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STUDY OF AERODYNAMIC TECHNOLOGY FOR VSTOL FIGHTER/ATTACK AIRCRAFT

PHASE I FINAL REPORT

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16. Abstract

A conceptual design study was performed of a vertical attitude takeoff and landing (VATOL) fighter/attack aircraft. The configuration has a close-coupled canard-delta wing, side two-dimensional ramp inlets, and two augmented turbofan engines with thrust vectoring capability. Performance and sensitivities to objective requirements was calculated. Aerodynamic characteristics were estimated based on contractor and NASA wind tunnel data. Computer simulations of VATOL transitions were performed. Successful transitions can be made, even with serious post-stall instabilities, if reaction controls are properly phased. Principal aerodynamic uncertainties identified were post-stall aerodynamics, transonic aerodynamics with thrust vectoring and inlet performance in VATOL transition. A wind tunnel research program was recommended to resolve the aerodynamic uncertainties.

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FOREWORD

This study of VSTOL aerodynamic technology was the initial phase of a research program jointly sponsored by the National Aeronautics and Space Administration and the United States Navy, and administered by NASA. The Technical Monitor was Mr. W. P. Nelms of the Aircraft Aerodynamics Branch, Ames Research Center. Navy representatives were Mr. R. L. Schaeffer of the David W. Taylor Naval Ship Research and Development Center, and Mr. M. W. Brown of the Naval Air Systems Command.

Administrative manager for the Vought Corporation, Advanced Technology Center was Dr. C. H. Haight, Manager, Aerodynamics and Propulsion. The Principal Investigator was Mr. H. H. Driggers, Project Engineer, Vought Corporation, who directed the study.

The following Vought Employees made important contributions to the Phase I study effort:

D. D. Bender	Installed Propulsion Data
K. W. Higham	Mass Properties
R. L. Mask	Transition Program
R. G. Musgrove	Configuration Design
W. W. Rhoades	Inlet Analysis
S. Romero	Aerodynamic Estimates Thrust Vectoring Analysis
R. T. Stancil	Configuration Synthesis Short Takeoff Analysis
W. L. Straub, Jr.	Transition Analysis Thrust Vectoring Analysis Research Program

CONTENTS

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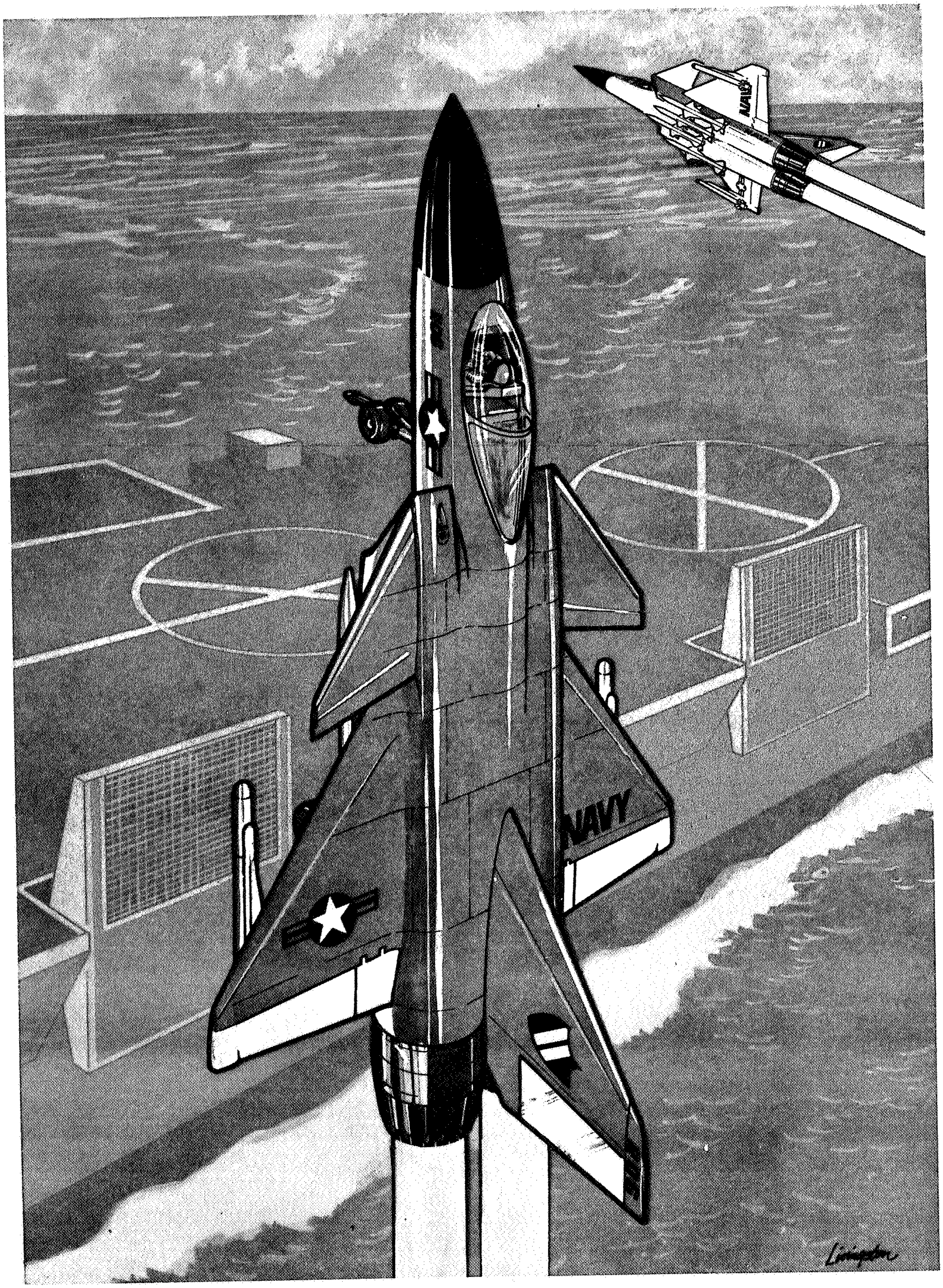


