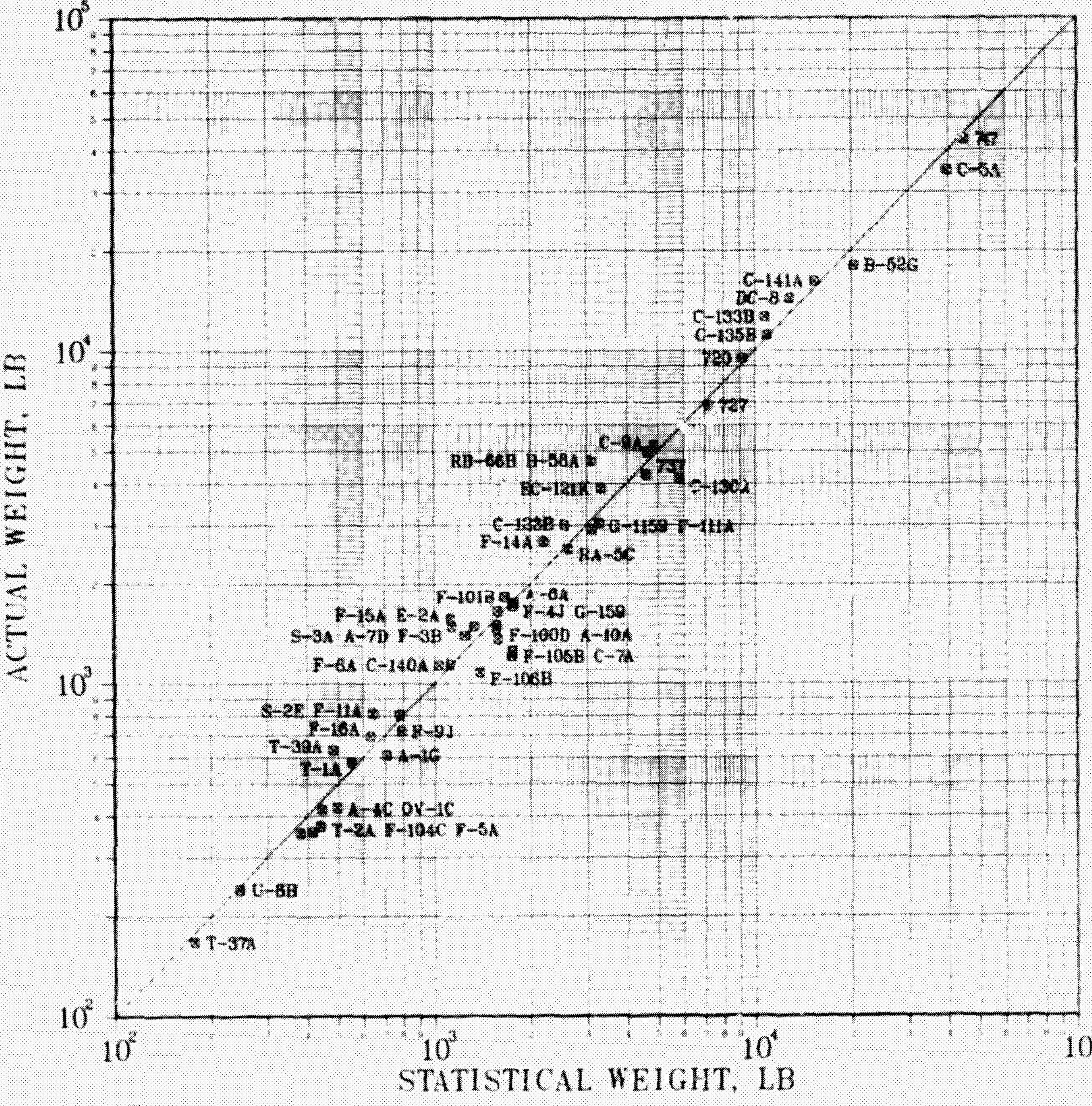

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Figure 10. Correlation of basic box cover weight estimate (new equation).

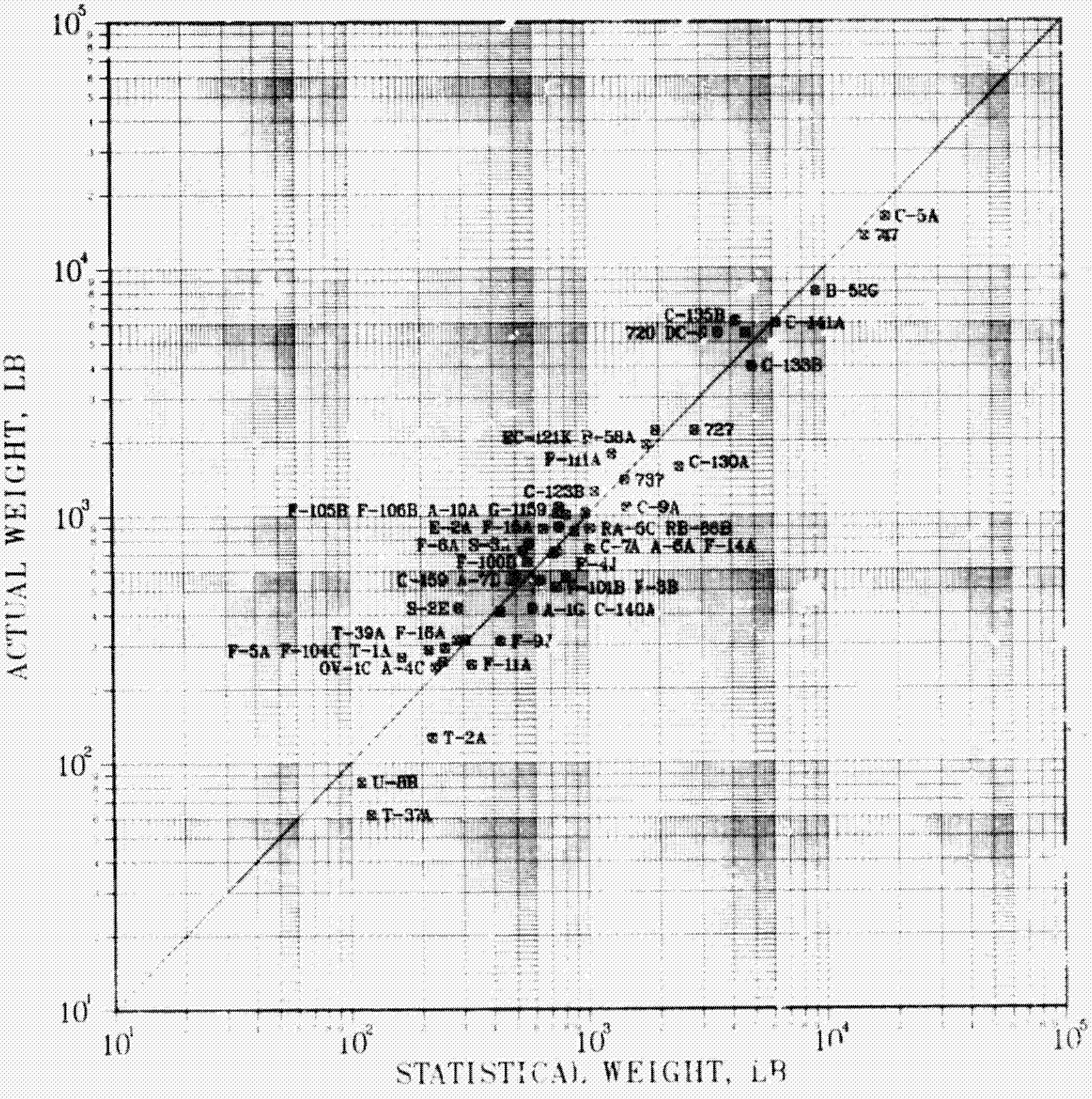

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Figure 11. Correlation of basic box substructure weight estimate (nev ation).

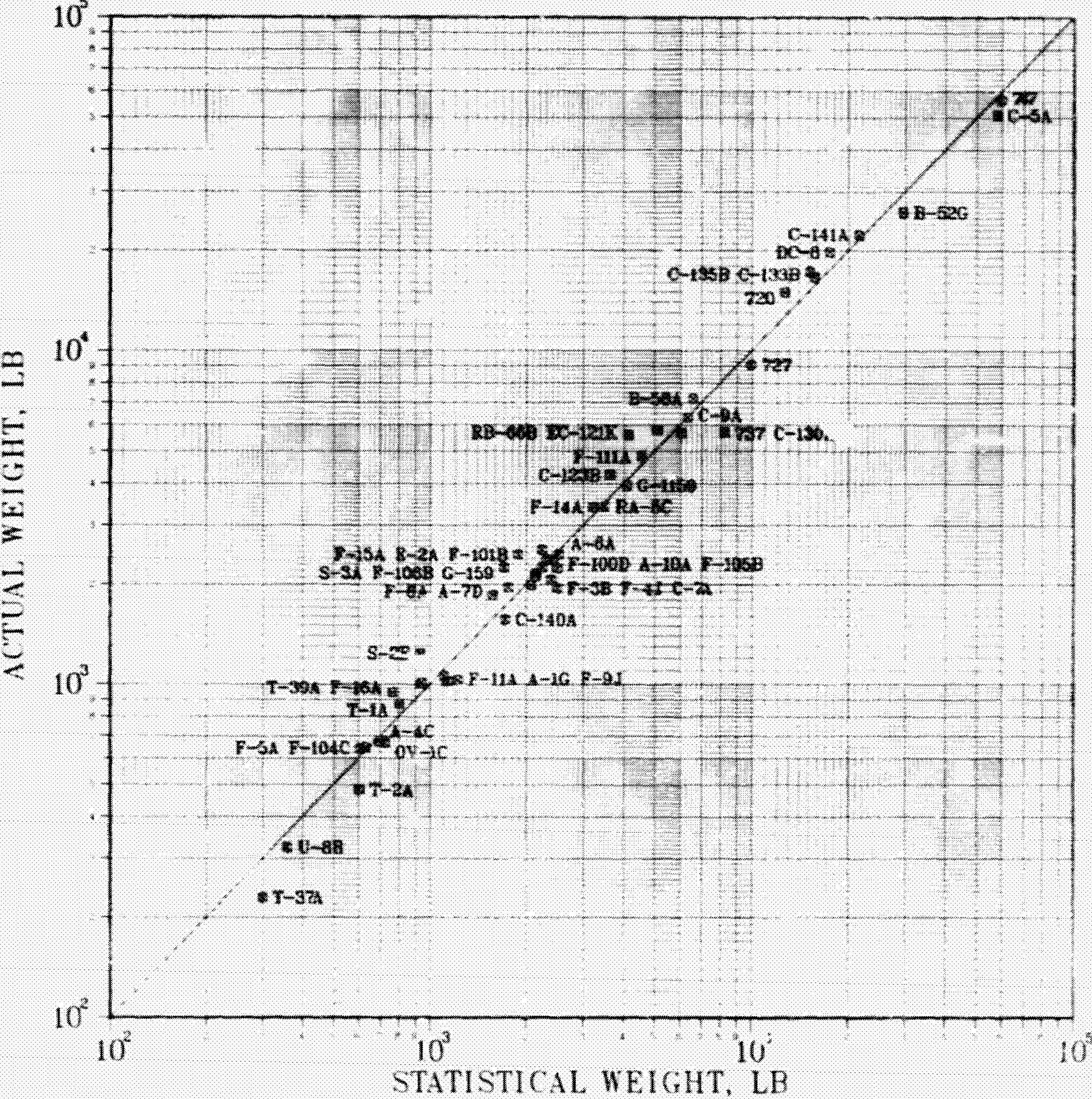

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Figure 12. Correlation of total basic box weight estimate (new equation).

TOTAL WING WEIGHT METHODOLOGY

The equations derived in the preceding sections, for wing box covers and wing box substructure, estimate only the basic wing box weight. To obtain a total wing weight, equations have been developed to account for weight penalties to the wing box plus the additional components of the wing.

Wing Box Penalty Functions

Store Penalty To Wing Box (Figure 13).

$$0.01\ W_{WSTORES}$$

$0.014\ W_{WSTORES}$ for sweeping store stations (i.e., F-111A)

Main Landing Gear Penalty To Wing Box (Figure 14).

$$0.001416\ N_{LDGW}\ LDGW\ K_{MG}$$

Where K_{MG} is 1.0 except 0.5938 if main landing gear are in engine nacelles on the wing.

Wing Fuel Penalty To Wing Box (Figure 15).

$$0.9191\left(W_{WFUEL}\right)^{0.5436}$$

Engine Penalty To Wing Box (Figure 16).

$$0.004\ F_W$$

or $0.03\ HP_W$

Wing Fold Or Wing Pivot Penalty (Figure 17).

$$0.03386\left(B\ n\right)^{0.2477}\left(S_W\right)^{1.248}\left(1-\frac{b'}{b}\right)^{1.307}K_{WS}$$

Where K_{WS} is 1.0 for folding wings and 0.556 for variable sweep wings.

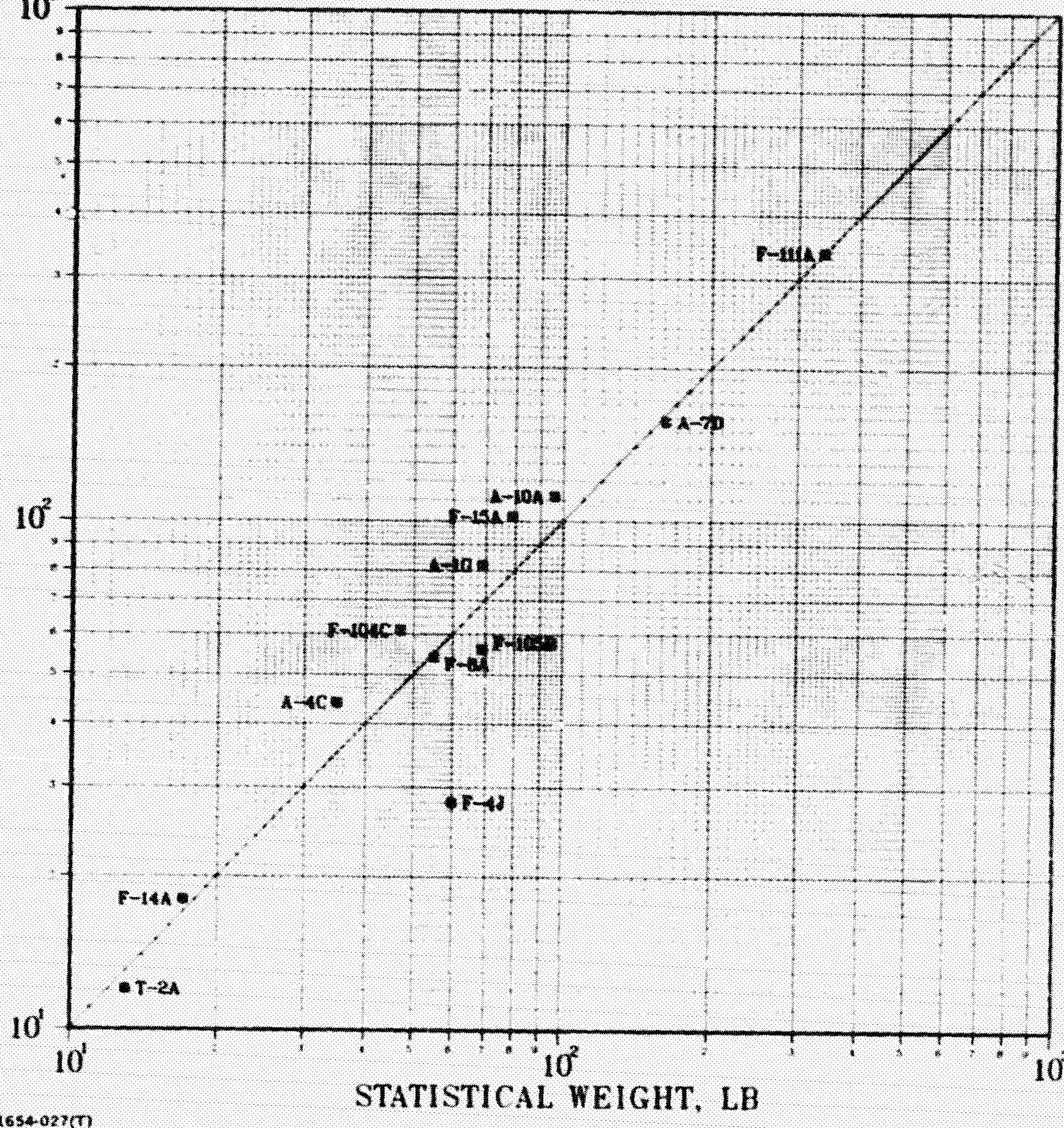

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Figure 13. Correlation of wing box stores weight penalty estimate.

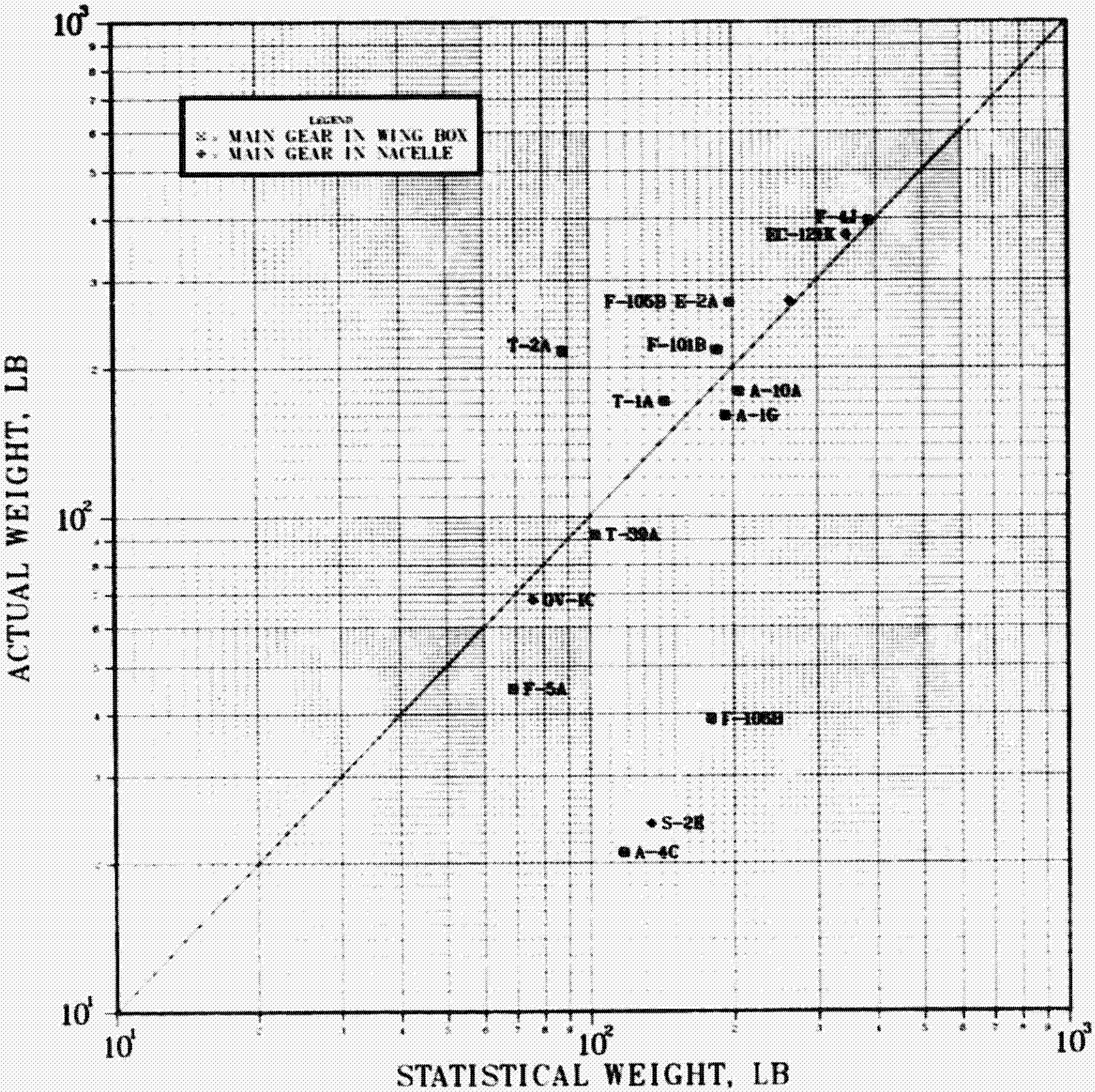

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Figure 14. Correlation of wing box main gear weight penalty estimate.

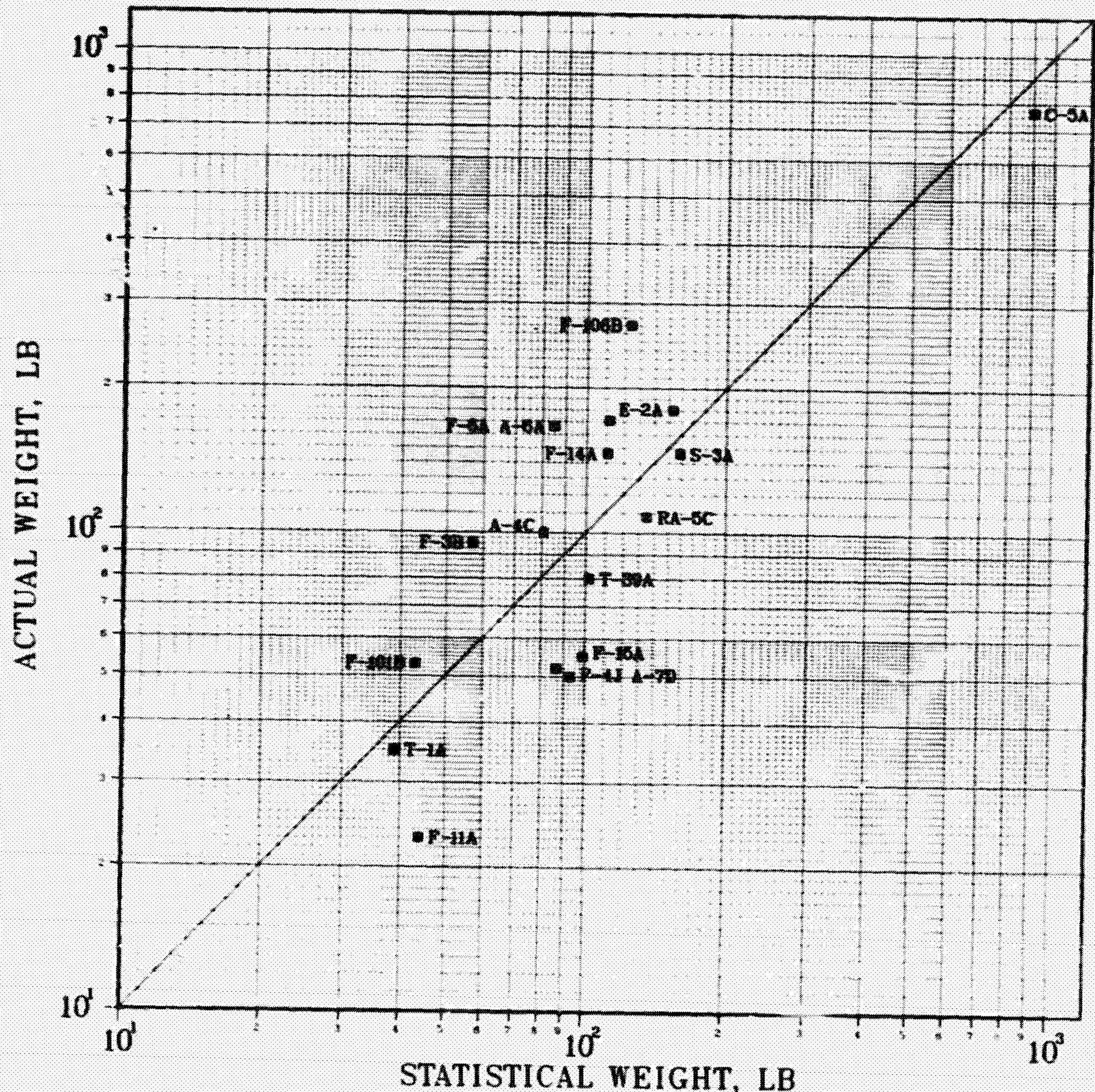

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Figure 15. Correlation of wing box fuel weight penalty estimate.