Superfly VATOL Concept

1.0 SUMMARY

Vought Corporation has conducted a conceptual design study and aerodynamic analysis of a Vertical Attitude Takeoff and Landing (VATOL) fighter/attack aircraft. The "Superfly" VATOL configuration is illustrated on the facing page. The salient features are the close coupled canard-delta wing planform and the two augmented turbofan engines fed by fixed ramp inlets. Axisymmetric gimballed nozzles and wingtip reaction jets provide attitude control in vertical attitude hover and transition. Conventional landing gear permit short takeoffs from ships or normal runway operation. Extensive use of composite materials make a single engine vertical landing capability a feasible design goal.

The SF-121 configuration was synthesized to objective performance guidelines. The principal sizing constraints were:

- o Supersonic Intercept mission radius = 150 NM (278 km) at Mach 1.6
- o Sustained load factor = 6.2 g at Mach 0.6 10,000 feet (3,048 m)
- o Single engine thrust/weight = 1.03 with afterburner.

The resulting point design has a VTO weight of 23,375 pounds (10,603 kg), a wing aspect ratio of 2.3 and a wing reference area of 354 ft^2 (32.89 m^2). The SF-121 is capable of short takeoffs with a 10,000 pound (4,536 kg) overload in 400 feet (122 m). The combat performance objectives were exceeded by a wide margin.

Detailed estimates of SF-121 aerodynamic characteristics were made based on Vought and NASA wind tunnel test data. This approach facilitated making predictions to 90 degrees angle of attack, but required linear superposition of essentially nonlinear flow phenomena to correct for geometry differences. Predicted longitudinal characteristics were well behaved except for a slight subsonic pitchup tendency. The 6.2 g design point can be met without buffet. The chief aerodynamic problem was directional instability at high angle of attack.

Three degree-of-freedom computer simulations were made of transition to and from vertical attitude hover. Outbound conversions presented no problems and could be completed in 17 seconds (to Mach 0.3). Decelerations were expedited by angles of attack beyond 90 degrees. Normal two engine

reconversions to hover were generally uncomplicated, with ample thrust and control power. Successful transitions with one engine (T/W = 1.03) were also feasible, but control margins were very small. The critical region is around 50 degrees angle of attack, where destabilizing moments are high and thrust settings are low.

The potential of high speed thrust vectoring was assessed. No improvement in specific excess power or sustained load factor was found. Combined canard flap deflection and thrust vectoring was useful for direct lift control and fuselage aiming. Canard lift was the limiting factor on TVC application.

Principal aerodynamic uncertainties are high angle of attack flying qualities, buffet characteristics and transonic aeropropulsion interactions. Other uncertainties peculiar to VATOL mode operations are ship wake turbulence effects, propulsion induced spray and pilot visibility requirements. A research program was proposed around a wind tunnel model concept compatible with the XM2R compact propulsion simulator.

2.0 INTRODUCTION

The SF-121 VATOL fighter concept which is the subject of this report is a product of a continuing Vought VSTOL configuration research program. VATOL emerged as a highly promising approach to VSTOL fighter propulsion, offering exceptional performance and a simple propulsion system.

Reference 1 provides a summary of the Vought propulsion concepts screening studies, including a description of the configurations, comparative performance and sensitivities to design constraints. A Remote Augmentor Lift System (RALS) configuration was also investigated. Figure 2-1 compares relative VTO weights of five VSTOL fighter concepts evaluated to common groundrules. The superiority of the VATOL candidate results from the absence of dedicated lift machinery or major aerodynamic compromises. The deflected thrust candidate in Figure 2-1 has essentially the same propulsion weight as the VATOL; the VTO weight difference is due to aerodynamic configuration compromises required to achieve balance.

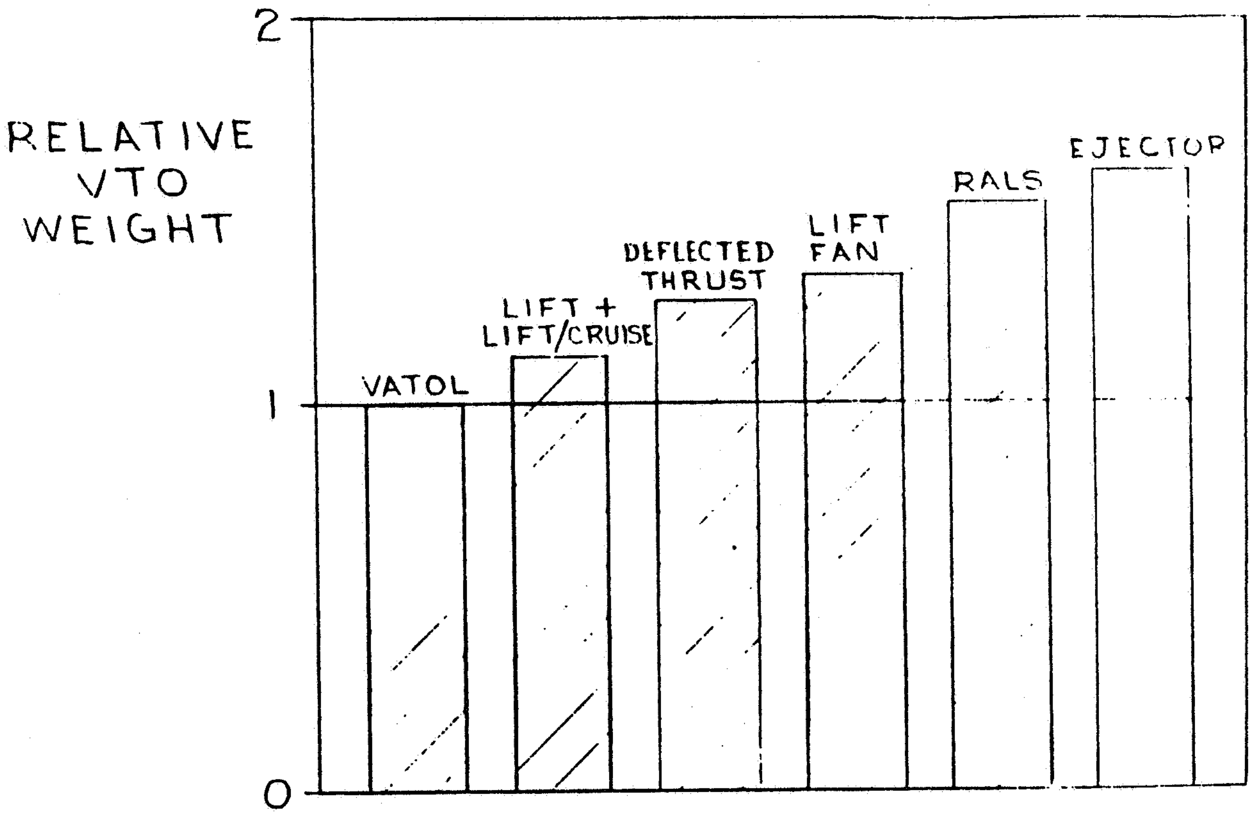


Figure 2-1 - Weight Comparison of VSTOL B Concepts

Figure 2-2 illustrates the original Vought approach to a Navy VATOL fighter, the SF-106 "Superfly". The configuration is aggressively simple to minimize empty weight, cost and maintenance. The propulsion system is sized to permit a minimum weight vertical landing with either engine disabled. The landing gear is compatible with vertical or horizontal attitude operations. The low aspect ratio delta wing was selected for low weight and supersonic drag and for its gradual stalling characteristics. This configuration was tested in the Vought 4 x 4 foot Supersonic Wind Tunnel to Mach 2.4 and to 35 degrees angle of attack. The SF-121 configuration incorporates lessons learned from these tests.