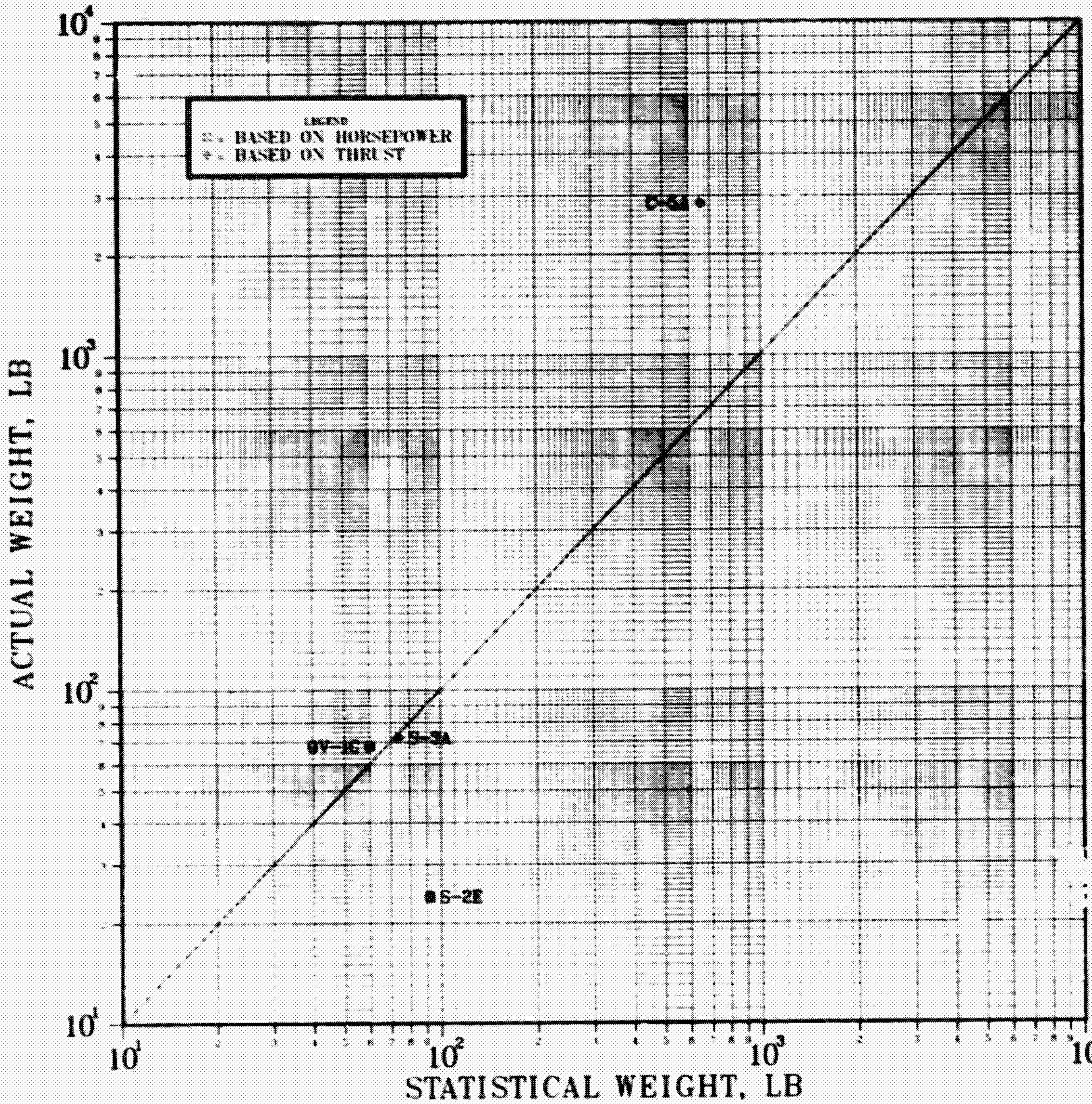

Figure 16. Correlation of wing box engine weight penalty estimate.

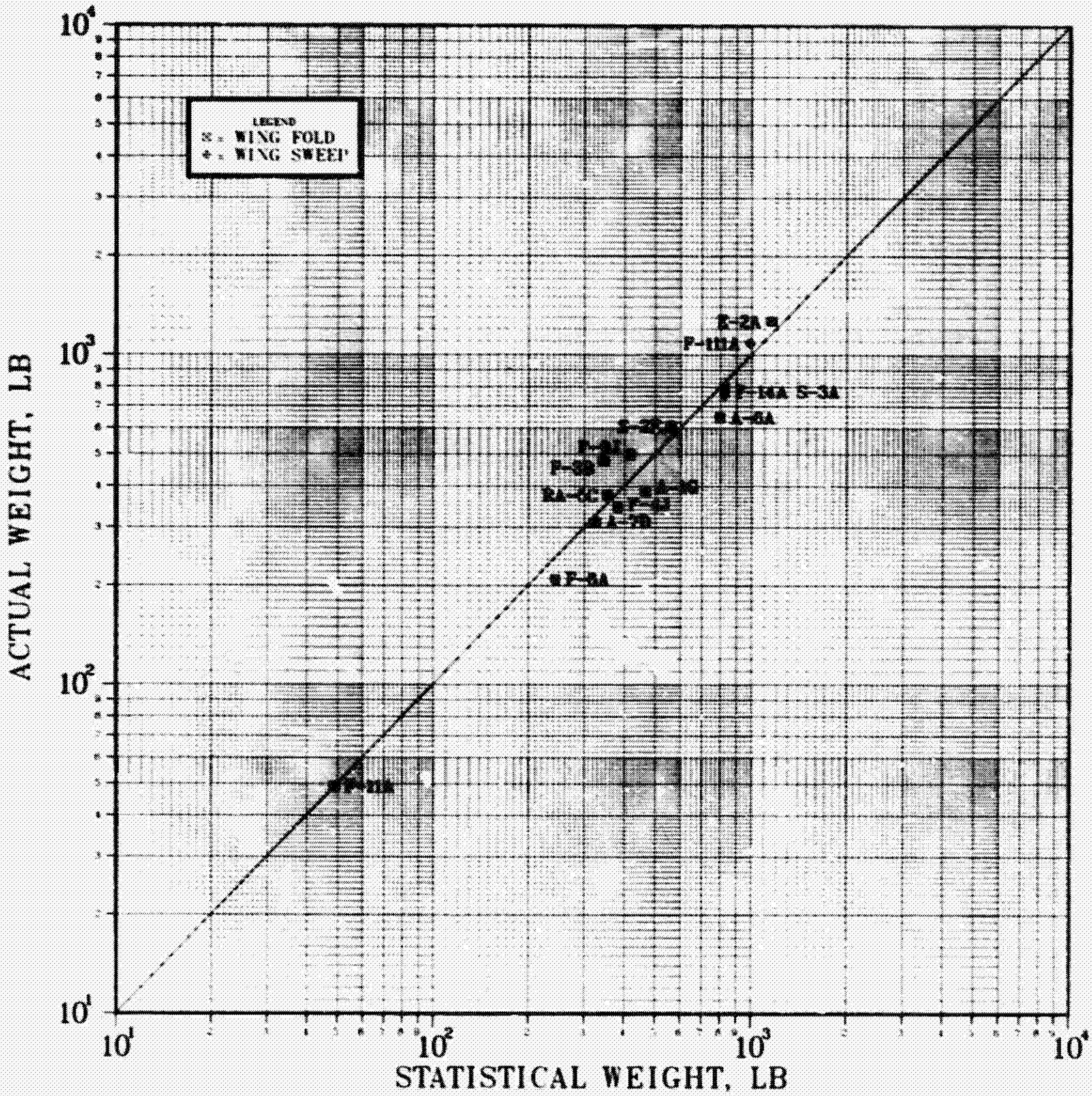

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Figure 17. Correlation of wing box fold/sweep weight penalty estimate.

Non-Wing Box Basic Structures, Secondary Structure and Control Surfaces

Leading Edge, Trailing Edge and Miscellaneous Secondary Structure (Figure 18).

$$0.07235\ (S_W - S_{BOX})^{0.2595}\ (TOGW)^{0.5281} (S_W)^{0.3192}\ K_{LED}$$

Where K_{LED} is 1.0 except 0.8470 for a leading edge device.

Landing Gear Doors and Mechanism (Figure 19).

$$0.8991 (S_{MGDR})^{1.067} (V_L)^{0.2252}$$

Ailerons, Elevons, Flaperons and Decelerons (Figure 20).

$$0.06564 (S_{ROLL})^{0.8697} \left(\frac{TOGW}{S_W}\right)^{1.049}\ K_{ROLL}\ K_{BW}$$

Where K_{ROLL} is 1.0 except 1.732 for elevons, 1.023 for flaperons, 1.609 for decelerons and K_{BW} is 1.0 except 1.541 for ailerons, elevons, flaperons or decelerons with balance weights.

Trailing Edge Flaps (Figure 21).

$$0.0008759\ S_{FLAP} (V_L)^{0.3565}\ n^{0.1576} (C_{L_{MAX}}\ LDGW)^{0.321} (V_S)^{0.5}\ K_{TS}$$

Where K_{TS} is 1.0 except 1.976 for triple slotted flaps.

Slats (Figure 22).

$$0.2727\ S_{SLAT}\ (V_L)^{0.4703}$$

Leading Edge Flaps (Figure 22).

$$0.31\ S_{LEF}\ (V_L)^{0.4703}$$

Spoilers (Figure 23).

$$0.2697\ (S_{SPOIL})^{0.8699}\ (V_L)^{0.3461}\ (S_W)^{0.8445}\ b^{-1.117}$$

Wing Speed Brakes (Figure 24).

$$.01053\ S_{WSB}\ (TOGW)^{0.5909}$$

Winglets*.

0.0386 W_{WING}

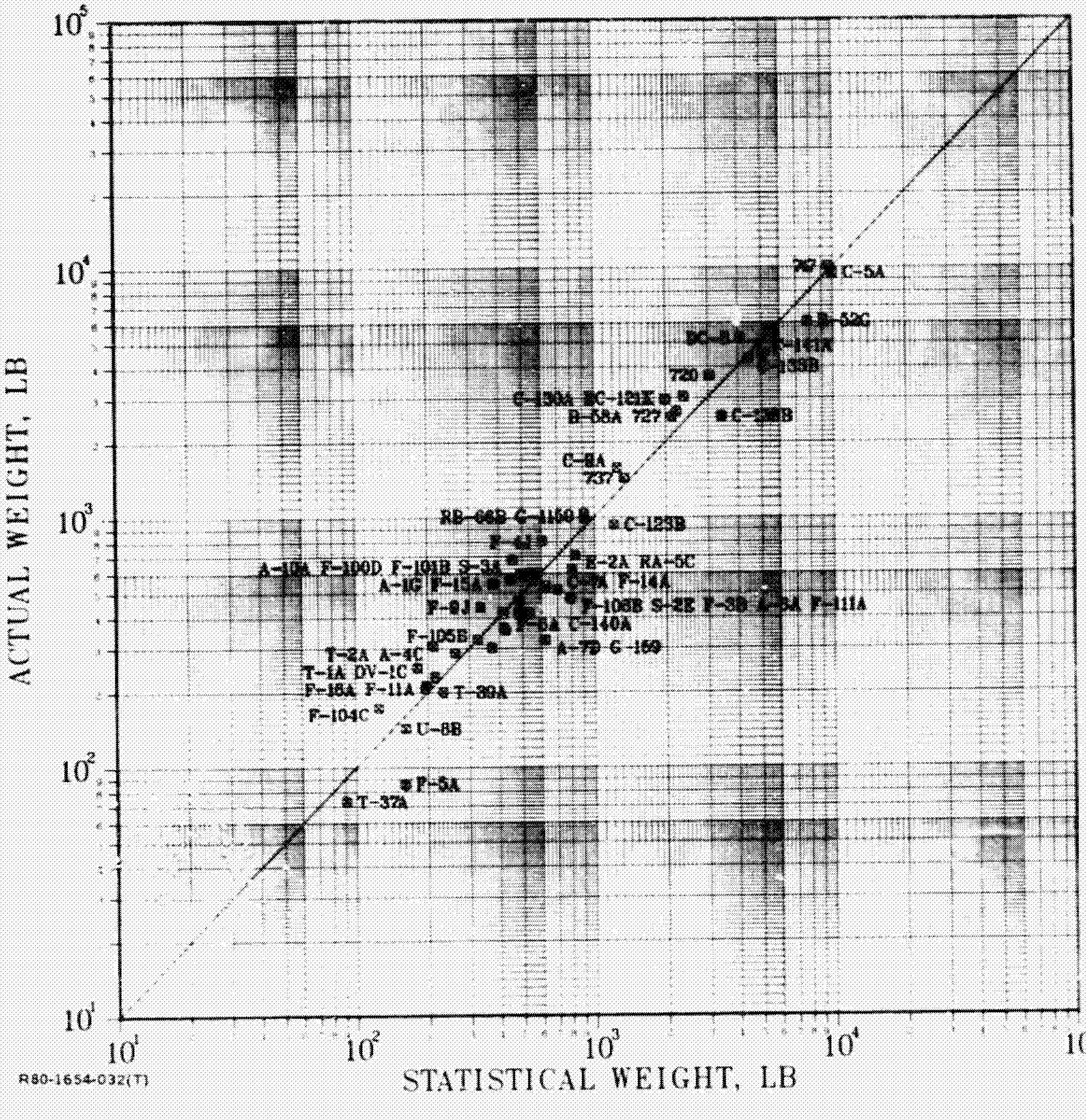

Figure 18. Correlation of L.E., T.E., & miscellaneous weight estimate.

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*Based only on Gulfstream III.

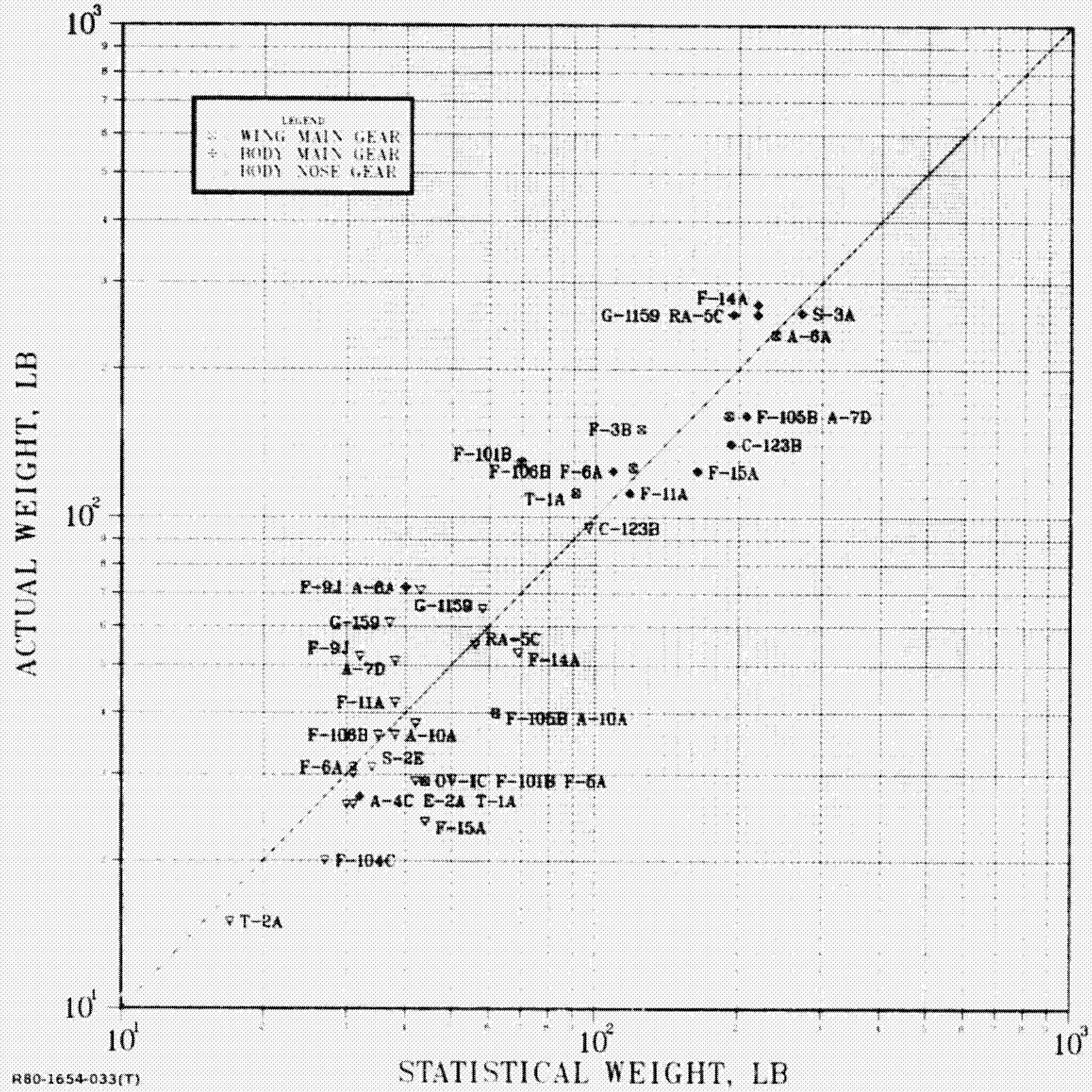

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Figure 19. Correlation of L.G. door and mechanism weight estimate.

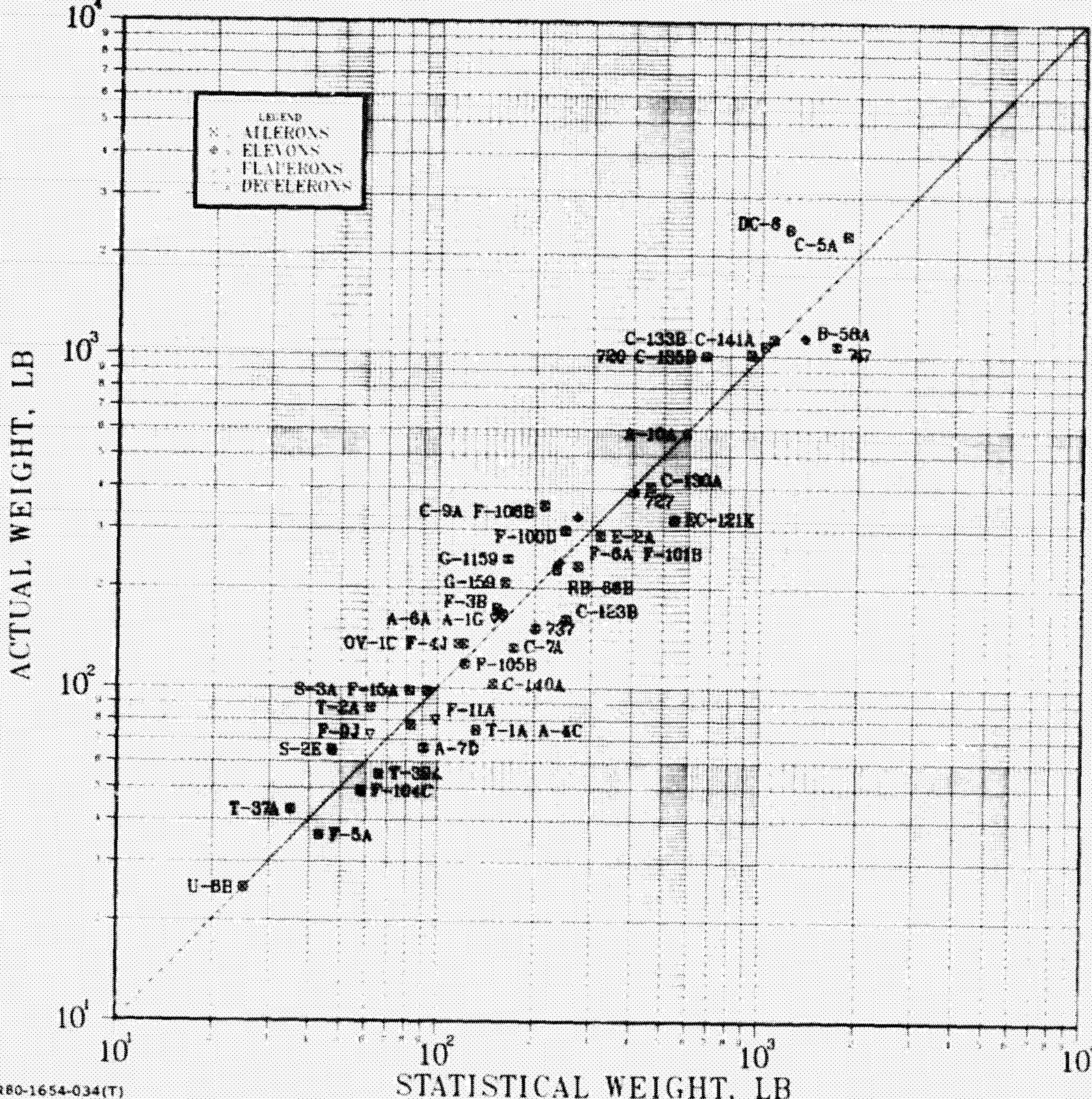

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Figure 20. Correlation of T.E. roll control device weight estimate.

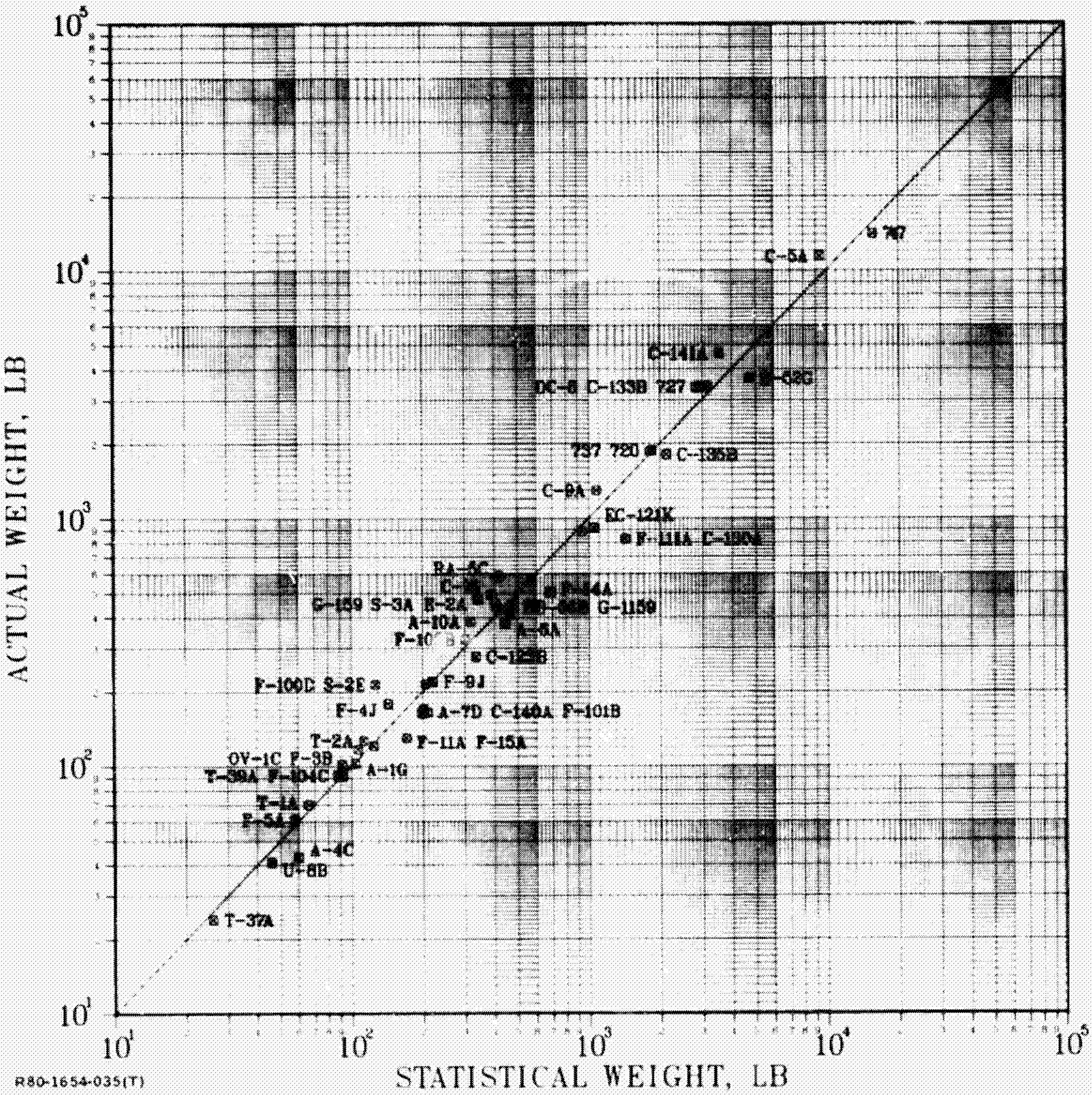

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Figure 21. Correlation of trailing edge flap weight estimate.

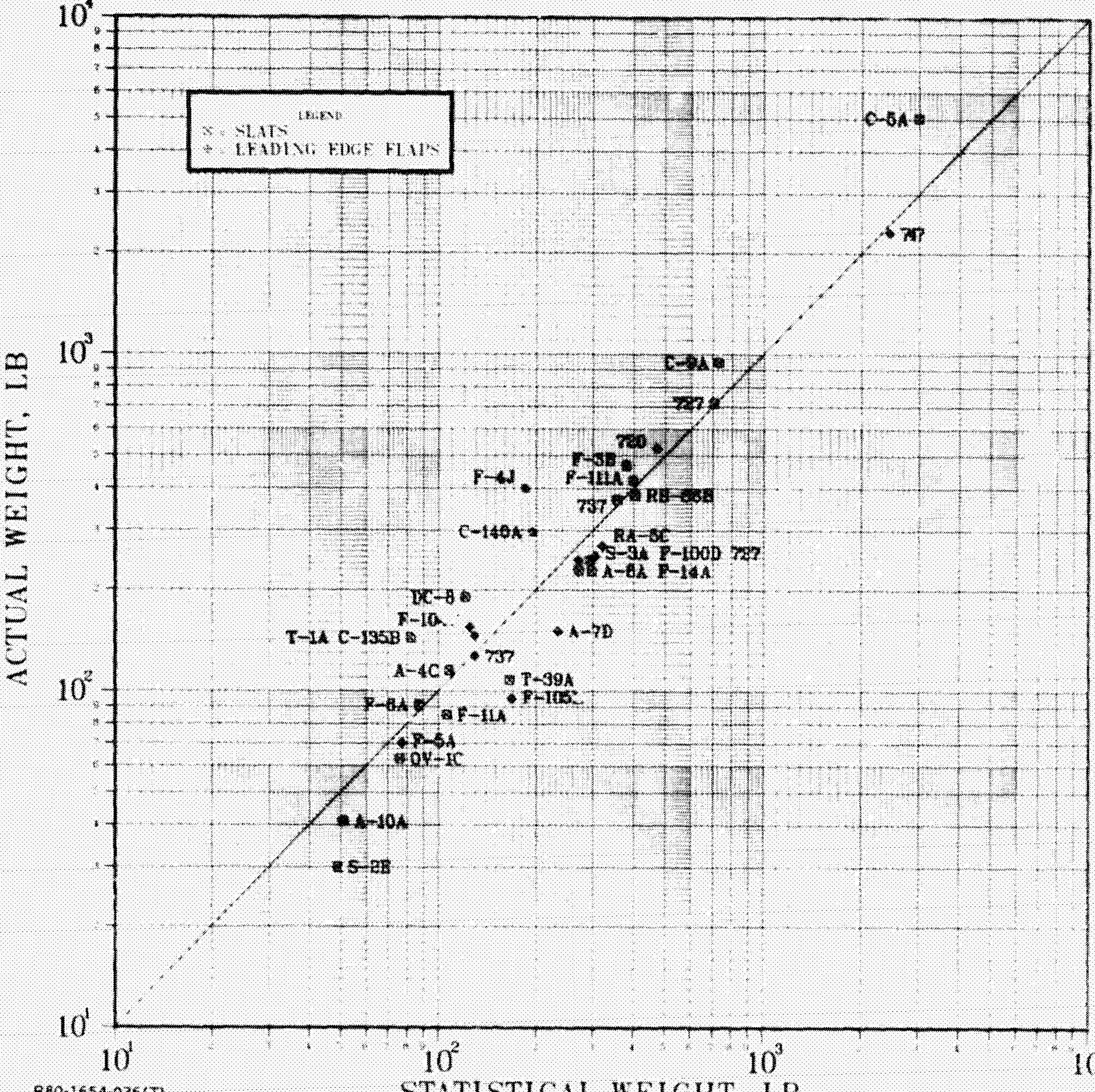

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Figure 22. Correlation of slat & leading edge flap weight estimate.

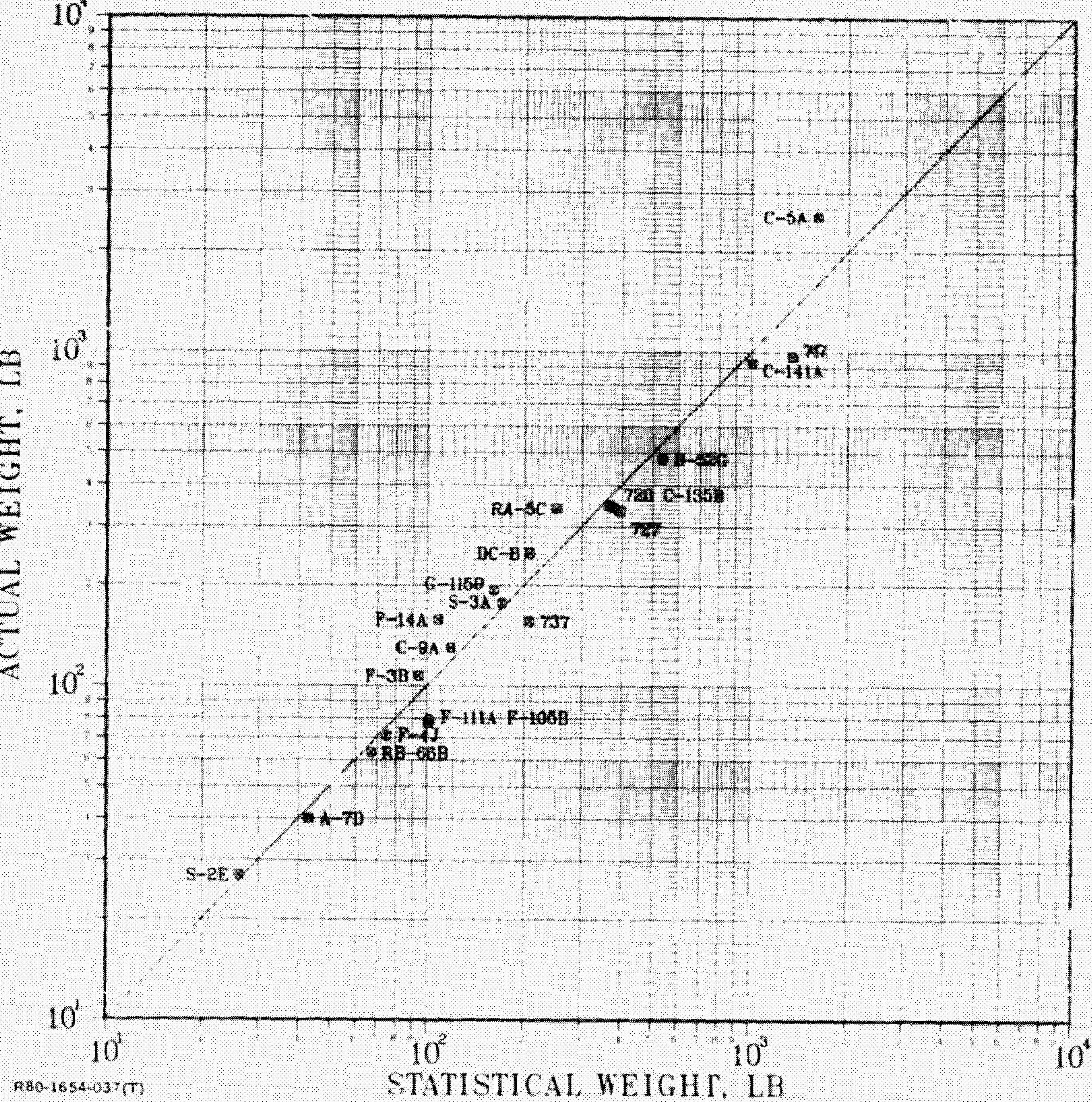

Figure 23. Correlation of spoiler weight estimate.

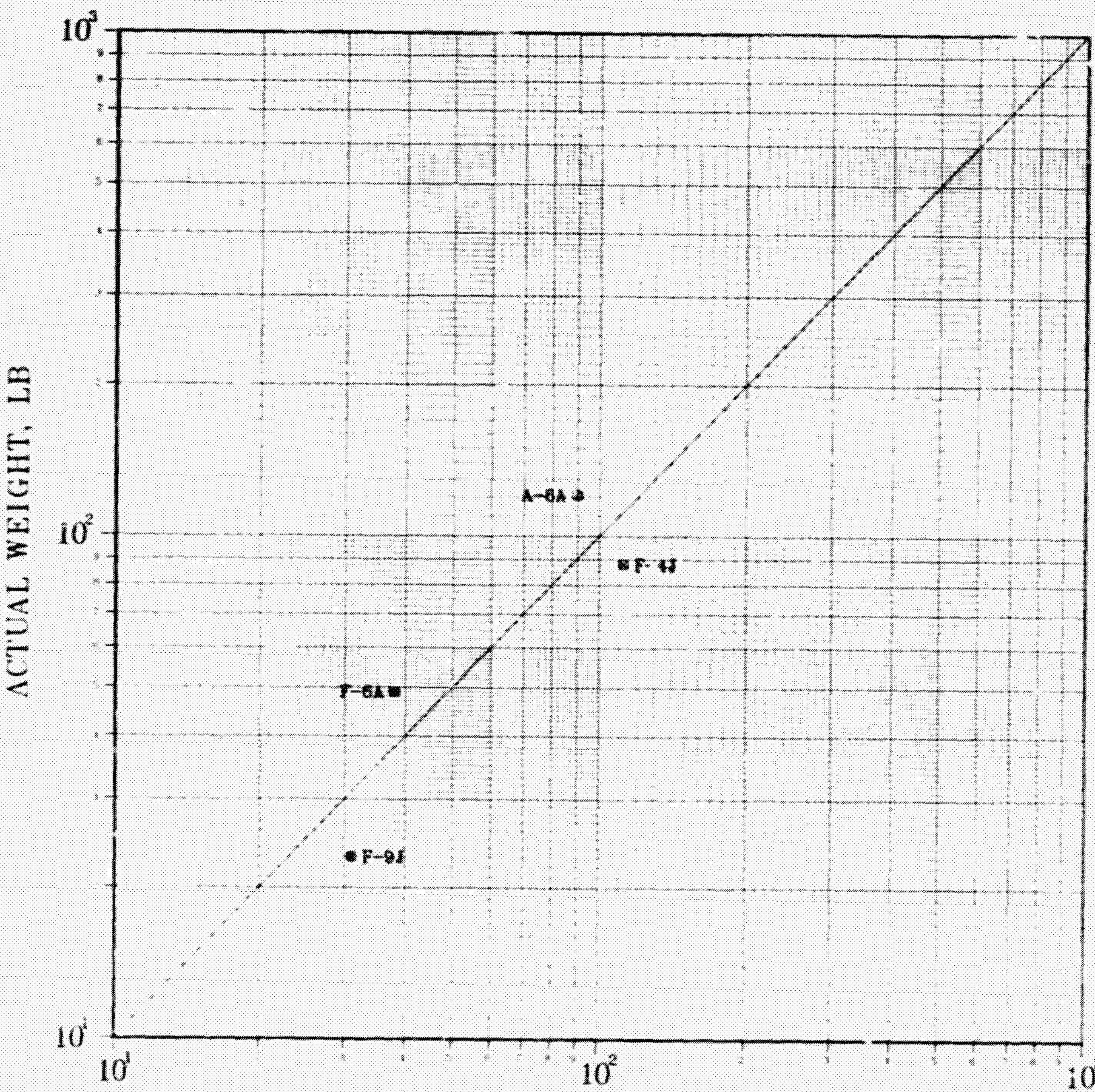

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Figure 24. Correlation of wing speed brake weight estimate.

DEFAULT ALGORITHMS

The inputs required for the developed methods may be too detailed for use early in the preliminary design cycle. The usual solution to this problem is to revert to a simplified equation, however, the generalized nature of these equations greatly reduces flexibility. In place of a simplified equation, a series of algorithms have been developed that allow defaulting the methods to approximate the input complexity of a simplified equation. This will retain the flexibility of the method to perform detail tradeoffs early in the design process, while retaining the inherent simplicity of inputs required for initial sizing.

Default Parameters

a) Wing Box Area (S_{BOX}) - Figure 25

$$0.4185\ S_W^{1.0159}$$

b) Trailing Edge Roll Control Device Area (S_{ROLL}) - Figure 26
(Aileron or Elevon Area)

$0.05\ S_W$ (for horizontal tail area greater than zero)

$0.10\ S_W$ (for horizontal tail area equal to zero)

c) Trailing Edge Flap Area (S_{FLAP}) - Figure 27

$0.08\ S_W$ (land based, fighter/attack)

$0.12\ S_W$ (carrier based, fighter/attack)

$0.18\ S_W$ (bomber, transport, cargo)

d) Leading Edge Device Area (S_{LED}) - Figure 28
(Slat (S_{SLAT}) and/or L. E. Flap (S_{LEF}) Area)

$0.08\ S_W$ (for horizontal tail area greater than zero)

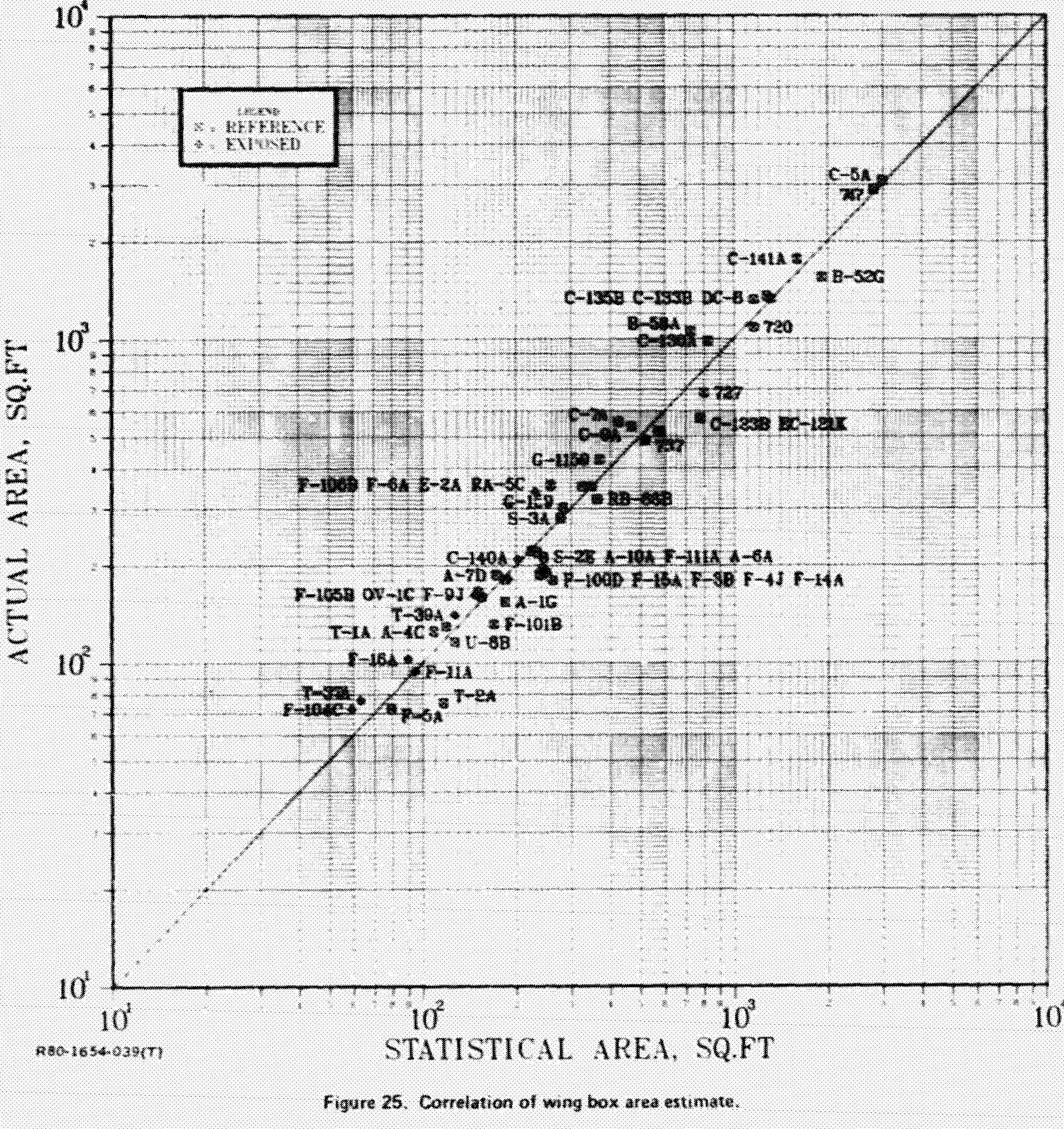

Figure 25. Correlation of wing box area estimate.

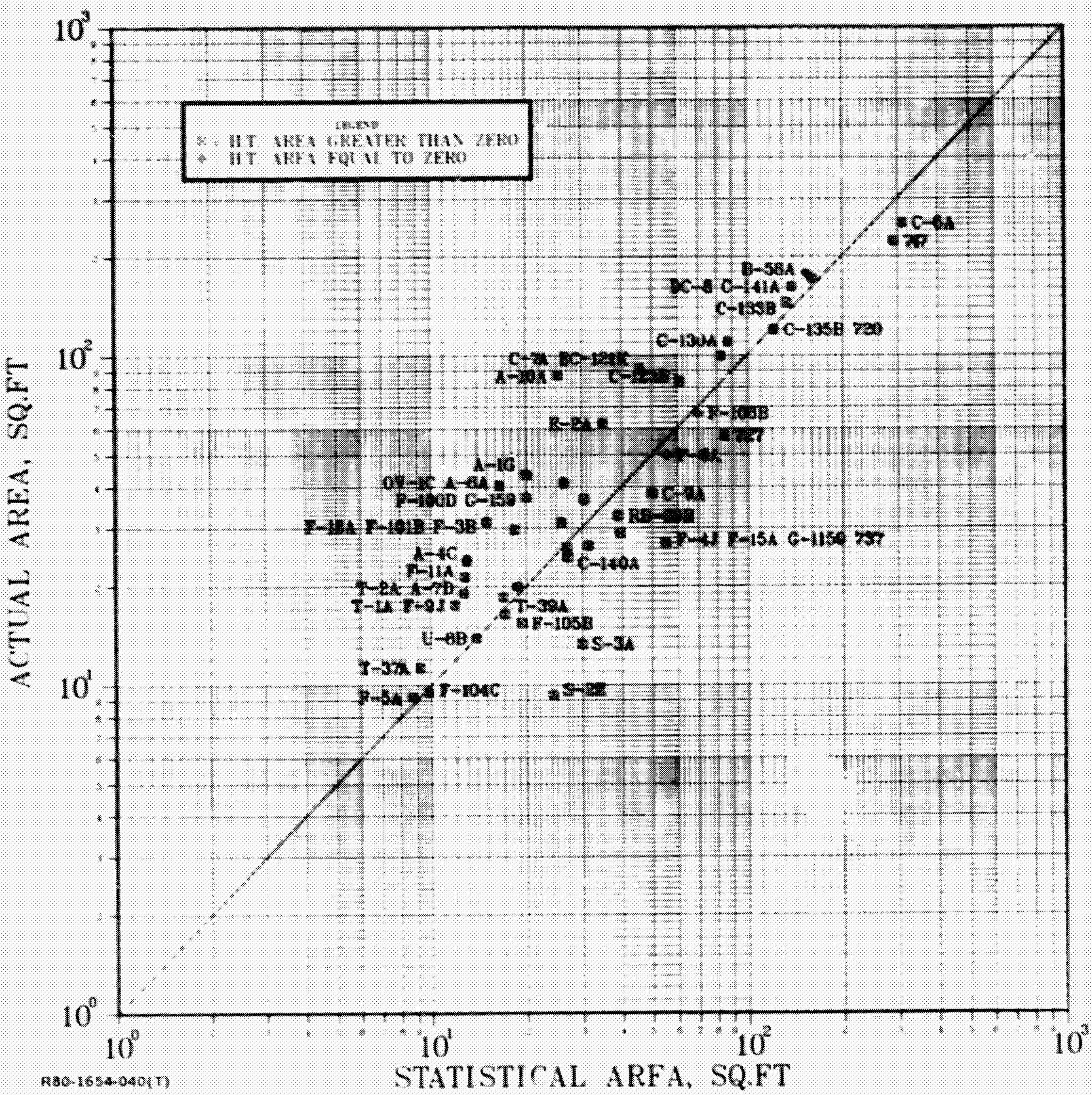

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Figure 26. Correlation of T.E. roll control device area estimate.