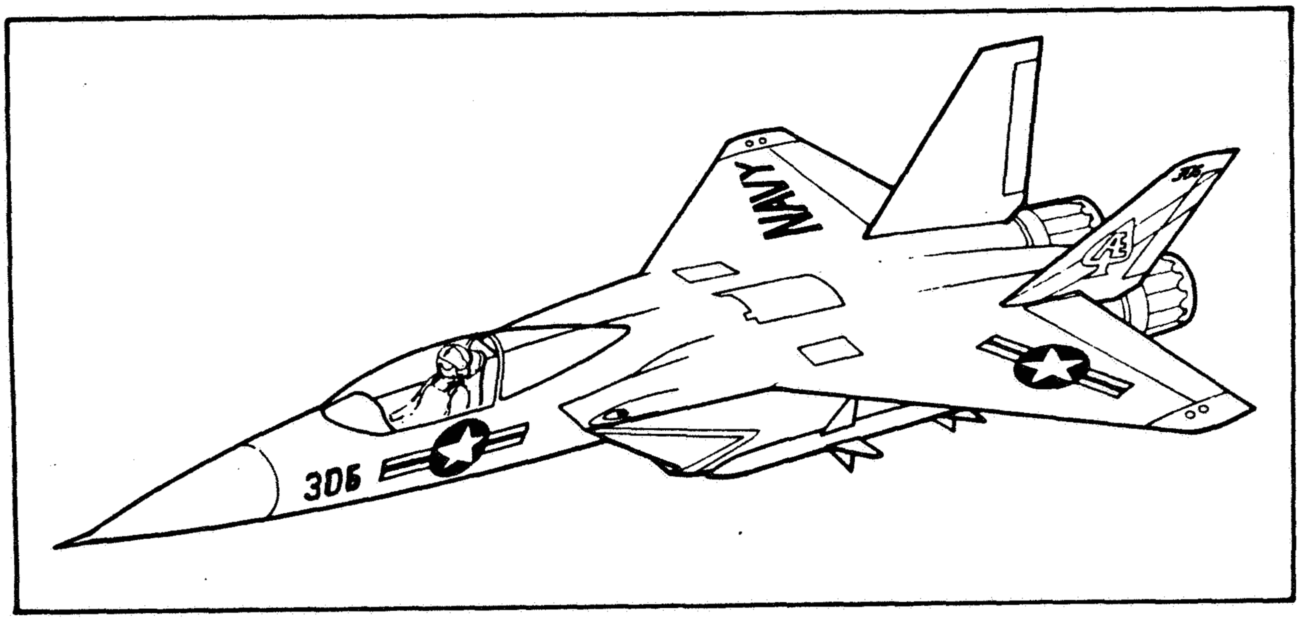


Figure 2.2 - SF-106 Superfly VATOL Fighter

From these studies it was apparent that the VATOL principle was ideally suited to a high performance fighter/attack requirement. Combat agility was impressive and the only mandatory propulsion development was the thrust vectoring system. The fundamental question of operational suitability must ultimately require a flight demonstration program to resolve.

Vought proposed the VATOL concept for detailed analysis to the objective performance guidelines in the Request for Proposal (Reference 2). Mission roles, weapons definition and technology projections were made by Vought.

The Phase I study objectives were:

- o Evaluate performance potential of a vertical attitude takeoff and landing (VATOL) fighter/attack aircraft concept.
- o Estimate aerodynamic characteristics of the design
- o Determine ability to transition to and from vertical attitude hover by computer simulation
- o Assess aerodynamic uncertainties associated with the study configuration and the VATOL operating mode
- o Develop a wind tunnel research program and model concept to explore aerodynamic uncertainties and acquire a data base.

3.0 SYMBOLS

A	Aspect Ratio
b	Wing Span
$\bar{c}$	Mean Aerodynamic Chord, in. (m)
C_D	Drag Coefficient
C_L	Lift Coefficient
C_M	Pitching Moment Coefficient
C_{n_β}	Directional Stability Derivative
C_{l_β}	Lateral Stability Derivative
e	Span Efficiency Factor
I_x	Moment of Inertia about Roll Axis, Slug-ft^2
I_y	Moment of Inertia about Pitch Axis, Slug-ft^2
I_z	Moment of Inertia about Yaw Axis, Slug-ft^2
M	Mach Number
n_x	Longitudinal Load Factor, g
n_z	Normal Load Factor, g
P_s	Specific Excess Power, ft/sec
S	Wing Reference Area, ft^2 (m^2)
T	Total Net Thrust, lb (N)
T_j	Reaction Jet Thrust, lb (N)
T_g	Gross Thrust, lb (N)
α	Angle of Attack, Degrees
α_{BO}	Buffet Onset Angle of Attack, Degrees
β	Sideslip Angle, Degrees
β_T	Thrust Deflection Angle in Yaw, Degrees
γ	Flight Path Angle
δ_C	Canard Incidence, Degrees
$\delta_{C_{TEF}}$	Canard Flap Deflection Angle, Degrees

δ_T Thrust Vector Angle in Pitch, Degrees

$\delta_{W_{LEF}}$ Wing Leading Edge Flap Deflection Angle, Degrees

$\delta_{W_{TEF}}$ Wing Trailing Edge Flap (Elevon) Deflection Angle, Degrees

Λ Leading Edge Sweep, Degrees

Θ Pitch Attitude Angle

4.0 SF-121 DESCRIPTION

4.1 DESIGN PHILOSOPHY

The Superfly SF-121 is the latest of a series of Vought high performance VSTOL fighter concepts. The SF-121 design philosophy centers around:

- o The Vertical Attitude Takeoff and Landing (VATOL) principle
- o Normal landing gear for conventional takeoff and landing capability
- o The ability to make a vertical landing on either of its two lift/cruise engines.

The VATOL approach offers the highest performance at the lowest weight of all candidates evaluated by Vought (Reference 1). It is also a very simple solution to achieving VSTOL capability; both airframe and propulsion can be relatively conventional, yet benefit fully from advanced technology.

Twin engines are a hallmark of the Superfly concept. Some alternative concepts require more than one engine for VTOL operation and are doubly vulnerable to an engine failure. VATOL is as feasible with one or two engines as conventional fighters are; the choice is not dictated by necessity. Considerations favoring twin engines for the SF-121 were:

- o Lower peacetime attrition rate
- o Fewer opportunities to rescue downed pilot with dispersed forces
- o Higher survivability probable
- o Ease of engine handling on shipboard
- o Practical Limitations on engine size

The philosophy of designing for a single engine vertical landing was continued on the SF-121; otherwise any survivability arguments for twin engines were invalidated. The engine-out consideration is an important one for VSTOL. The effect is to place a premium on empty weight.

The SF-121 design philosophy was influenced by the Phase I study philosophy. This was to define a basic configuration for in-depth analysis and as a point of departure for a comprehensive wind tunnel test program. To this end the configuration was kept "aggressively simple". The aerodynamic fixes evaluated in the Vought high speed wind tunnel tests, reported in Reference 3, were held in reserve for future use. (This decision was reinforced by the observation that many devices which suppress stall departure effects cause a more severe departure at a higher angle of attack.) Similarly, active lift enhancement, such as spanwise blowing or Vought's ATC wing were not considered appropriate for a reference test configuration, but could be factored into future test programs.

4.2 DESIGN GUIDELINES

4.2.1 Performance Guidelines

The Request for Proposal, Article II, lists certain objective performance guidelines. These are:

- o High performance VSTOL fighter/attack aircraft
- o Supersonic dash capability with sustained Mach number capability of at least 1.6
- o Operational from land and from ships smaller than CVs without catapults and arresting gear (good STO capability)
- o Sustained load factor of 6.2 at Mach 0.6, 10,000 foot altitude at 88 percent VTOL gross weight.
- o Specific excess power at 1G (Ps1G) of 900 fps at Mach 0.9, 10,000 foot altitude at 88 percent VTOL gross weight.
- o VTOL gross weight = 20,000 to 35,000 pounds.
- o STO sea-based gross weight - VTOL gross weight plus 10,000 pounds.

Previous VATOL studies indicated the only constraining parameter would be the sustained load factor, which would size the wing. The ability to meet the STO requirement had to be confirmed.

Several other guidelines must be stated to uniquely define a point design; chief among them are the design mission profile and radius. The mission most compatible with RFP guidelines and the intrinsic merits of VSTOL is the Supersonic Intercept (Deck Launched Intercept) mission, diagrammed in Figure 4-1. This typical radius of 150 NM was selected for the SF-121 study. The design mission establishes internal full load. Previous Vought studies indicate that good attack mission performance can be obtained with the DLI mission internal fuel plus external tasks. No minimum alternate mission radius or time on station requirements were imposed for the subject study.

One important guideline to be resolved by the study was the VTOL engine sizing constraint. For a single engine configuration this would be VTO thrust/weight, typically 1.10. The twin engine Superfly concept is better characterized by a one engine vertical landing requirement, as applied to the Navy Type A VSTOL (T/W = 1.03).