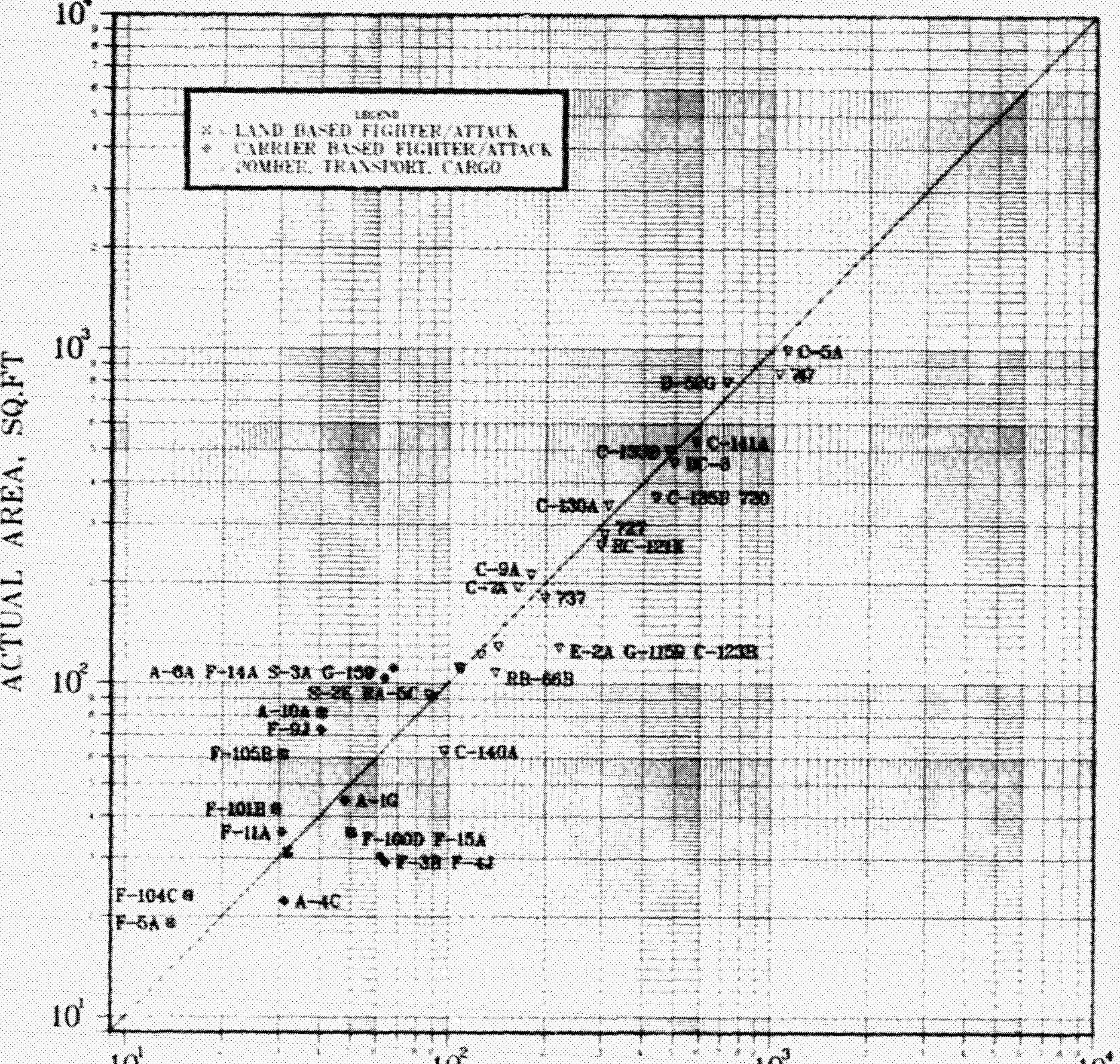


Figure 27. Correlation of trailing edge flap area estimate.

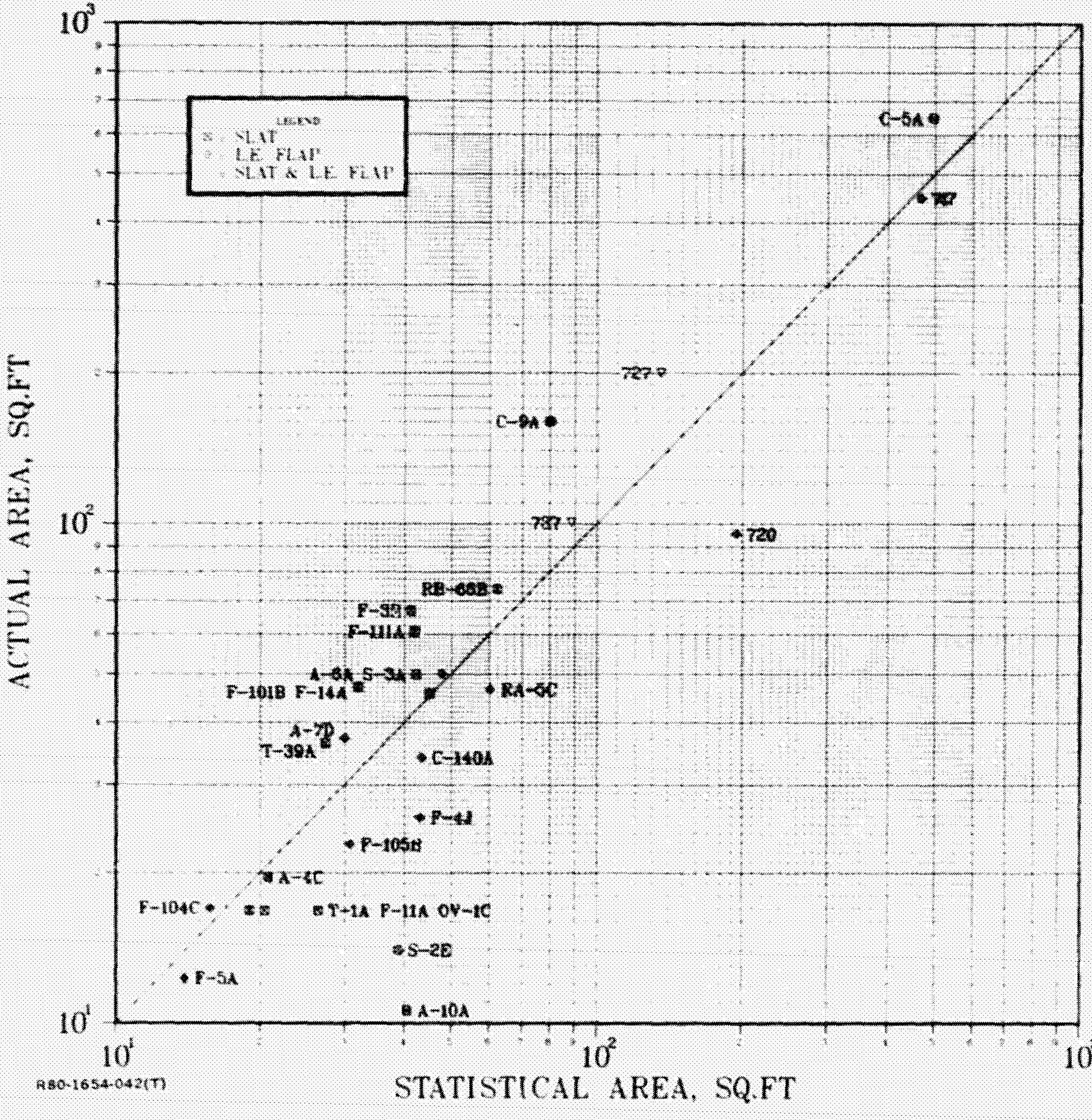


Figure 28. Correlation of leading edge device area estimate.

e) Spoiler Area (S_{SPOIL}) - Figure 29

$0.05\ S_W$ (carrier based, fighter/attack)

$0.07\ S_W$ (bomber, transport, cargo)

for horizontal tail area greater than zero

f) Wing Speed Brake Area (S_{WSB}) - Figure 30

$0.03\ S_W$ (carrier based, fighter/attack)

g) $C_{L_{MAX}}$

$$\frac{295 \times LDGW}{V_S^2 \times S_W} - 0.8\left(\frac{S_{SLAT} + S_{LEF}}{S_W}\right)$$

h) Ultimate Load Factor at LDGW (N_{LDGW}) - Figure 31

4.2 (average for land based)

7.4 (average for carrier based)

i) Landing Design Gross Weight (LDGW) - Figure 32

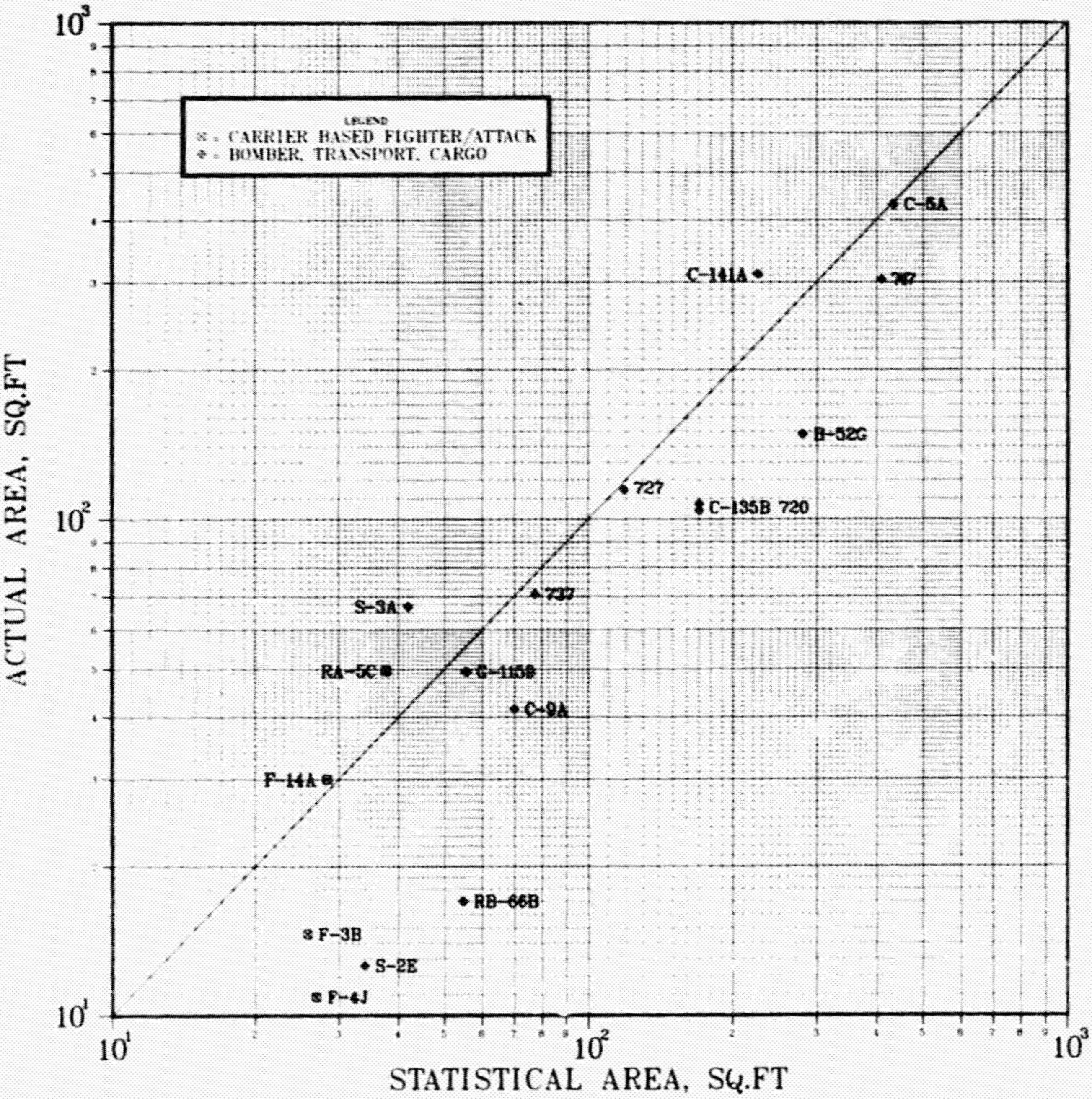
$$1.6149\ (TOGW)^{0.93983}\ K_{BASE}\ K_{TYPE}$$

Where K_{BASE} is 1.0 except 0.9712 for carrier based
and K_{TYPE} is 1.0 except 0.9407 for fighter/attack
and 1.0201 for bomber, transport and cargo

j) Wing Fuel Weight (W_{WFUEL}) - Figure 33

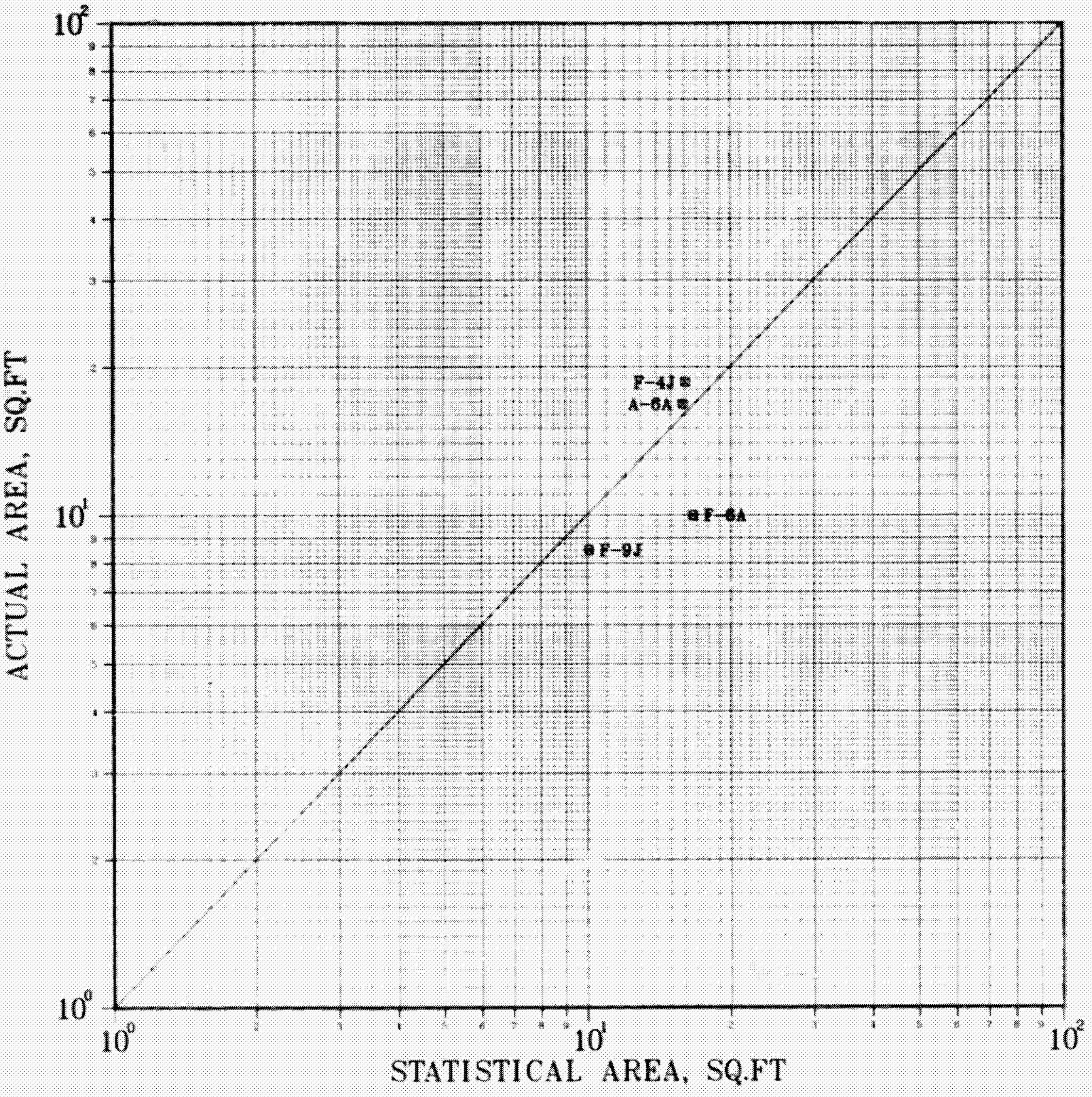
$$0.09179\ (S_{BOX})^{0.74358}\ (TOGW)^{0.68475}\ K_{BASE}\ K_{TYPE}$$

Where K_{BASE} is 1.0 except 0.9659 for carrier based
and K_{TYPE} is 1.0 except 0.6031 for fighter/attack
and 0.8958 for bomber, transport and cargo



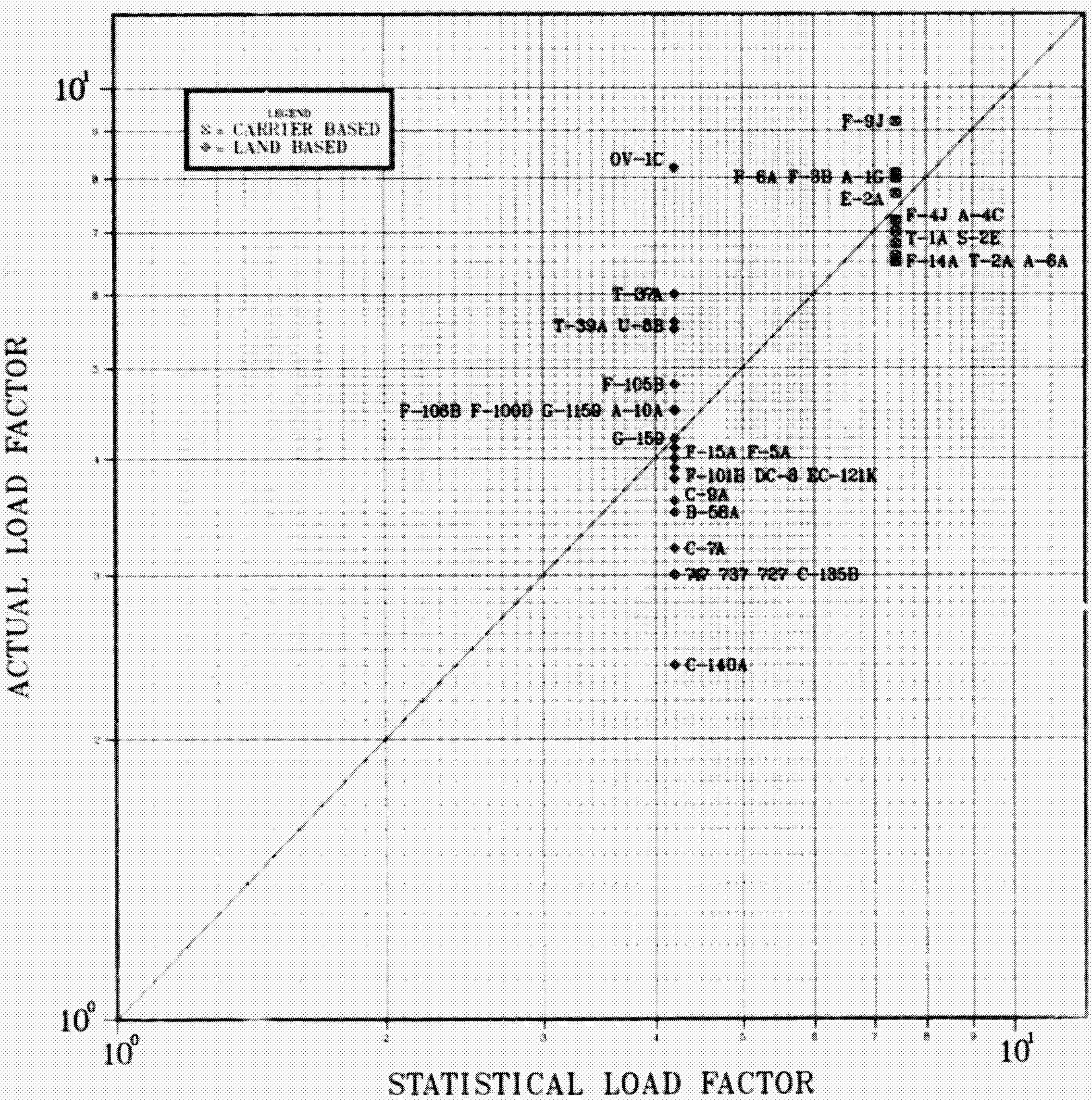
R80 ...54-043(T)

Figure 29. Correlation of spoiler area estimate.