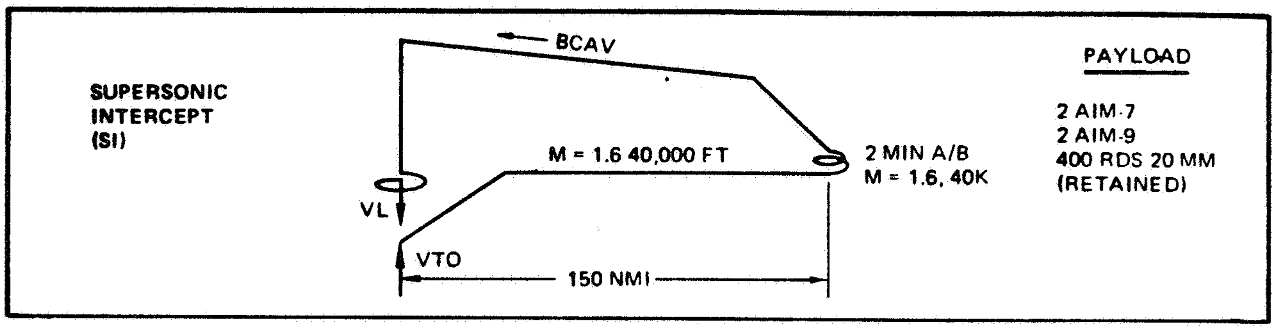


Figure 4-1 - SF-121 Design Mission Profile

4.2.2 VATOL Design Considerations

Vought recognized that stated requirements were nominal and that system optimization was not the purpose of the subject program. There was, however, one aspect of VATOL which deserved close scrutiny: wing planform. In general, horizontal attitude configurations can employ whatever wing geometry is desirable for high speed flight. Even wing area and high lift systems are likely to be defined by maneuver constraints on a high performance fighter. Wing planform (e.g., aspect ratio) may have some effect on "HATOL" propulsion induced effects, but no fundamental limitations are likely. The situation is different for a VATOL fighter. It is highly desirable for VATOL that the aerodynamics be "well behaved" throughout transition. Selection of the low aspect ratio delta wing, characteristic of the Superfly VATOL, was influenced by this consideration. Vortex lift counteracts an abrupt stall causing a smooth, gentle peak in the C_L vs α curve which peaks near 35 degrees. Higher aspect ratios and/or reduced sweep may be acceptable, with other propulsion concepts, particularly if integrated with strakes and body contouring, but the suitability for VATOL is uncertain.

A wing aspect ratio study was conducted at the beginning of the Phase I effort, as summarized in Section 8.2.1. The SF-121 wing results from that study.

4.3 SF-121 CONFIGURATION

The SF-121 Superfly is a close coupled canard-delta wing configuration. The canard is mounted high on two-dimensional side inlets above a moderately blended mid wing of low aspect ratio. The fixed geometry ramp inlets feed two augmented turbofan engines equipped with axisymmetric gimballed nozzles.

Figure 4-2 illustrates the parent configuration, emphasizing the compact proportions, the closely spaced vectoring nozzles and the conformal stores installation. The SF-121 is designed to achieve high combat agility and mission versability, yet be compatible with dispersed basing on sea and land.