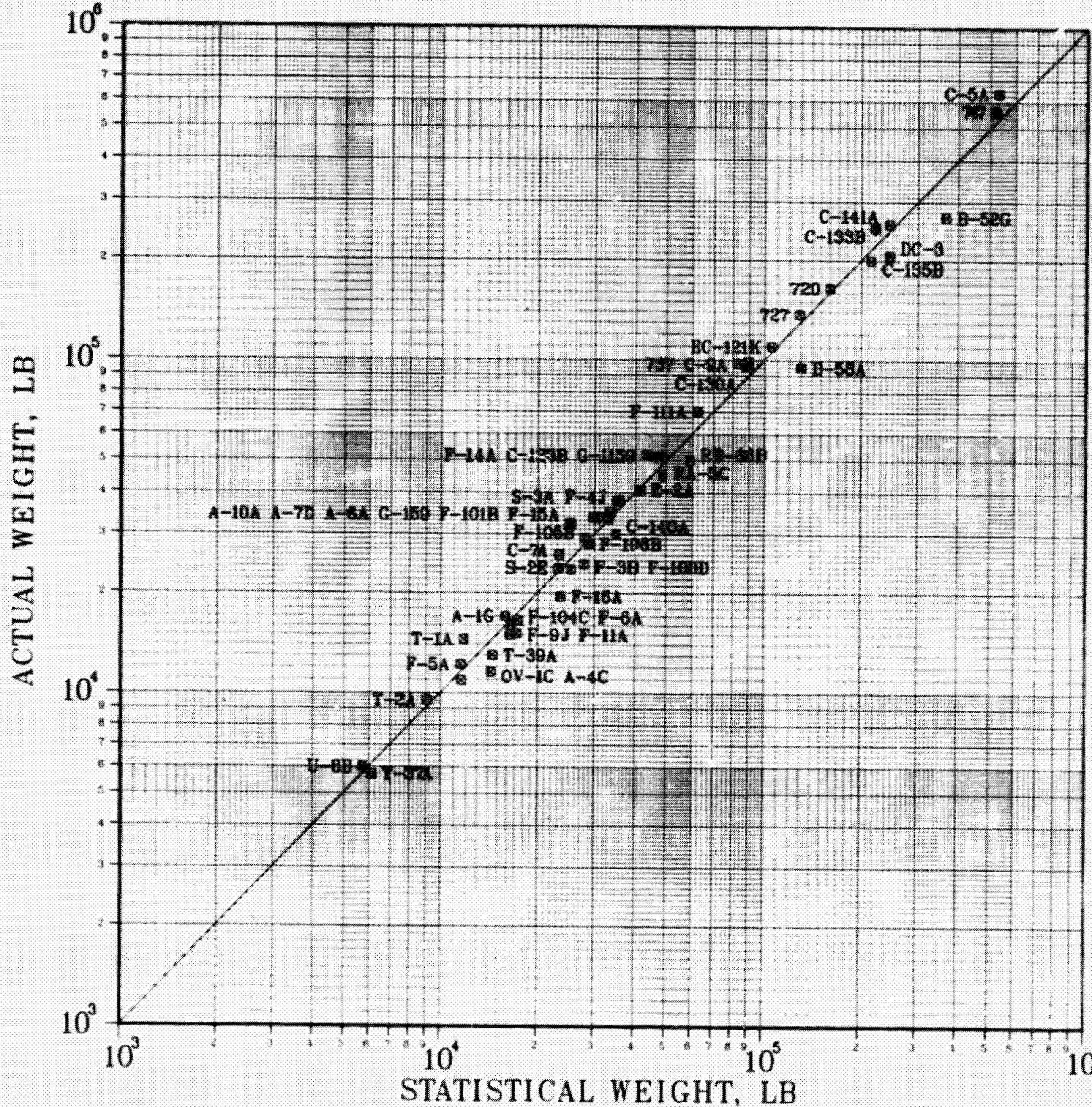
R80-1654-044(T)

Figure 30. Correlation of wing speedbrake area estimate.



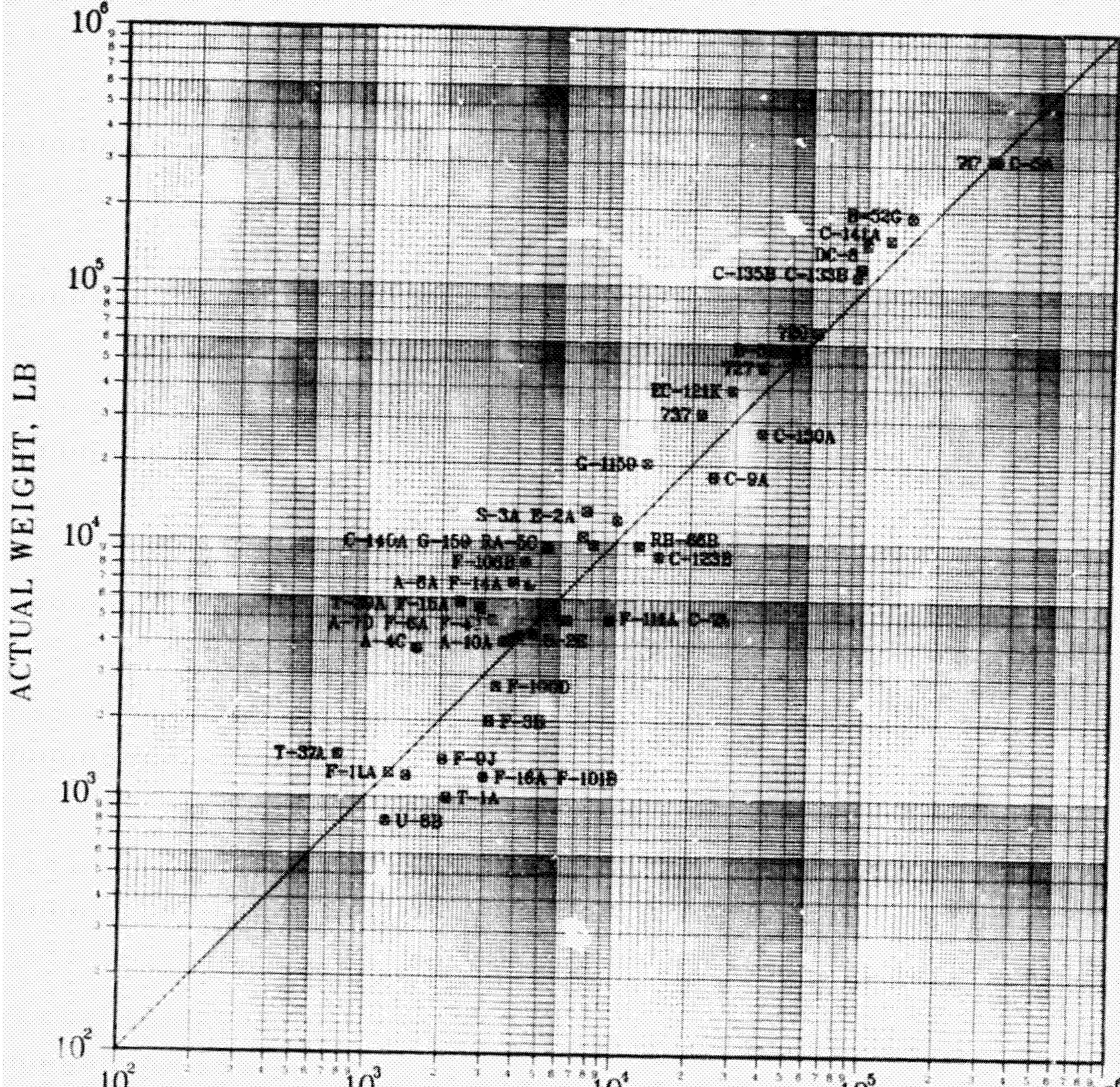
R80-1654-045(T)

Figure 31. Correlation of ultimate load factor at LDGW estimate.



R80-1654-046(T)

Figure 32. Correlation of landing design gross weight estimate.



R80-16[illegible]-047(T)

Figure 33. Correlation of wing fuel weight estimate.

k) Main Landing Gear Door Area (S_{MGDR}) - Figure 34

$$0.01027\ (LDGW)^{0.72629}\ K_{BASE}$$

Where K_{BASE} is 1.0 except 1.957 for carrier based

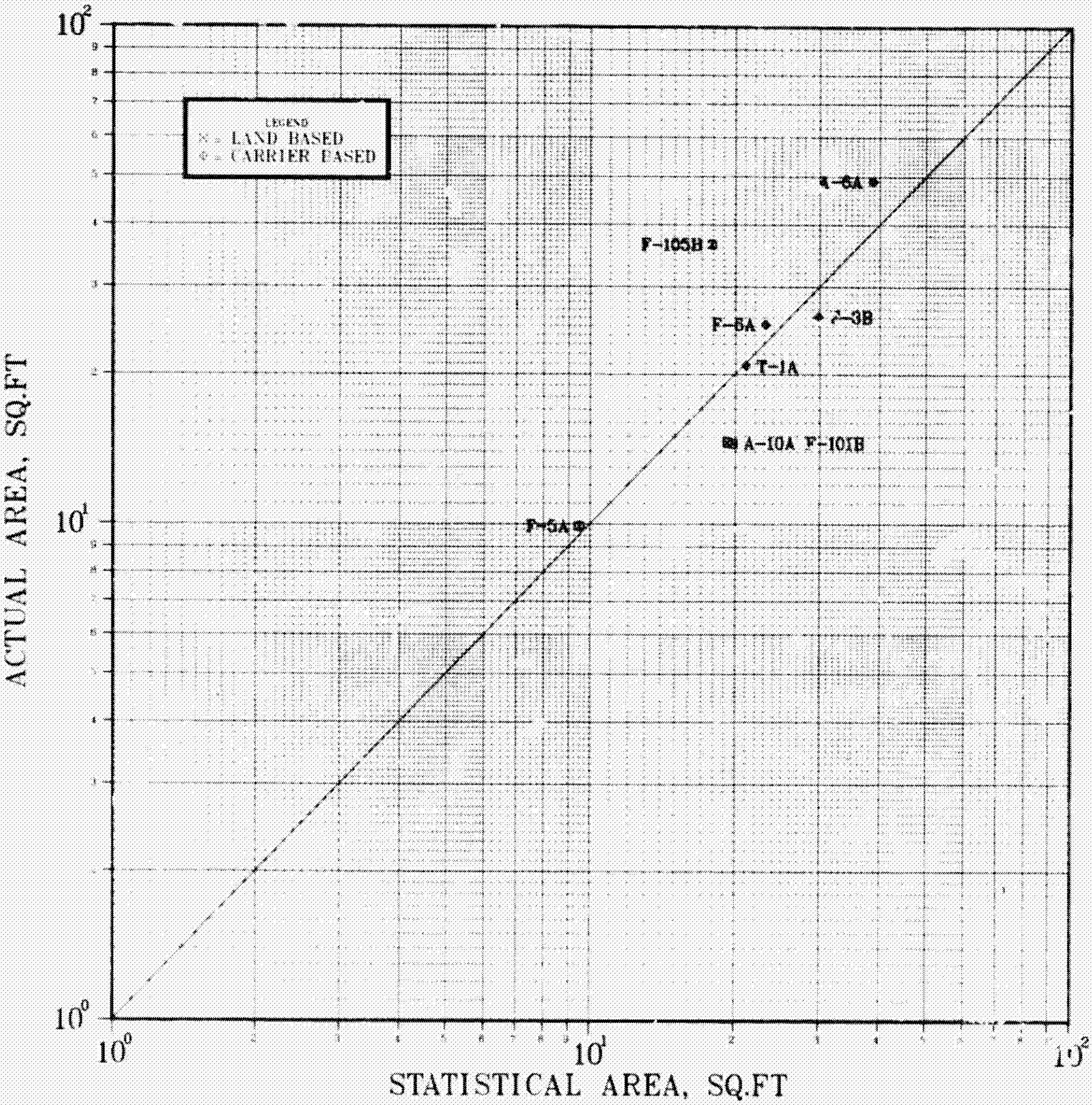
l) Maximum Zero Wing Fuel Weight (MZWFW) - Figure 35

$$2.923\ (TOGW)^{0.8819}$$ for bomber, transport, cargo only

m) Ultimate Load Factor (N_{GUST})

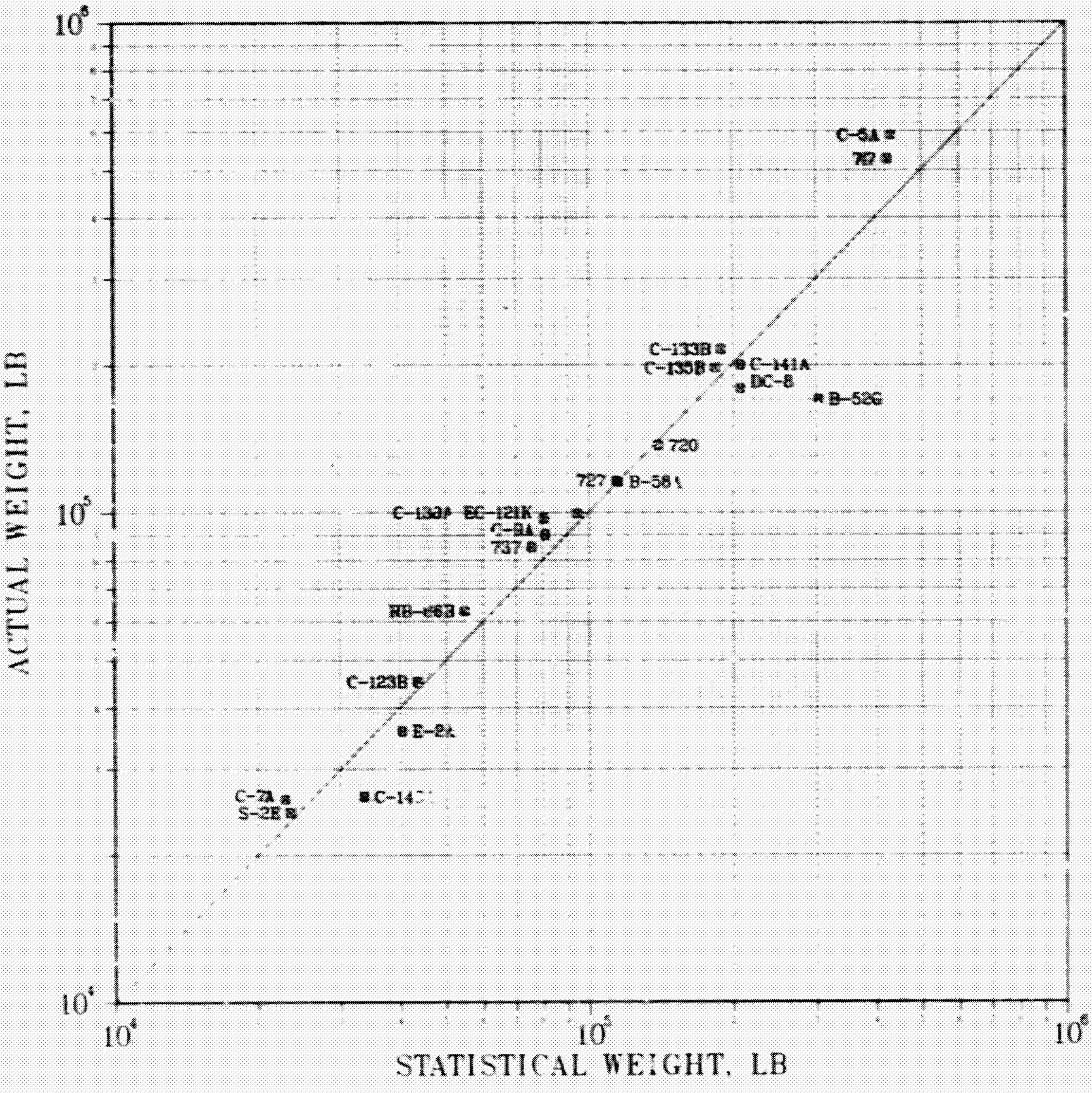
$$1.5 + \frac{0.8\, b^2\, V_L}{MZWFW \left[2 + \left(\frac{b^4}{S_W^2\, \text{Cos}^2 \Lambda} + 4 \right)^{0.5} \right]}$$

as an approximation of load factor for gust conditions.



R80-1654-048(T)

Figure 34. Correlation of main landing gear door ... imate.



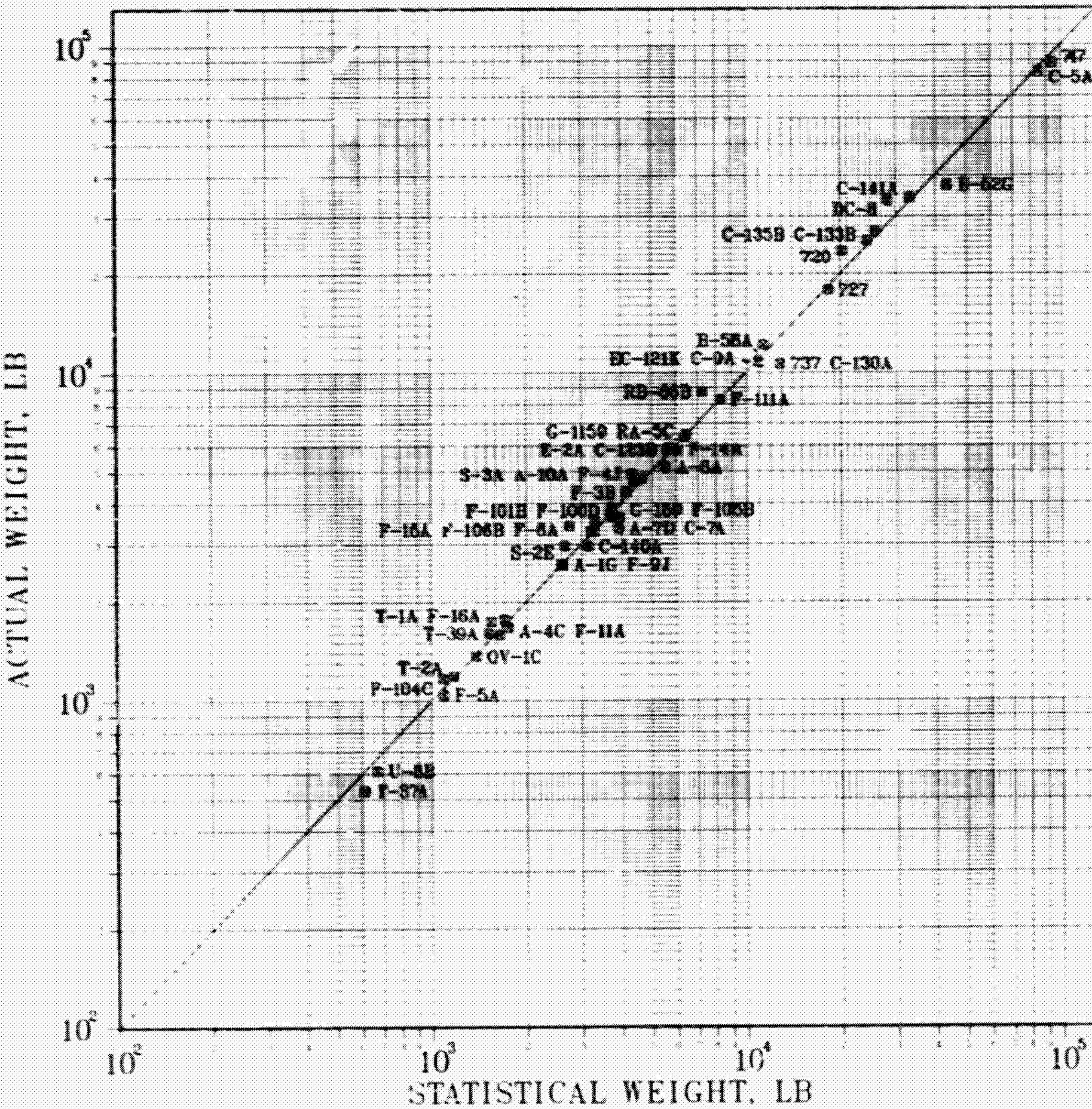
R80-1654-055(T)

Figure . Correlation of maximum zero wing fuel weight estimate.

SUMMARY OF METHOD AND INPUTS

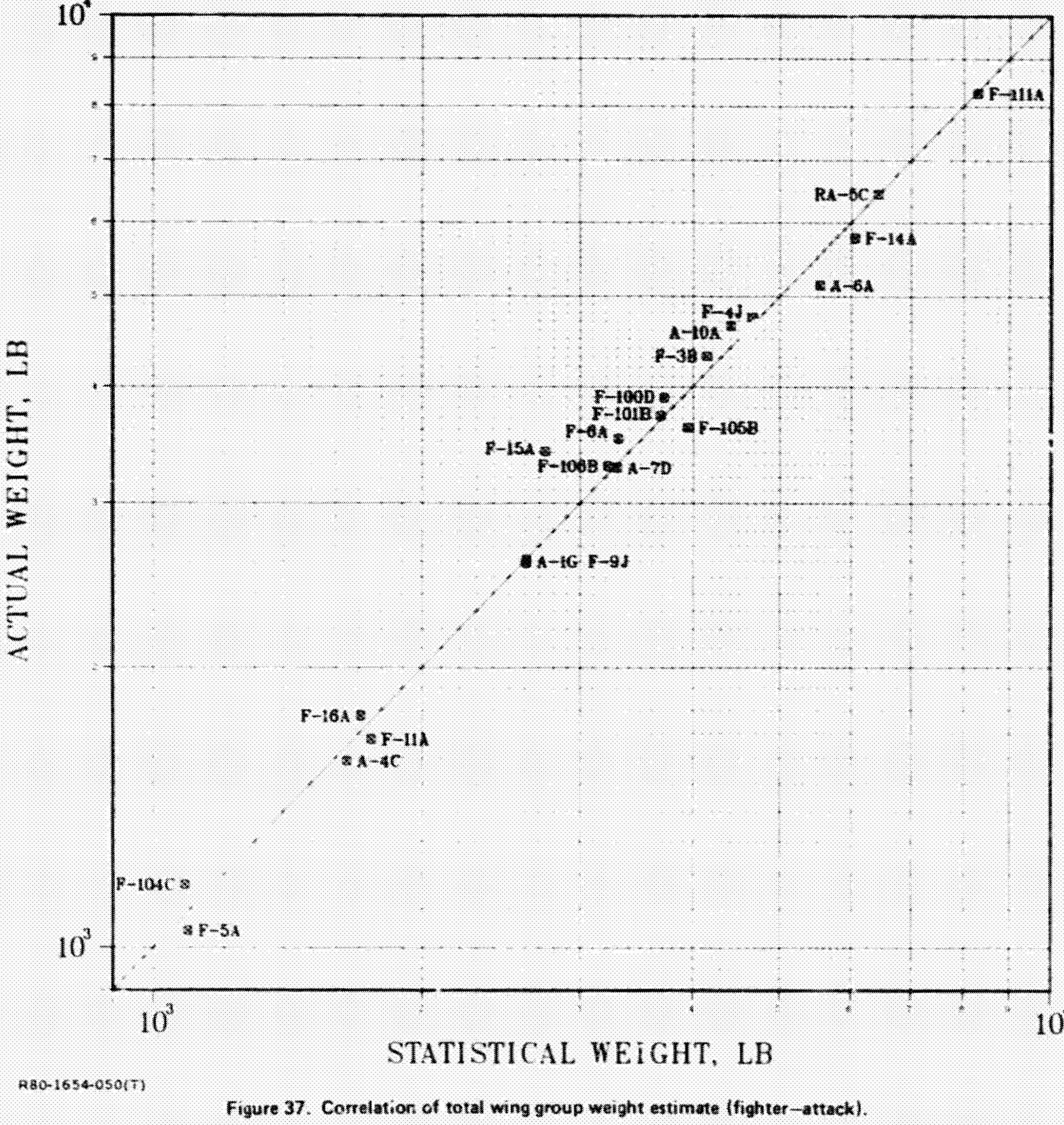
This wing weight estimating method is unique not only in the area of material and construction techniques where a substantial amount of data has been accumulated, but also in the utilization of default values. Default values allow the use of summing type Level II methodology with only Level 0 or Level I input information. This provides a method that is accurate for trending early in the preliminary design phase and leads to continuity later in the design cycle when a more accurate estimate can be obtained by merely upgrading the inputs. This eliminates the problems frequently encountered when having to change methods.