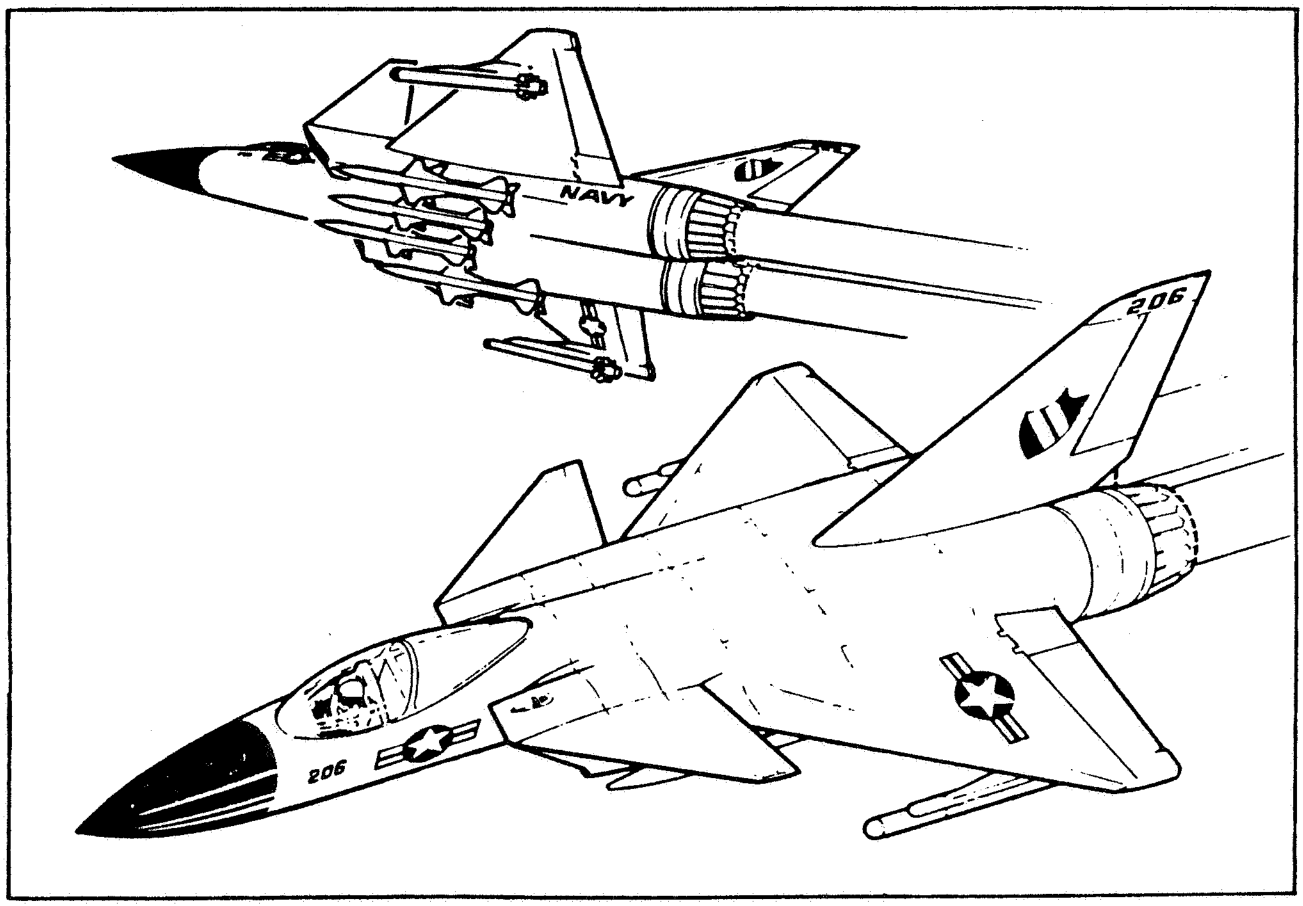


Figure 4-2 - SF-120 Series Superfly VATOL Fighter

The General Arrangement drawing, Figure 4-3, and the Armament Installation drawing, Figure 4-4, reveal additional design details. These will be amplified in Section 6.0.

Overall span and fuselage length are 28.53 and 45.25 (8.70 and 13.79 m) feet, respectively. Spotting factor relative to the A-7E is only 0.83, so a singfold is not required. Static ground height is 14.17 feet (4.32 m), which is compatible with hangar height of any contemplated basing ship. Figure 4-5