The actual weights of 50 different aircraft (attack, fighter, bomber, transport, anti-submarine, trainer and light utility) were used to develop these formulas which estimate the weights of major components of the Wing Group with a standard deviation of 8.6% for the total wing weight (Figure 36). Figures 37 through 41 show the 50 total wing weights classified by aircraft type.



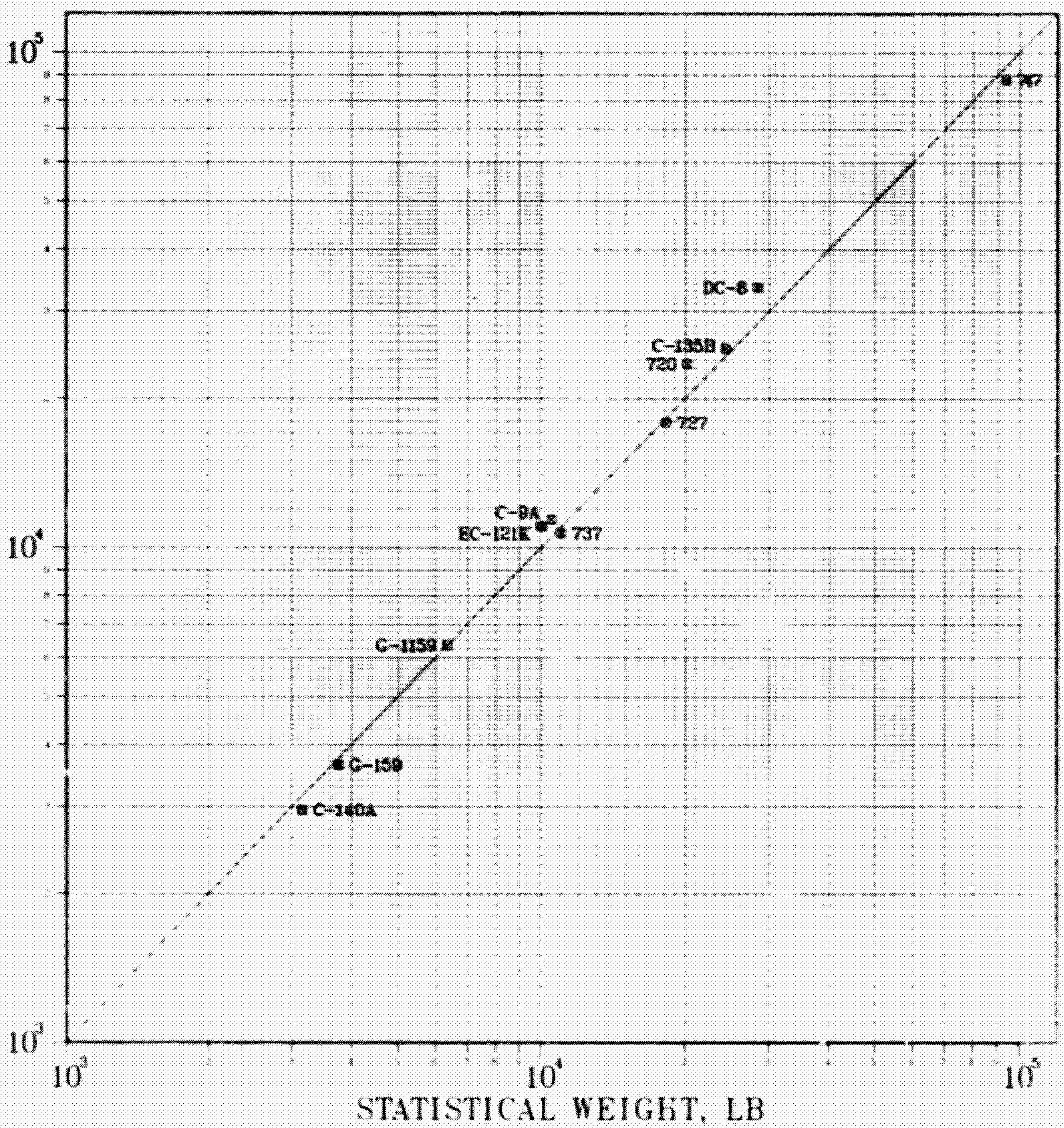
R80-1654-049(T)

Figure 36. Correlation of total wing group weight estimate.



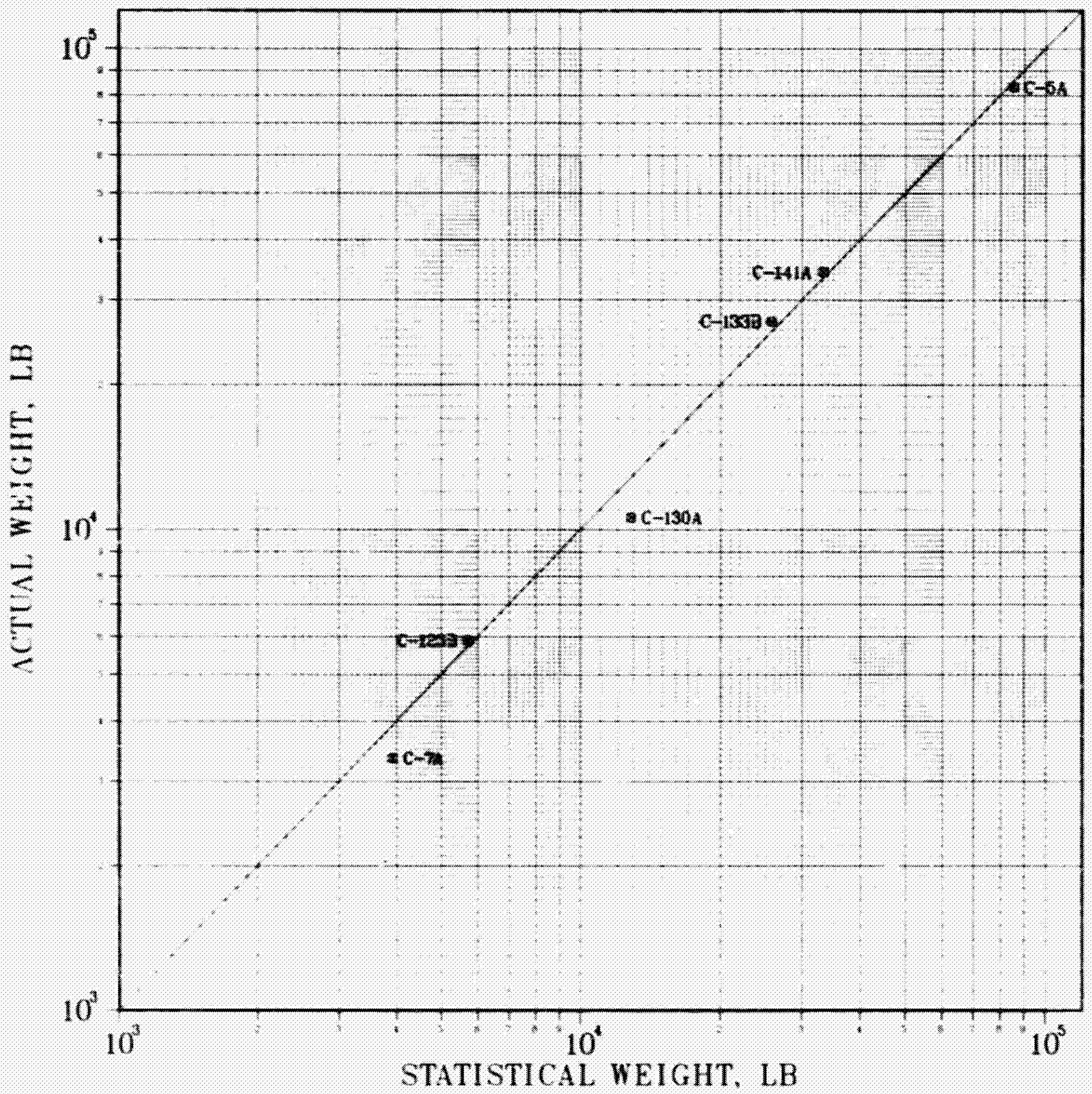
R80-1654-050(T)

Figure 37. Correlation of total wing group weight estimate (fighter–attack).



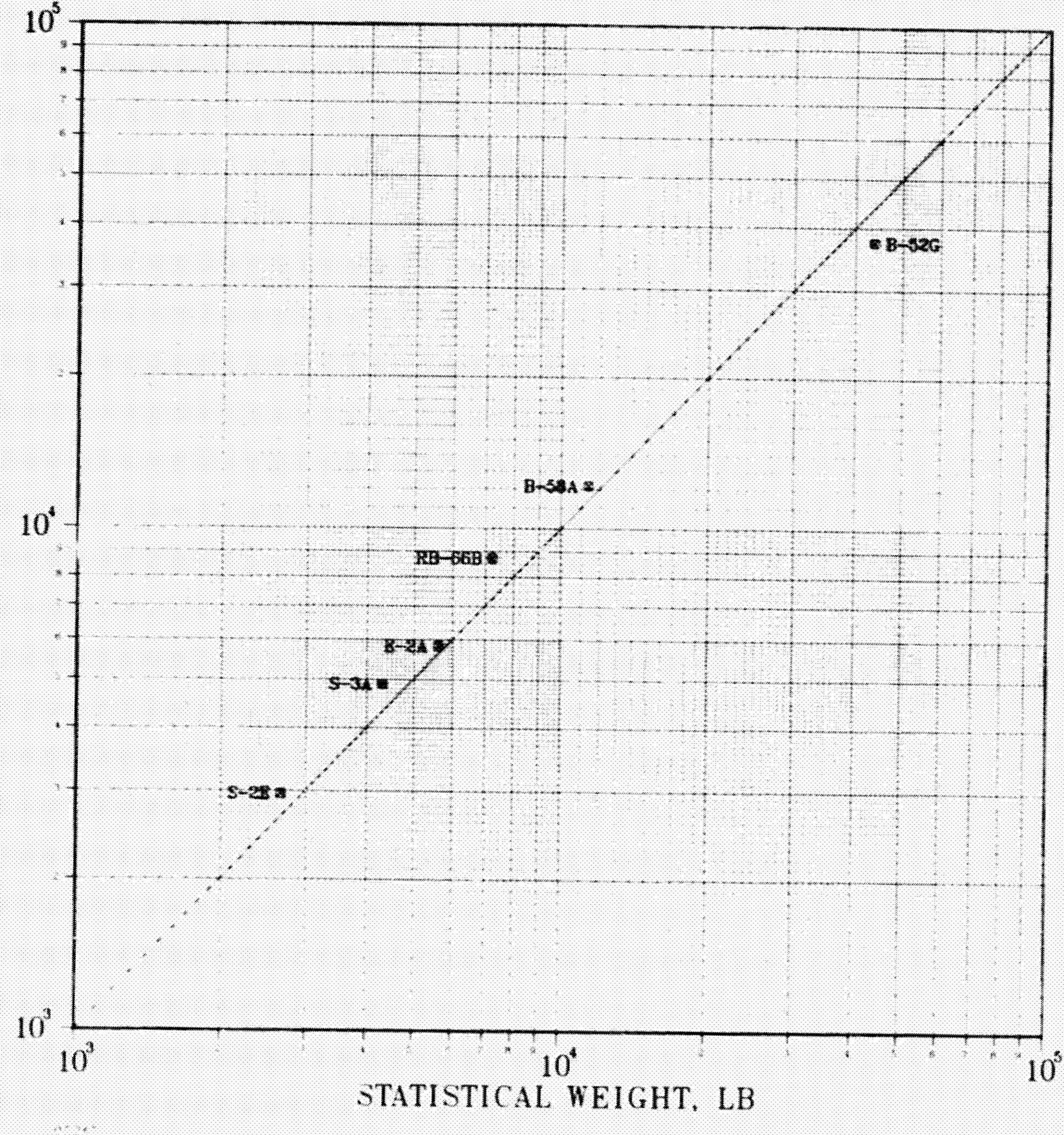
R80-1654-051(T)

Figure 38. Correlation of total wing group weight estimate (transports).

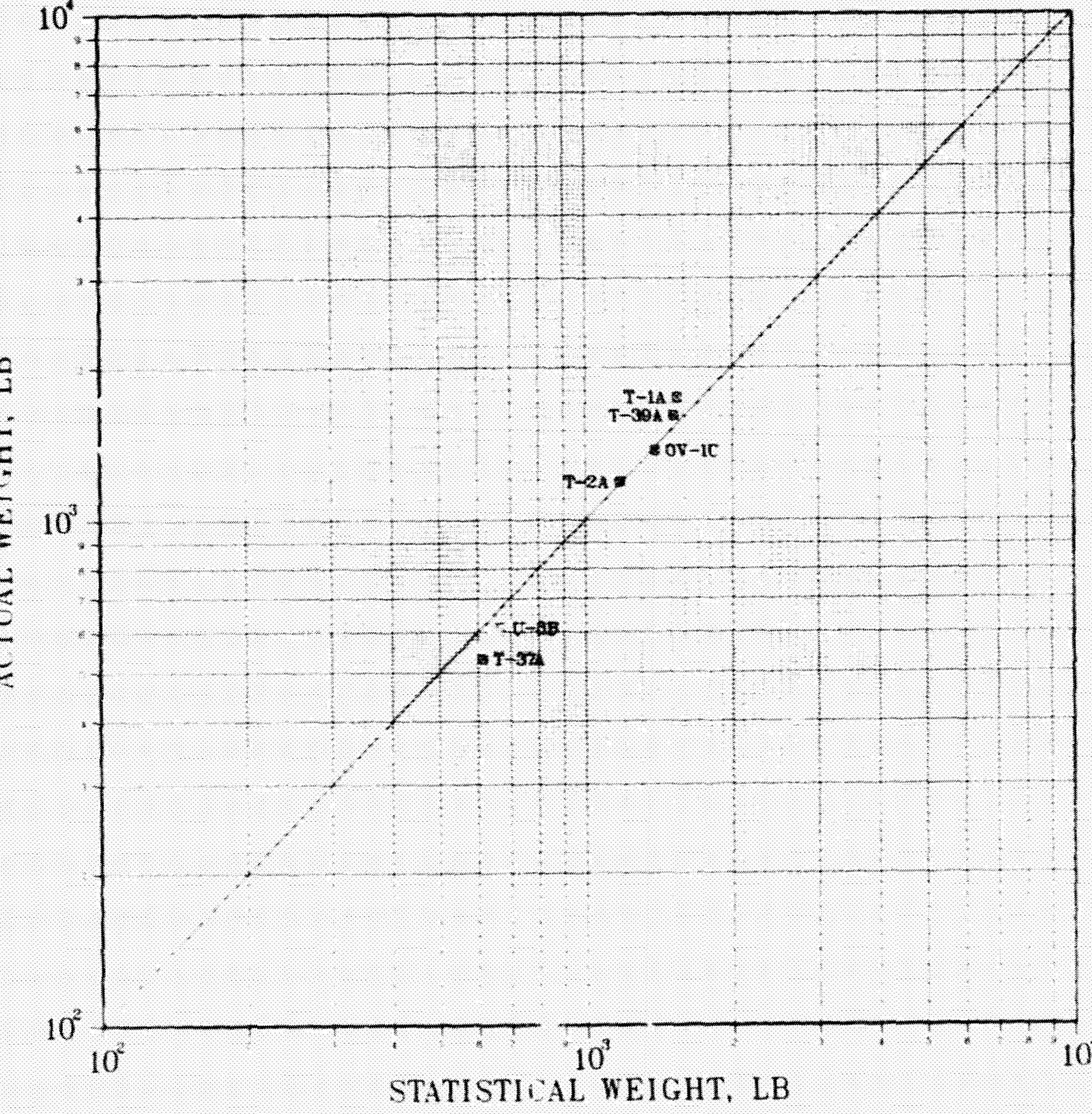


R80-1654-052(T)

Figure 39. Correlation of total wing group weight estimate (cargo).



..gure 40. Correlation of total wing group weight estimate (bombers, AEW, ASW).



R80-1654-054(T)

Figure 41. Correlation of total wing group weight estimate (trainers, etc).

WING GROUP

WEIGHT ESTIMATING METHOD

DEFINITION OF PARAMETERS

S_W	Wing area per airplane, ft^2 (see note *)
S_{BOX}	Wing box area per airplane, ft^2 (see note *)
N_{BOX}	Maximum of, ultimate load factor at FDGW (n), or approximation of ultimate load factor for gust conditions (N_{GUST})
B	Body and contents weight, lb (see note *). defined as: Maximum Clean Gross Weight or Maximum Zero Wing Fuel Weight Less: Wing Group Wing Fuel (Amount in above gross weight) Main landing gear if in the wing Nacelle Group if in the wing Propulsion Group if engines are in the wing Electrical Group if engines are in the wing Oil and Unusuable Fuel if engines are in the wing
C_R	Wing root chord length, in. (see note *)
C_T	Wing tip chord length, in.
T_R	Wing root thickness, in. (see note *)
T_T	Wing tip thickness, in.
b	Wing span (tip to tip), ft (see note *)
$\cos \Lambda$	Cosine of sweep angle of 40% chord
V_L	Limit speed, knots EAS
K_{FSCVR}	1.0 except 1.261 for cover fail-safe design

WING GROUP

WEIGHT ESTIMATING METHOD

DEFINITION OF PARAMETERS (CONTD)

K_{MTLCVR} — 1.0 if all wing covers are baseline material: 7075-T6,'Z' stiffened aluminum, 12 inch rib spacing. For other materials see factors in Appendix C to calculate K_{MTLCVR}, where

$$K_{MTLCVR} = (K_{upper\ center\ section} + K_{lower\ center\ section} + K_{upper\ outer\ panel} + K_{lower\ outer\ panel})/4$$

e.g.,

$K_{upper\ center\ section}$	= 0.893	6-6-2 Titanium 'Integ'
$K_{lower\ center\ section}$	= 0.931	6-6-2 Titanium 'Integ'
$K_{upper\ outer\ panel}$	= 0.976	7075-T6 Aluminum 'Integ'
$K_{lower\ outer\ panel}$	= 1.133	7075-T6 Aluminum 'Integ'

$$K_{MTLCVR} = 3.933/4 = 0.983$$

$K_{TEMPCVR}$ — 1.0 if all wing covers utilize room temperature materials, for other temperatures see factors in Appendix D to calculate $K_{TEMPCVR}$, where

$$K_{TEMPCVR} = (K_{upper\ center\ section} + K_{lower\ center\ section} + K_{upper\ outer\ panel} + K_{lower\ outer\ panel})/4$$

K_{CT} — 1.0 except 2.0 if wing carry-thru is in Body Group weight (see note *)

K_{MTLSUB} — 1.0 for aluminum substructure, 0.787 for titanium substructure

$K_{WSTORES}$ — Summation of heaviest stores weight on all wing stations including drop tanks, lb

WING GROUP

WEIGHT ESTIMATING METHOD

DEFINITION OF PARAMETERS (CONTD)

N_{LDGW}	Ultimate load factor at LDGW
LDGW	Landing Design Gross Weight, lb
K_{MG}	1.0 except 0.5938 if main landing gear are in engine nacelles on wing
W_{WFUEL}	Internal wing fuel weight, lb
F_W	Total thrust of wing mounted engines
HP_W	Total horsepower of wing mounted engines
b'	Folded wing span or pivot span for variable sweep, ft
n	Ultimate load factor at FDGW (for maneuver)
K_{WS}	1.0 except 0.556 for variable sweep wings
TOGW	Take-Off Gross Weight, lb (see note *)
K_{LED}	1.0 except 0.847 for leading edge device
S_{MGDR}	Main landing gear door area, ft^2
S_{ROLL}	Aileron, elevon, flaperon or deceleron area per airplane, ft^2
K_{ROLL}	1.0 except 1.732 for elevons, 1.023 for flaperons, 1.609 for decelerons
K_{BW}	1.0 except 1.541 for ailerons, elevons, flaperons or decelerons with balance weights
S_{FLAP}	Trailing edge flap area per airplane, ft^2

WING GROUP

WEIGHT ESTIMATING METHOD

DEFINITION OF PARAMETERS (CONTD)

$C_{L_{MAX}}$	See $C_{L_{MAX}}$ equation in Default Algorithm section
V_S	Stall speed at LDGW, knots
K_{TS}	1.0 except 1.976 for triple slotted flaps
S_{SLAT}	Slat area per airplane, ft^2
S_{LEF}	Leading edge flap area per airplane, ft^2
S_{SPOIL}	Spoiler area per airplane, ft^2
S_{WSB}	Wing speed brake area per airplane, ft^2
W_{WING}	Total wing weight, lb
N_{GUST}	See N_{GUST} equation in Default Algorithm section
MZWFW	Maximum Zero Wing Fuel Weight, lb
MCGW	Maximum Clean Gross Weight, lb

NOTE

* This method calculates either a total wing weight (center section/carry - thru plus outer panel), or an exposed wing weight (outer panel), consistent with the coding of weight in weight reports. For wings with carry - thru in the Body Group weight, use a K_{CT} = 2.0 and exposed wing values for parameters with asterisks (*)

S^*_W, S^*_{BOX}, T^*_R, C^*_R, b^*

WING GROUP

WEIGHT ESTIMATING METHOD

DEFINITION OF PARAMETERS (CONT'D)

$B^* = (S^*_W / S_W)\ B$

$TOGW^* = (S^*_W / S_W)\ TOGW$

Use full values, for those parameters without asterisks

WING GROUP

WEIGHT ESTIMATING METHOD

BASIC WING BOX COVERS

$$0.039041 \left[\frac{b^* \, (C_R^* + 2C_T) \; B^* \, N_{BOX} \, S_W^*}{\cos^2\Lambda \; (C_R^* + C_T) \; (2T_R^* + T_T) \; (2C_R^* + C_T)} \right]^{0.5074} \left[S^*_{BOX} \right]^{0.5279} \left[V_L \right]^{0.1634} K_{FSCVR} \, K_{MTLCVR} \, K_{TEMPCVR} \quad \text{or}$$

$$0.001552 \left[\frac{(AR)^{1.5} \; (1 + 2\lambda)(1 + \lambda) \; B^* \, N_{BOX} (S_W)^{0.5}}{\cos^2\Lambda \qquad (2 + \lambda) \qquad (2(t/c)_R^* + \lambda (t/c)_T)} \right]^{0.5074} \left[S^*_{BOX} \right]^{0.5279} \left[V_L \right]^{0.1634} K_{FSCVR} \, K_{MTLCVR} \, K_{TEMPCVR}$$

BASIC WING BOX SUBSTRUCTURE

$$[\text{illegible}] \left[S_W^* \; N_{BOX} \, B^* \right]^{0.5598} \left[S^*_{BOX} \, (T_R^* + T_T) \right]^{0.1877} K_{CT}^{0.518} \; K_{MTLSUB} \quad \text{or}$$

$$0.0[\text{illegible}]30 \left[S_W^* \; N_{BOX} \, B^* \right]^{0.5598} \left[\frac{S^*_{BOX} \, (S_W^*)^{0.5}}{(AR)^{0.5} \, (1 + \lambda)} \left(\left(\frac{t}{c}\right)_R^* + \lambda \left(\frac{t}{c}\right)_T \right) \right]^{0.1877} K_{CT}^{0.518} \; K_{MTLSUB}$$

STORES PENALTY TO WING BOX

$0.01 \, [W_{WSTORES}]$

$0.014 \, [W_{WSTORES}]$ (for sweeping store stations, i.e, F-111A)

MAIN LANDING GEAR PENALTY TO WING BOX (No Doors)

$0.001416 \, [N_{LDGW}] \, [LDGW] \; K_{MG}$

WING FUEL PENALTY TO WING BOX

$0.9191 \, [W_{WFUEL}]^{0.5436}$

ENGINE PENALTY TO WING BOX

$0.004 \, [F_W]$ or

$0.03 \, [HP_W]$

WING FOLD OR WING SWEEP PENALTY

$$0.03386 \, [B \, n]^{0.2477} \, [S_W]^{1.244} \left[1 - \frac{b'}{b} \right]^{1.307} K_{WS}$$

WING GROUP

WEIGHT ESTIMATING METHOD

LEADING EDGE, TRAILING EDGE & MISCELLANEOUS

$$0.07235\,[S^*_W - S^*_{B\cdot\,[illegible]}]^{0.2595}\,[TOGW^*]^{0.5281}\,[S^*_W]^{0.3192}\,K_{LED}$$

LANDING GEAR DOORS & MECHANISM

$$0.8991\,[S_{MGDR}]^{1.067}\,[V_L]^{0.2252}$$

AILERONS, ELEVONS, FLAPERONS & DECELERONS

$$0.06564\,[S_{ROLL}]^{0.8697}\left[\frac{TOGW}{S_W}\right]^{1.049} K_{ROLL}\;K_{BW}$$

TRAILING EDGE FLAPS

$$0.0008759\,[S_{FLAP}]\,[V_L]^{0.3565}\,[n]^{0.1576}\,[C_{L_{MAX}}\,LDGW]^{0.3210}\,[V_S]^{0.5}\,K_{TS}$$

SLATS

$$0.2727\,[S_{SLAT}]\,[V_L]^{0.4703}$$

LEADING EDGE FLAPS

$$0.3100\,[S_{LEF}]\,[V_L]^{0.4703}$$

SPOILERS

$$0.2697\,[S_{SPOIL}]^{0.8699}\,[V_L]^{0.3461}\,[S_W]^{0.8445}\,[b]^{-1.117}$$

WING SPEED BRAKES

$$0.01053\,[S_{WSB}]\,[TOGW]^{0.5909}$$

WINGLETS

$$0.0386\,[W_{WING}]$$

C-2

APPENDIX A

INPUTS FOR REGRESSION ANALYSIS

This appendix contains a list of variables, used in the various regression analysis, for the data base of 50 aircraft. In general, values presented are taken directly from dimensional data sheets of the actual weight reports of the particular aircraft; others have been generated from actual values.

APPENDIX A

VARIABLES USED IN REGRESSION ANALYSIS

Aircraft	S_W, ft^2	S^*_W, ft^2	S^*_{BOX}, ft^2	c_R, in.	c^*_R, in.	c_T, in.	T^*_R, in.	T_T, in.	b, ft	b*, ft
1 A-1G	400	400	154.5	128	128	64.5	22	8.4	50	50
2 A-4C	260	260	129.6	186	186	42	14.9	2.1	27.5	27.5
3 A-6A	528.9	528.9	211	182.6	182.6	57	16.4	3.4	53	53
4 A-7D	375	375	187.1	185.9	185.9	46.8	13	3.3	38.7	38.7
5 A-10A	506	506	219.5	132.9	132.9	78.4	22	10.1	57.5	57.5
6 RA-5C	753.7	753.7	351.4	286.6	286.6	53.3	13.2	2.6	53	53
7 B-52G	4000	4000	1554	371	371	148	59.7	11.8	185	185
8 B-58A	1542.5	1542.5	1053.6	651.1	651.1	0	22.2	0	56.9	56.9
9 RB-66B	780	780	321	193.6	193.6	64.9	19.3	5.4	72.5	72.5
10 C-5A	6200	6200	3079	484.2	484.2	184	59.6	19.8	222.7	222.7
11 C-7A	912.4	912.4	554.6	161.3	161.3	67.8	29.5	10.1	95.6	95.6
12 C-9A	1000.7	1000.7	535	213.8	213.8	43.5	26.9	4.1	93.3	93.3
13 C-123B	1223.2	1223.2	522	179.5	179.5	94.8	30.7	8.3	110	110
14 C-130A	1745	1745	987	215.9	215.9	100	40.4	12	132.6	132.6
15 C-133B	2673.1	2673.1	1352	290.9	290.9	66.4	50	10	179.6	179.6
16 C-135B	2433	2433	1322	338	338	112	56.4	10.1	130.8	130.8
17 C-140A	542.5	440	208	180	163.8	60.5	19.7	5	53.7	46.7
18 C-141A	3228.1	3228.1	1767	352.4	352.4	132.7	43	13.3	159.7	159.7
19 EC-121K	1650	1650	567	220	220	102	39.6	12.2	123	123
20 DC-8	2772.5	2772.5	1331	380.8	380.8	87.5	46.3	9.5	142.4	142.4
21 720	2433	2433	1085	336	336	112	52.1	10.1	130.8	130.8
22 727	1695	1695	682	285.1	285.1	91.6	41.8	8.2	108	108
23 737	1106	1106	487	222.1	222.1	63.3	25.2	7.1	93	93
24 747	5849	5849	2904	556.6	556.6	160.8	55.8	12.5	195.7	195.7
25 G-159	609.7	609.7	302	133.8	133.8	53	18.7	7.4	78.3	78.3
26 G-1159	793.5	793.5	424	200	200	76	21.3	6.4	68.9	68.9
27 E-2A	700	700	351	156	156	52	25	7.3	80.6	80.6
28 F-3B	515.8	515.8	186.8	250	250	103	16.9	6.5	35.3	35.3
29 F-4J	538.3	538.3	190	282	282	51.7	18	1.4	38.4	38.4
30 F-5A	173.8	173.8	72.3	134.6	134.6	26.9	6.5	1.3	25.2	25.2
31 F-6A	557	557	354	301	301	100	21.1	4.5	33.5	33.5
32 F-9J	337	337	160	152	152	79	20	6.9	34.5	34.5
33 F-11A	255.3	207	94	126.5	117.2	63.2	6.9	2.5	31.6	27
34 F-14A	565	565	180	167.2	167.2	44.2	21.3	4	64.1	64.1
35 F-15A	627.6	410.2	181.5	291.1	236.5	78	13.9	2	40.8	31.3
36 F-16A	300	197.3	103	195.5	160.7	44.4	6.4	1.8	31	23.1
37 F-100D	400	400	180.8	199.1	199.1	49.8	13.3	3.5	38.6	38.6
38 F-101B	368	368	131.7	173.2	173.2	49.3	11.6	2.8	39.7	39.7
39 F-104C	196.1	130.8	72	155.8	131.5	58.7	4.4	2	21.9	16.5
40 F-105B	385	320	162	180	167.7	84	9.1	3.4	35	30.5
41 F-106B	697.8	498.2	338	427.6	361.2	11.3	14	0.4	38.3	32.1
42 F-111A	525	525	195	150.9	150.9	48.9	17.8	3.4	63	63
43 S-2E	485	485	221	119	119	48	23.8	7.2	69.7	69.7
44 S-3A	598	598	279.8	169	169	42	28.7	5	68	68
45 T-1A	238.1	238.1	125	110	110	41.9	14.3	5.4	37.6	37.6
46 T-2A	254.9	254.9	75.4	114.2	114.2	57.1	13.7	6.9	35.9	35.9
47 T-37A	183.9	139.7	76.6	79.2	73.5	54	12.5	6.4	33.8	26.3
48 T-39A	342	277.3	140.4	139.9	127.5	44.9	14.3	4.2	44.4	38.6
49 U-8B	277.1	277.1	115.9	104.8	104.8	42	15.2	5	45.3	45.3
50 OV-1C	330	330	166	126	126	63	15.1	7.6	42	42

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APPENDIX A - CONTINUED

Aircraft	Cos Λ	n	N_{BOX}	B*, ft	TOGW, lb	TOGW*, lb	V_L, kt	K_{CT}	K_{FS}	V_S, kt
1 A-1G	0.9995	9	9	14,924	19,521	19,521	276	1	1	78.2
2 A-4C	0.886	10.5	10.5	9,350	17,521	17,521	600	1	1	97.5
3 A-6A	0.9262	9.8	9.8	28,690	38,042	38,042	585	1	1	85
4 A-7D	0.8536	10.5	10.5	20,514	31,211	31,211	595	1	1	106
5 A-10A	1	11	11	26,391	30,839	30,839	450	1	1	91.5
6 RA-5C	0.8356	6.8	6.8	35,644	64,091	64,091	730	1	1	106
7 B-52G	0.8307	2.7	5.4	202,600	488,000	488,000	400	1	1	105
8 B-58A	0.7011	3	3.9	76,321	163,000	163,000	600	1	1	150
9 RB-66B	0.8271	5.5	5.5	49,932	70,000	70,000	589	1	1	109
10 C-5A	0.9236	3.7	4	456,478	728,000	728,000	410	1	1	102
11 C-7A	0.9974	5	6.3	14,041	26,000	26,000	208	1	1	59
12 C-9A	0.9252	3.8	4.2	74,910	109,000	109,000	400	1	2	99.3
13 C-123B	0.9998	4.5	5	27,806	54,000	54,000	200	1	1	84
14 C-130A	0.9998	4.5	5.3	57,233	108,000	108,000	321	1	1	79
15 C-133B	0.9995	3.7	3.8	157,961	286,000	286,000	275	1	1	100
16 C-135B	0.8358	3.8	3.8	128,929	277,500	277,500	360	1	2	101
17 C-140A	0.8866	4.5	6.7	16,128	40,148	32,562	497	2	2	99
18 C-141A	0.9336	3.8	5.4	131,110	318,000	318,000	410	1	1	95
19 EC-121K	0.9982	3.7	4.2	52,776	130,000	130,000	250	1	1	87
20 DC-8	0.8851	3.7	5.4	99,843	318,000	318,000	450	1	2	102
21 720	0.8356	3.7	4.9	80,498	203,000	203,000	360	1	2	105
22 727	0.8661	3.8	5.1	92,089	161,000	161,000	460	1	2	94.3
23 737	0.9191	3.8	4.7	59,583	100,800	100,800	420	1	2	94.3
24 747	0.8144	3.8	4	365,187	712,000	712,000	445	1	2	94.3
25 G-159	1	5.3	6.1	14,585	35,100	35,100	341	1	2	86
26 G-1159	0.9215	3.8	7.3	29,958	56,500	56,500	416	1	2	101
27 E-2A	0.9995	5.4	5.9	20,550	49,477	49,477	350	1	1	83
28 F-3B	0.7688	11.2	11.2	25,059	32,037	32,037	635	1	1	94
29 F-4J	0.7619	9.8	9.8	34,127	46,508	46,508	800	1	1	121
30 F-5A	0.9472	9.8	9.8	11,525	13,471	13,471	600	1	1	120
31 F-6A	0.7419	9.4	9.4	12,034	21,342	21,342	625	1	1	101.5
32 F-8J	0.8395	10.5	10.5	15,598	20,198	20,198	630	1	1	96
33 F-11A	0.8384	9.8	9.8	14,852	21,233	17,216	800	2	1	107.6
34 F-14A	0.9732	9.8	9.8	46,373	56,497	56,497	850	1	1	100.8
35 F-15A	0.8316	11	11	20,170	41,809	27,326	850	2	1	100
36 F-16A	0.8S	11	11	12,884	28,730	18,895	794	2	1	105
37 F-100D	0.74	11	11	22,570	34,328	34,328	775	1	1	115
38 F-101B	0.8328	10.2	10.2	38,739	41,442	41,442	760	1	1	143
39 F-104C	0.9767	11	11	12,338	19,665	13,117	835	2	1	125
40 F-105B	0.7318	13	13	24,193	34,483	28,661	843	2	1	120
41 F-106B	0.7042	9	9	17,233	35,637	25,443	835	2	1	136
42 F-111A	0.971	11	11	67,778	80,977	80,977	873	1	1	96
43 S-2E	0.9989	4.5	4.5	10,470	26,700	26,700	225	1	1	87
44 S-3A	0.9764	5.2	5.2	22,394	41,614	41,614	440	1	1	83
45 T-1A	0.9989	11	11	9,775	13,338	13,338	460	1	1	89
46 T-2A	1	11.2	11.2	8,323	10,092	10,092	400	1	1	67
47 T-37A	0.9998	10	10	3,176	6,436	4,889	382	2	1	68
48 T-39A	0.8974	6	8.1	7,161	16,117	13,066	400	2	1	78
49 U-8B	0.9997	6.6	7.7	2,832	6,000	6,000	210	1	1	58
50 OV-1C	0.9993	7.4	7.4	7,611	12,708	12,708	390	1	1	63.5

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Aircraft		LDGW, lb	N_{LDGW}	K_{MG}	$W_{WSTORES}$, lb	K_{STORES}	W_{WFUEL}, lb	F_W	HP_W	b', ft	K_{WS}
1	A-1G	17,000	8.0	1	8,000	1	–	–	–	23.9	1
2	A-4C	11,556	7.2	1	4,000	1	3,808	–	–	27.5	–
3	A-6A	33,637	6.5	1	–	–	6,923	–	–	25.2	1
4	A-7D	32,251	–	–	19,000	1	4,927	–	–	23.8	1
5	A-10A	32,334	4.5	1	11,143	1	–	–	–	57.5	–
6	RA-5C	45,000	–	–	–	–	9,724	–	–	42.0	1
7	B-52G	270,000	–	–	–	–	–	–	–	185.0	–
8	B-58A	95,000	3.5	1	–	–	–	–	–	56.9	–
9	RB-66B	50,000	–	–	–	–	–	–	–	72.5	–
10	C-5A	635,850	–	–	–	–	318,500	164,400	–	222.7	–
11	C-7A	26,000	3.2	0.5938	–	–	–	–	–	95.6	–
12	C-9A	99,000	3.6	1	–	–	–	–	–	93.3	–
13	C-123B	51,350	–	–	–	–	–	–	–	110.0	–
14	C-130A	96,000	–	–	–	–	–	–	–	132.6	–
15	C-133B	250,500	–	–	–	–	–	–	–	179.6	–
16	C-135B	200,000	3.0	1	–	–	–	–	–	130.8	–
17	C-140A	30,000	2.4	1	–	–	–	–	–	53.7	–
18	C-141A	257,500	–	–	–	–	–	–	–	159.7	–
19	EC-121K	110,000	3.8	0.5938	–	–	–	–	–	123.0	–
20	DC-8	207,000	3.8	1	–	–	–	–	–	142.4	–
21	720	165,000	–	–	–	–	–	–	–	130.8	–
22	727	137,500	3.0	1	–	–	–	–	–	108.0	–
23	737	98,000	3.0	1	–	–	–	–	–	93.0	–
24	747	564,000	3.0	0.5938	–	–	–	–	–	195.7	–
25	G-159	33,600	4.2	0.5938	–	–	–	–	–	78.3	–
26	G-1159	51.430	4.5	1	–	–	–	–	–	68.9	–
27	E-2A	40,660	7.7	0.5938	–	–	12,133	–	–	29.3	1
28	F-3B	23,500	8.1	1	–	–	1,989	–	–	25.3	1
29	F-4J	33,500	7.2	1	7,000	1	4,284	–	–	27.5	1
30	F-5A	12,200	4.0	1	–	–	–	–	–	25.2	–
31	F-6A	16,559	8.1	1	6,400	1	4,160	–	–	25.5	1
32	F-8J	14,969	9.2	1	–	–	–	–	–	15.7	1
33	F-11A	15,100	–	–	–	–	1,242	–	–	27.3	1
34	F-14A	51,830	6.6	1	1,970	1	6,732	–	–	17.8	2
35	F-15A	35,000	4.1	1	9,240	1	5,500	–	–	40.8	–
36	F-16A	19,500	–	–	–	–	–	–	–	31.0	–
37	F-100D	24,354	4.5	1	–	–	–	–	–	38.6	–
38	F-101B	33,500	3.9	1	–	–	1,203	–	–	39.7	–
39	F-104C	16,000	–	–	5,400	1	–	–	–	21.9	–
40	F-105B	29,227	4.8	1	8,000	1	–	–	–	35.0	–
41	F-106B	28,060	4.5	1	–	–	8,281	–	–	38.3	–
42	F-111A	70,000	–	–	24,160	2	–	–	–	11.7	2
43	S-2E	23,713	6.8	0.5938	–	–	–	–	3,050	27.3	1
44	S-3A	37,695	–	–	–	–	13,142	18,550	–	29.5	1
45	T-1A	14,500	7.0	1	–	–	1,001	–	–	37.6	–
46	T-2A	9.507	6.5	1	1.500	1	–	–	–	35.9	–
47	T-37A	5,713	6.0	1	–	–	–	–	–	33.8	–
48	T-38A	13,000	5.6	1	–	–	5,805	–	–	44.4	–
49	U-8B	6,000	5.5	0.5938	–	–	–	–	–	45.3	–
50	OV-1C	10,924	8.2	0.5938	–	–	–	–	2,010	42.0	–

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APPENDIX A - CONTINUED

	Aircraft	K_{LED}	S_{MGDR}, ft²	S_{ROLL}, ft²	K_{ROLL} (ELVN)	K_{ROLL} (FLPN)	K_{ROLL} (DCLN)	K_{ABW}	S_{FLAP}, ft²	K_{TS}	S_{LED}, ft²
1	A-1G	1	–	43.5	1	1	1	2	44.7	1	–
2	A-4C	1	[7.0]	23.9	1	1	1	2	22.2	1	19.6
3	A-6A	2	49.0 [9.8]	41	1	2	1	1	104	1	49.8
4	A-7D	2	(42.7)[8.6]	19.9	1	1	1	1	43.5	1	37.2
5	A-10A	2	14.6 [9.2]	86.8	1	1	2	2	81.3	1	10.6
6	RA-5C	2	(43.0)[12.0]	–	–	–	–	–	91.7	1	46.6
7	B-52G	1	–	–	–	–	–	–	797	1	–
8	B-58A	1	–	177.8	2	1	1	1	–	–	–
9	RB-66B	2	–	32.6	1	1	1	2	108	1	73.9
10	C-5A	2	–	252.8	1	1	1	2	991.7	1	648.6
11	C-7A	1	–	91	1	1	1	2	194	1	–
12	C-9A	2	–	38	1	1	1	1	210.8	1	160.0
13	C-123B	1	(49.9)[26.2]	83.3	1	1	1	2	128	1	–
14	C-130A	1	–	110	1	1	1	2	342	1	–
15	C-133B	1	–	144.3	1	1	1	2	496.5	1	–
16	C-135B	2	–	119.6	1	1	1	2	362	1	26.1
17	C-140A	2	–	24.4	1	1	1	2	62.6	1	34.0
18	C-141A	1	–	171.1	1	1	1	2	528.7	1	–
19	EC-121K	1	–	99.6	1	1	1	2	259.2	1	–
20	DC-8	2	–	161.6	1	1	1	2	456.9	1	25.0
21	720	2	–	119.6	1	1	1	2	361.6	1	95.4
22	727	2	–	57	1	1	1	2	281	2	200.0
23	737	2	–	26.9	1	1	1	2	180.3	2	100.6
24	747	2	–	222	1	1	1	2	847	2	448.0
25	G-159	1	[9.5]	36.6	1	1	1	2	110.8	1	–
26	G-1159	1	(43.2)[13.8]	28.9	1	1	1	2	128.8	1	–
27	E-2A	1	[8.0]	62	1	1	1	2	122	1	–
28	F-3B	2	26.1	31.2	1	1	1	2	30.5	1	66.9
29	F-4J	2	–	26.2	1	1	1	1	29.2	1	25.8
30	F-5A	2	9.9	9.2	1	1	1	1	19	1	12.3
31	F-8A	1	25.2 [7.0]	49.7	2	1	1	2	–	–	15.5
32	F-9J	1	(9.0) [7.2]	18.5	1	2	1	1	72.8	1	–
33	F-11A	2	(23.6)[8.2]	21.3	1	2	1	1	35.8	1	16.8
34	F-14A	2	(41.7)[14.0]	–	–	–	–	–	111.1	1	45.6
35	F-15A	1	(31.6)[9.3]	26.5	1	1	1	1	35.8	1	–
36	F-16A	2	–	31.3	1	2	1	1	–	1	36.7
37	F-100D	1	–	37.1	1	1	1	2	31.3	1	47.1
38	F-101B	1	14.5 [9.1]	29.6	1	1	1	2	42	1	–
39	F-104C	2	[5.8]	9.5	1	1	1	1	23	1	17.0
40	F-105B	2	36.5 [8.9]	15.4	1	1	1	2	61.4	1	22.8
41	F-106B	1	(21.7)[7.5]	66.6	2	1	1	1	–	–	–
42	F-111A	2	–	–	–	–	–	–	117.8	1	60.7
43	S-2E	1	[9.5]	9.3	1	1	1	2	92.7	1	14.0
44	S-3A	2	(58.4)	13.3	1	1	1	2	111.6	1	50.0
45	T-1A	2	20.8 [7.8]	17.5	1	1	1	2	22.1	1	16.8
46	T-2A	1	[4.4]	19	1	1	1	2	50.5	1	–
47	T-37A	1	–	11.3	1	1	1	2	15.1	1	–
48	T-38A	1	–	16.4	1	1	1	2	40.3	1	36.3
49	U-8B	1	–	13.9	1	1	1	2	37.8	1	–
50	OV-1C	2	[7.9]	40.2	1	1	1	2	43.6	1	16.8

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APPENDIX A - CONCLUDED

Aircraft	S_{SPOIL}, ft^2	S_{WSB}, ft^2	MZWFW, lb							
1 A-1G	–	–	–							
2 A-4C	–	–	–							
3 A-6A	–	16.8	–							
4 A-7D	9.2	–	–							
5 A-10A	–	–	–							
6 RA-5C	49.6	–	–							
7 B-52G	148.0	–	170,000							
8 B-58A	–	–	115,322							
9 RB-66B	17.0	–	63,000							
10 C-5A	430.7	–	588,904							
11 C-7A	–	–	26,000							
12 C-9A	41.5	–	90,000							
13 C-123B	–	–	45,228							
14 C-130A	–	–	97,210							
15 C-133B	–	–	215,000							
16 C-135B	107.2	–	197,000							
17 C-140A	–	–	26,300							
18 C-141A	311.0	–	199,776							
19 EC-121K	–	–	99,500							
20 DC-8	48.5	–	179,000							
21 720	104.0	–	136,760							
22 727	114.5	–	116,000							
23 737	70.7	–	85,000							
24 747	304.0	–	526,500							
25 G-159	–	–	–							
26 G-1159	49.4	–	–							
27 E-2A	–	–	35,807							
28 F 3B	14.6	–	–							
29 F-4J	10.9	18.6	–							
30 F-5A	–	–	–							
31 F-6A	–	10.0	–							
32 F-8J	–	8.5	–							
33 F-11A	–	–	–							
34 F-14A	30.0	–	–							
35 F-15A	–	–	–							
36 F-16A	–	–	–							
37 F-100D	–	–	–							
38 F-101B	–	–	–							
39 F-104C	–	–	–							
40 F-105B	18.7	–	–							
41 F-106B	–	–	–							
42 F-111A	28.6	–	–							
43 S-2E	12.6	–	24,435							
44 S-3A	66.3	–	–							
45 T-1A	–	–	–							
46 T-2A	–	–	–							
47 T-37A	–	–	–							
48 T-38A	–	–	10,896							
49 U-8B	–	–	5,760							
50 OV-1C	–	–	13,378							
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APPENDIX B

MATERIAL/CONSTRUCTION DATA

Complete material/construction data is presented on 22 aircraft and is used in obtaining the factor K_{MTLCVR}. Partial information collected on a number of other aircraft is also included for reference. The majority of Phase I of the study effort was expended in this area since the depth of detail required (type of alloy, stiffener spacing, rib and beam spacing, construction and design philosophy) was not readily available.

APPENDIX B

MATERIAL AND CONSTRUCTION DATA - CENTER SECTION

Aircraft	Upper cover Material	Upper cover Construction	Upper cover Stiffener spacing, inches	Lower cover Material	Lower cover Construction	Lower cover Stiffener spacing, inches	Beam spacing, inches	Rib spacing, inches
1 A-1G								
2 A-4C	7075-T6	'Z'	4.0	7075-T6	'Z'		—	11.0
3 A-6A	7079-T651	Integ	4.5	7079-T651	Integ		—	15.0
4 A-7D	7079-T651	Thick skin	—	7075-T651	Thick skin	—	16.4	34.2
5 A-10A	7075-T651	Integ	4.1	2024-T351	Integ		—	22.0
6 RA-5C								
7 B-52G								
8 B-58A		Honeycomb						
9 RB-66B								
10 C-5A								
11 C-7A								
12 C-9A								
13 C-123B								
14 C-130A	7178-T6	Integ	6.6	7075-T6	'Hat'			
15 C-133B								
16 C-135B								
17 C-140A	Coded to body	—	—	—	—	—	—	—
18 C-141A								
19 EC-121K								
20 DC-8		'Y'						
21 720								
22 727	7178-T651	'Z'	5.2	2024-T351	'Z'		—	26.5
23 737	7075-T6	'Z'	5.25	2024-T351	'Z'		—	25.0
24 747	7075-T6	'Z'	5.25	2024-T351	'Z'		—	25.0
25 G-159	7075-T6	Integ	2.0	7075-T6	Integ		—	14.0
26 G-1159	7075-T651	Integ	4.0	2024-T351	Integ		—	14.0
27 E-2A	7075-T651	Integ	4.0	7075-T651	Integ		—	17.0
28 F-3B		'Hat'						
29 F-4J	7075-T651	Thick skin	—	7075-T651	Thick skin	—	8.3	40.0
30 F-5A								
31 F-6A								
32 F-8J								
33 F-11A	Coded to body	—	—	—	—	—	—	—
34 F-14A	6-4	Integ	4.0	6-4	Integ		—	11.0
35 F-15A	Coded to body	—	—	—	—	—	—	—
36 F-16A	Coded to body	—	—	—	—	—	—	—
37 F-100D								
38 F-101B								
39 F-104C	Coded to body	—	—	—	—	—	—	—
40 F-105B	Coded to body	—	—	—	—	—	—	—
41 F-106B	Coded to body	—	—	—	—	—	—	—
42 F-111A	D6AC	Thick skin	—	D6AC	Thick skin	—	24.0	7.0
43 S-2E	7075-T6	'Hat'	3.5	7075-T6	'Hat'		—	7.0
44 S-3A	7075-T7651	Integ	3.1	7075-T7651	Integ		—	20.0
45 T-1A								
46 T-2A								
47 T-37A	Coded to body	—	—	—	—	—	—	—
48 T-39A	Coded to body	—	—	—	—	—	—	—
49 U-8B								
50 OV-1C	7075-T6	'Z'	6.0	7075-T6	'Z'		—	14.0

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APPENDIX B

MATERIAL AND CONSTRUCTION DATA - OUTER PANEL

Aircraft	Upper cover			Lower cover			Beam spacing, inches	Rib spacing, inches
	Material	Construction	Stiffener spacing, inches	Material	Construction	Stiffener spacing, inches		
1 A-1G								
2 A-4C	7075-T6	'Z'	4.0	7075-T6	'Z'		–	13.9
3 A-6A	7075-T651	Integ	5.0	7075-T651	Integ		–	25.0
4 A-7D	7079-T651	Thick skin	–	7075-T651	Thick skin	–	12.6	23.1
5 A-10A	7075-T6	'Z'	5.25	2024-T3	'Z'		–	15.7
6 RA-5C								
7 B-52B								
8 B-58A								
9 RB-66B								
10 C-5A								
11 C-7A								
12 C-9A								
13 C-123B								
14 C-130A								
15 C-133B								
16 C-135B								
17 C-140A								
18 C-141A	7075-T651	Integ	4.1	7075-T651	Integ		–	23.0
19 EC-121K								
20 DC-8		'Y'						
21 720								
22 727	7178-T651	'Z'	4.5	2024-T351	'Z'		–	26.5
23 737	7075-T6	'Z'	4.5	2024-T351	'Z'		–	25.0
24 747	7075-T6	'Z'	4.5	2024-T351	'Z'		–	25.0
25 G-159	7075-T6	Integ	2.0	7075-T6	Integ		–	14.0
26 G-1159	7075-T651	Integ	4.0	2024-T351	Integ		–	16.0
27 E-2A	7075-T6	'Z'	2.6	7075-T6	'Z'		–	13.0
28 F-3B								
29 F-4J	7178-T651	Thick skin	–	7178-T651	Thick skin	–	11.4	11.1
30 F-5A								
31 F-8A								
32 F-9J								
33 F-11A	7075-T6	Thick skin	–	7075-T6	Thick skin	–	9.0	22.0
34 F-14A	6-6-2	'Y'	4.0	6-4	'Z'		–	14.0
35 F-15A	2024-T851	Integ	4.5	6-4	Integ		–	18.0
36 F-16A	2124-T851	Thick skin	–	7475-T73	Thick skin	–	8.3	35.0
37 F-100D								
38 F-101B	7178-T6	Thick skin	–	2024-T4	Thick skin	–		46.0
39 F-104C								
40 F-105B	7075-T6	Thick skin	–	7075-T6	Thick skin	–	28.4	35.0
41 F-106B	7075-T6	Thick skin	–	7075-T6	Thick skin	–	46.0	7.7
42 F-111A	2024-T851	Thick skin	–	2024-T851	Thick skin	–	13.9	58.4
43 S-2E	7075-T6	'Hat'	3.5	2014-T6	'Hat'		–	15.0
44 S-3A	7075-T7651	Integ	3.3	7075-T7651	Integ		–	21.0
45 T-1A								
46 T-2A	7075-T6	'Z'	6.0				–	9.0
47 T-37A	2024-T3	'Hat'		2024-T3	'Hat'		–	24.0
48 T-39A								
49 U-8B								
50 OV-1C	7075-T6	'Z'	6.0	7075-T6	'Z'		–	14.0

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APPENDIX C

MATERIAL/CONSTRUCTION FACTORS

This appendix contains the cover material/construction factors (K_{MTLCVR}) generated by a multiple station analysis program. These factors allow the center section upper and lower covers, and the outer panel upper and lower covers to be different materials/constructions. Materials avaliable are:

- Aluminum - 7075-T6
- Titanium - 6Al-6V-2Sn Ann.
- Steel - PH15-7MO
- Graphite/Epoxy

Constructions available are:

- 'Z' Stiffened Sheet (Multi-rib design)
- 'HAT' Stiffened Sheet (Multi-rib design)
- 'Y' Stiffened Sheet (Multi-rib design)
- Integrally Stiffened Sheet (Multi-rib design)
- Flat/Thick Sheet (Multi-spar design)

APPENDIX C

MATERIAL/CONSTRUCTION FACTORS - UPPER COVER

Material: Aluminum 7075-T6 (R.T.) – Baseline

Limit load factor	Construction: 'Z' stiff.			Construction: Hat stiff.		
	Rib spacing, inches			Rib spacing, inches		
	12	16	20	12	16	20
2.5	1.000	1.020	1.053	0.925	0.948	0.994
3.0	1.000	1.017	1.050	0.918	0.947	0.990
4.0	1.000	1.021	1.061	0.916	0.950	0.997
5.0	1.000	1.025	1.071	0.923	0.958	1.004
6.5	1.000	1.033	1.088	0.940	0.981	1.030
7.0	1.000	1.036	1.093	0.944	0.996	1.039
7.5	1.000	1.035	1.091	0.953	1.002	1.044
Limit load factor	**Construction: 'Y' stiff.**			**Construction: Integ. stiff.**		
	Rib spacing, inches			Rib spacing, inches		
	12	16	20	12	16	20
2.5	0.982	0.985	0.990	0.903	0.943	1.004
3.0	0.962	0.966	0.977	0.918	0.956	1.014
4.0	0.943	0.946	0.971	0.953	0.994	1.048
5.0	0.934	0.947	0.974	0.976	1.025	1.082
6.5	0.934	0.953	0.994	0.997	1.054	1.116
7.0	0.934	0.959	1.002	1.002	1.065	1.127
7.5	0.936	0.960	1.006	1.007	1.067	1.131
Limit load factor	**Construction: Flat sheet**					
	Spar spacing, inches					
	6	9	12			
2.5	1.410	1.737	2.086			
3.0	1.350	1.666	2.004			
4.0	1.274	1.563	1.884			
5.0	1.215	1.480	1.786			
6.5	1.143	1.378	1.663			
7.0	1.122	1.349	1.625			
7.5	1.099	1.317	1.586			

R80-1654-016(T)

Material: Titanium 6AL-6V-2SN Ann. (R.T.)

Limit load factor	Construction: 'Z' stiff.			Construction: Hat stiff.		
	Rib spacing, inches			Rib spacing, inches		
	12	16	20	12	16	20
2.5	1.185	1.197	1.239	1.149	1.170	1.208
3.0	1.156	1.175	1.213	[illegible]	1.128	1.164
4.0	1.103	1.155	1.194	1.056	1.074	1.114
5.0	1.068	1.143	1.179	1.019	1.033	1.080
6.5	1.037	1.111	1.169	0.970	0.985	1.031
7.0	1.027	1.099	1.164	0.958	0.976	1.016
7.5	1.014	1.080	1.157	0.943	0.963	1.005
Limit load factor	Construction: 'Y' stiff.			Construction: Integ. stiff.		
	Rib spacing, inches			Rib spacing, inches		
	12	16	20	12	16	20
2.5	1.308	1.308	1.309	0.909	0.974	1.097
3.0	1.246	1.246	1.247	0.894	0.941	1.061
4.0	1.166	1.166	1.168	0.888	0.925	1.022
5.0	1.110	1.110	1.113	0.893	0.931	1.005
6.5	1.047	1.048	1.054	0.916	0.939	0.992
7.0	1.030	1.031	1.038	0.918	0.943	0.994
7.5	1.014	1.014	1.021	0.918	0.945	0.991
Limit load factor	Construction: Flat sheet					
	Spar spacing, inches					
	6	9	12			
2.5	1.971	2.394	2.859			
3.0	1.882	2.293	2.742			
4.0	1.751	2.144	2.570			
5.0	1.645	2.023	2.430			
6.5	[illegible]	1.867	2.249			
7.0	[illegible]	1.820	2.195			
7.5	1.431	1.772	2.137			

R80-1654-016(T)

Material: Stainless Steel PH15-7MO (R.T.)

Limit load factor	Construction: 'Z' stiff.			Construction: Hat stiff.		
	Rib spacing, inches			Rib spacing, inches		
	12	16	20	12	16	20
2.5	1.645	1.655	1.693			
3.0	1.595	1.600	1.646			
4.0	1.525	1.535	1.599			
5.0	1.494	1.504	1.570			
6.5	1.453	1.470	1.525			
7.0	1.444	1.462	1.512			
7.5	1.430	1.448	1.498			
Limit load factor	Construction: 'Y' stiff.			Construction: Integ. stiff.		
	Rib spacing, inches			Rib spacing, inches		
	12	16	20	12	16	20
2.5				1.413	1.485	1.619
3.0				1.357	1.440	1.552
4.0				1.311	1.388	1.497
5.0				1.282	1.365	1.473
6.5				1.272	1.350	1.453
7.0				1.270	1.352	1.450
7.5				1.270	1.343	1.444
Limit load factor	Construction: Flat sheet					
	Spar spacing, inches					
	6	9	12			
2.5	2.900	3.478	4.122			
3.0	2.764	3.330	3.950			
4.0	2.567	3.110	3.703			
5.0	2.411	2.929	3.530			
6.5	2.213	2.699	3.233			
7.0	2.154	2.630	3.152			
7.5	2.091	2.559	3.069			

R80-1654-016(T)

Material: Graphite/Epoxy with Holes (R.T.)

Limit load factor	Construction: 'Z' stiff.			Construction: Hat stiff.		
	Rib spacing, inches			Rib spacing, inches		
	12	16	20	12	16	20
2.5						
3.0						
4.0						
5.0						
6.5						
7.0						
7.5						

Limit load factor	Construction: 'Y' stiff.			Construction: Integ. stiff.		
	Rib spacing, inches			Rib spacing, inches		
	12	16	20	12	16	20
2.5						
3.0						
4.0						
5.0						
6.5						
7.0						
7.5						

Limit load factor	Construction: Flat sheet					
	Spar spacing, inches					
	6	9	12			
2.5	0.873	1.032	1.225			
3.0	0.852	1.000	1.189			
4.0	0.805	0.958	1.135			
5.0	0.789	0.933	1.093			
6.5	0.782	0.890	1.037			
7.0	0.790	0.882	1.008			
7.5	0.796	0.866	1.006			

:654-016(T)

APPENDIX C

MATERIAL/CONSTRUCTION FACTORS - LOWER COVER

Material: Aluminum 7075-T6 (R.T.) – Baseline

Limit load factor	Construction: 'Z' stiff.			Construction: Hat stiff.		
	Rib spacing, inches			Rib spacing, inches		
	12	16	20	12	16	20
2.5	1.000	1.004	1.015	1.022	1.044	1.055
3.0	1.000	0.998	1.009	1.019	1.042	1.054
4.0	1.000	1.004	1.010	1.033	1.064	1.083
5.0	1.000	1.002	1.022	1.058	1.094	1.105
6.5	1.000	1.010	1.035	1.071	1.114	1.137
7.0	1.000	1.007	1.036	1.065	1.126	1.141
7.5	1.000	1.006	1.036	1.072	1.129	1.143

Limit load factor	Construction: 'Y' stiff.			Construction: Integ. stiff.		
	Rib spacing, inches			Rib spacing, inches		
	12	16	20	12	16	20
2.5	1.098	1.098	1.100	1.025	1.045	1.064
3.0	1.086	1.087	1.091	1.054	1.064	1.087
4.0	1.080	1.085	1.092	1.107	1.114	1.130
5.0	1.084	1.089	1.096	1.133	1.158	1.172
6.5	1.083	1.080	1.131	1.142	1.171	1.185
7.0	1.077	1.090	1.135	1.144	1.178	1.196
7.5	1.075	1.094	1.141	1.148	1.171	1.185

Limit load factor	Construction: Flat sheet		
	Spar spacing, inches		
	6	9	12
2.5	1.532	1.851	2.213
3.0	1.434	1.741	2.083
4.0	1.292	1.563	1.872
5.0	1.211	1.418	1.702
6.5	1.137	1.249	1.496
7.0	1.121	1.207	1.438
7.5	1.100	1.165	1.383

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Material: Titanium 6AL-6V-2SN Ann. (R.T.)

Limit load factor	Construction: 'Z' stiff.			Construction: Hat stiff.		
	Rib spacing, inches			Rib spacing, inches		
	12	16	20	12	16	20
2.5	1.203	1.214	1.241	1.238	1.254	1.302
3.0	1.137	1.147	1.175	1.158	1.179	1.220
4.0	1.039	1.047	1.080	1.041	1.062	1.096
5.0	0.965	0.971	1.001	0.979	0.993	1.014
6.5	0.912	0.918	0.929	0.930	0.942	0.961
7.0	0.898	0.904	0.913	0.921	0.927	0.951
7.5	0.884	0.889	0.899	0.905	0.915	0.941

Limit load factor	Construction: 'Y' Stiff.			Construction: Integ. stiff.		
	Rib spacing, inches			Rib spacing, inches		
	12	16	20	12	16	20
2.5	1.495	1.495	1.495	1.037	1.089	1.192
3.0	1.382	1.382	1.382	0.988	1.029	1.116
4.0	1.213	1.213	1.213	0.946	0.973	1.016
5.0	1.094	1.094	1.094	0.931	0.952	0.987
6.5	0.997	0.997	0.997	0.935	0.946	0.977
7.0	0.977	0.977	0.977	0.934	0.973	0.977
7.5	0.957	0.957	0.957	0.931	0.948	0.972

Limit load factor	Construction: Flat sheet		
	Spar spacing, inches		
	6	9	12
2.5	2.153	2.566	3.036
3.0	2.013	2.408	2.854
4.0	1.789	2.153	2.558
5.0	1.610	1.946	2.319
6.5	1.393	1.694	2.026
7.0	1.334	1.625	1.946
7.5	1.275	1.556	1.864

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APPENDIX C - CONTINUED

Material: Stainless Steel PH15-7MO (R.T.)

Limit load factor	Construction: 'Z' stiff. Rib spacing, inches			Construction: Hat stiff. Rib spacing, inches		
	12	16	20	12	16	20
2.5	1.716	1.738	1.757			
3.0	1.614	1.635	1.656			
4.0	1.460	1.475	1.501			
5.0	1.348	1.358	1.382			
6.5	1.252	1.258	1.270			
7.0	1.233	1.238	1.248			
7.5	1.209	1.213	1.230			

Limit load factor	Construction: 'Y' stiff. Rib spacing, inches			Construction: Integ. stiff. Rib spacing, inches		
	12	16	20	12	16	20
2.5				1.613	1.717	1.824
3.0				1.501	1.576	1.695
4.0				1.368	1.401	1.494
5.0				1.307	1.331	1.382
6.5				1.259	1.272	1.303
7.0				1.246	1.258	1.293
7.5				1.229	1.260	1.283

Limit load factor	Construction: Flat sheet Spar spacing, inches		
	6	9	12
2.5	3.197	3.737	4.393
3.0	2.984	3.503	4.130
4.0	2.644	3.128	3.699
5.0	2.374	2.828	3.347
6.5	2.048	2.451	2.915
7.0	1.959	2.358	2.797
7.5	1.870	2.256	2.679

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Material: Graphite/Epoxy with Holes (R.T.)

Limit load factor	Construction: 'Z' stiff.			Construction: Hat stiff.		
	Rib spacing, inches			Rib spacing, inches		
	12	16	20	12	16	20
2.5						
3.0						
4.0						
5.0						
6.5						
7.0						
7.5						

Limit load factor	Construction: 'Y' stiff.			Construction: Integ. stiff.		
	Rib spacing, inches			Rib spacing, inches		
	12	16	20	12	16	20
2.5						
3.0						
4.0						
5.0						
6.5						
7.0						
7.5						

Limit load factor	Construction: Flat sheet		
	Spar spacing, inches		
	6	9	12
2.5	0.944	1.116	1.316
3.0	0.906	1.068	1.252
4.0	0.877	0.975	1.136
5.0	0.875	0.928	1.059
6.5	0.854	0.875	0.962
7.0	0.853	0.875	0.943
7.5	0.853	0.862	0.913

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APPENDIX D

TEMPERATURE EFFECTS FACTORS

The factors presented in Appendix C were all based on room temperature. The factors ($K_{TEMPCVR}$) contained in this appendix modify the cover weight for temperature effects.

Materials available are:

- Aluminum - (200°F - 300°F)
- Titanium - (200°F - 500°F)
- Steel - (400°F - 1000°F)
- Advanced Composite - (180°F - 300°F)

APPENDIX D

TEMPERATURE EFFECTS FACTORS - UPPER COVER

Limit load factor	Aluminum			Titanium			
	Maximum structural temperature						
	200°F	300°F		200°F	300°F	400°F	500°F
2.5	1.037	1.093		1.022	1.052	1.084	1.119
3.0	1.039	1.097		1.026	1.063	1.104	1.144
4.0	1.039	1.104		1.047	1.095	1.142	1.188
5.0	1.039	1.121		1.060	1.119	1.167	1.214
6.5	1.048	1.152		1.068	1.127	1.188	1.239
7.0	1.053	1.163		1.070	1.131	1.193	1.248
7.5	1.054	1.169		1.070	1.132	1.199	1.257
Limit load factor	**Advanced composite**			**Steel**			
	Maximum structural temperature						
	180°F	**260°F**	**300°F**	**400°F**	**600°F**	**800°F**	**1000°F**
2.5	1.014	1.020	1.022	1.010	1.014	1.035	1.147
3.0	1.004	1.008	1.012	1.008	1.022	1.049	1.189
4.0	1.012	1.030	1.039	1.016	1.037	1.085	1.278
5.0	1.014	1.020	1.034	1.023	1.051	1.105	1.328
6.5	1.011	1.025	1.042	1.033	1.062	1.127	1.389
7.0	1.009	1.019	1.029	1.035	1.067	1.127	1.408
7.5	1.005	1.013	1.024	1.035	1.063	1.131	1.422

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APPENDIX D

TEMPERATURE EFFECTS FACTORS - LOWER COVER

Limit load factor	Aluminum			Titanium			
	Maximum structural temperature						
	200°F	300°F		200°F	300°F	400°F	500°F
2.5	1.086	1.259		1.013	1.033	1.067	1.107
3.0	1.102	1.285		1.021	1.063	1.112	1.159
4.0	1.109	1.300		1.057	1.124	1.185	1.239
5.0	1.111	1.307		1.083	1.152	1.221	1.280
6.5	1.110	1.317		1.092	1.168	1.259	1.322
7.0	1.113	1.324		1.099	1.179	1.268	1.334
7.5	1.113	1.331		1.106	1.197	1.275	1.343
Limit load factor	**Advanced composite**			**Steel**			
	Maximum structural temperature						
	180°F	**260°F**	**300°F**	**400°F**	**600°F**	**800°F**	**1000°F**
2.5	1.008	1.022	1.035	1.005	1.010	1.021	1.250
3.0	1.014	1.032	1.038	1.006	1.013	1.039	1.328
4.0	1.005	1.019	1.029	1.013	1.036	1.108	1.488
5.0	1.004	1.011	1.019	1.034	1.080	1.177	1.625
6.5	1.008	1.022	1.037	1.059	1.116	1.240	1.780
7.0	1.008	1.019	1.037	1.065	1.118	1.252	1.810
7.5	1.008	1.014	1.031	1.070	1.130	1.267	1.848

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