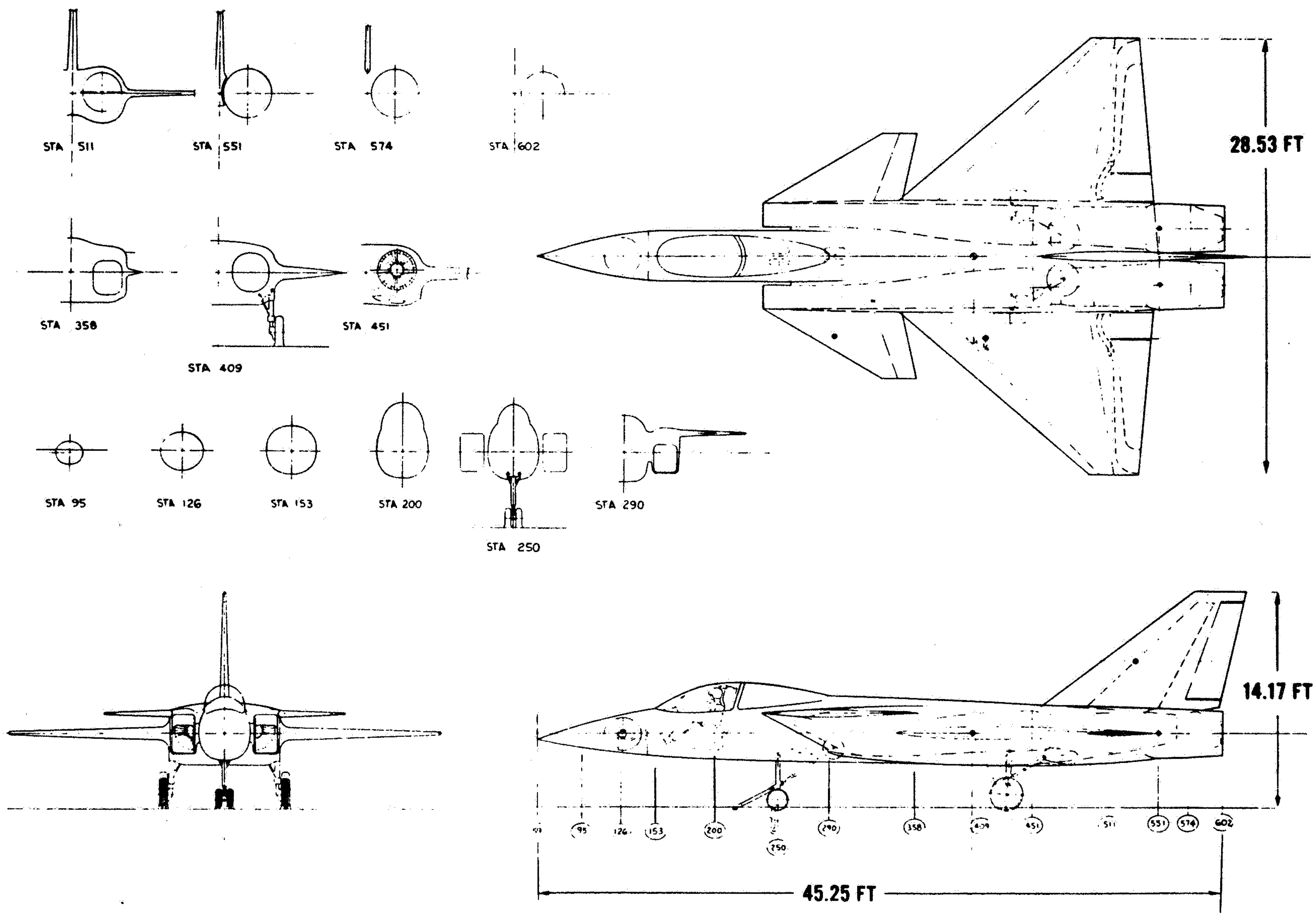


Figure 4-3 - SF-121 General Arrangement

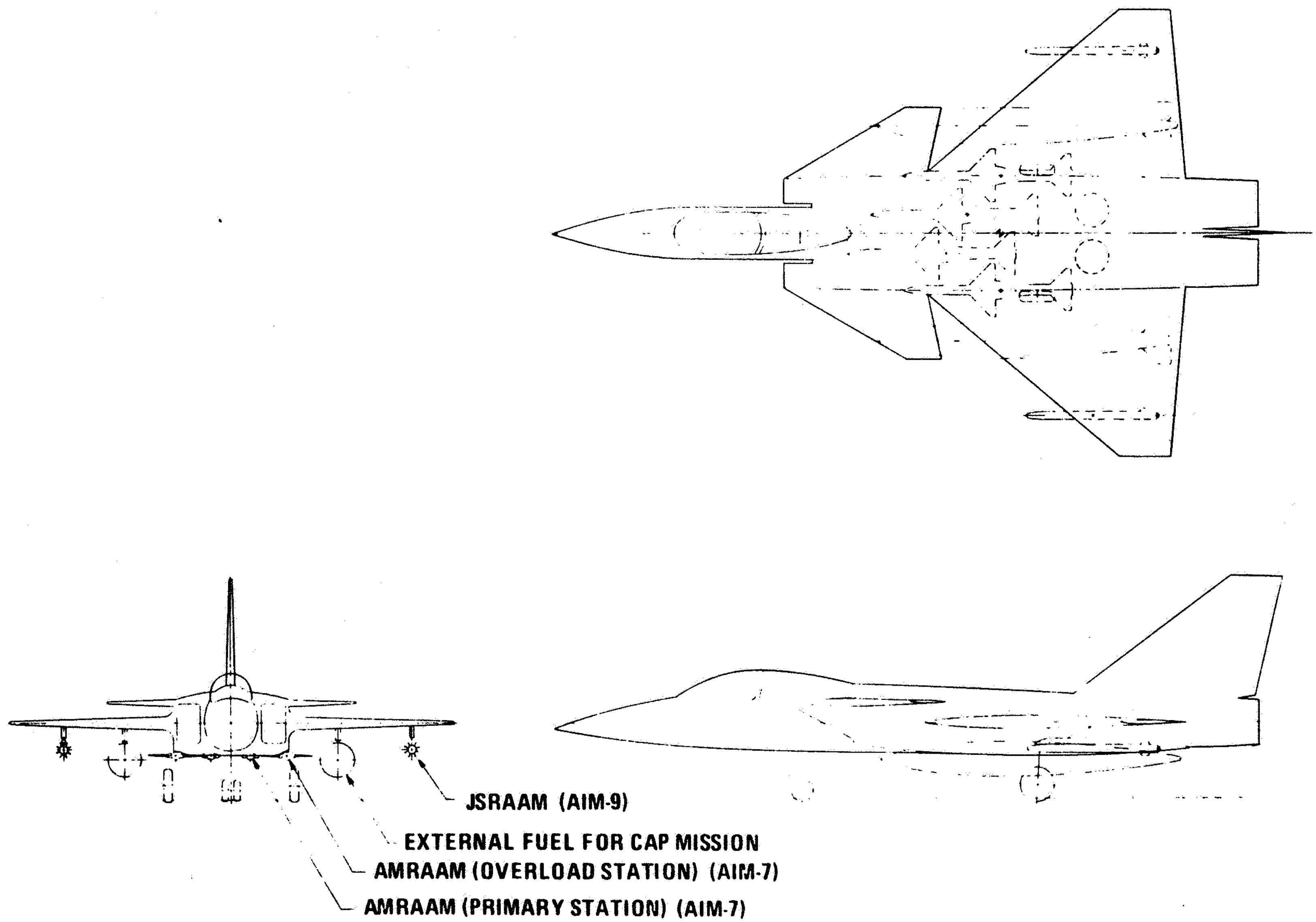


Figure 4-4 - SF-121 Armament Installation

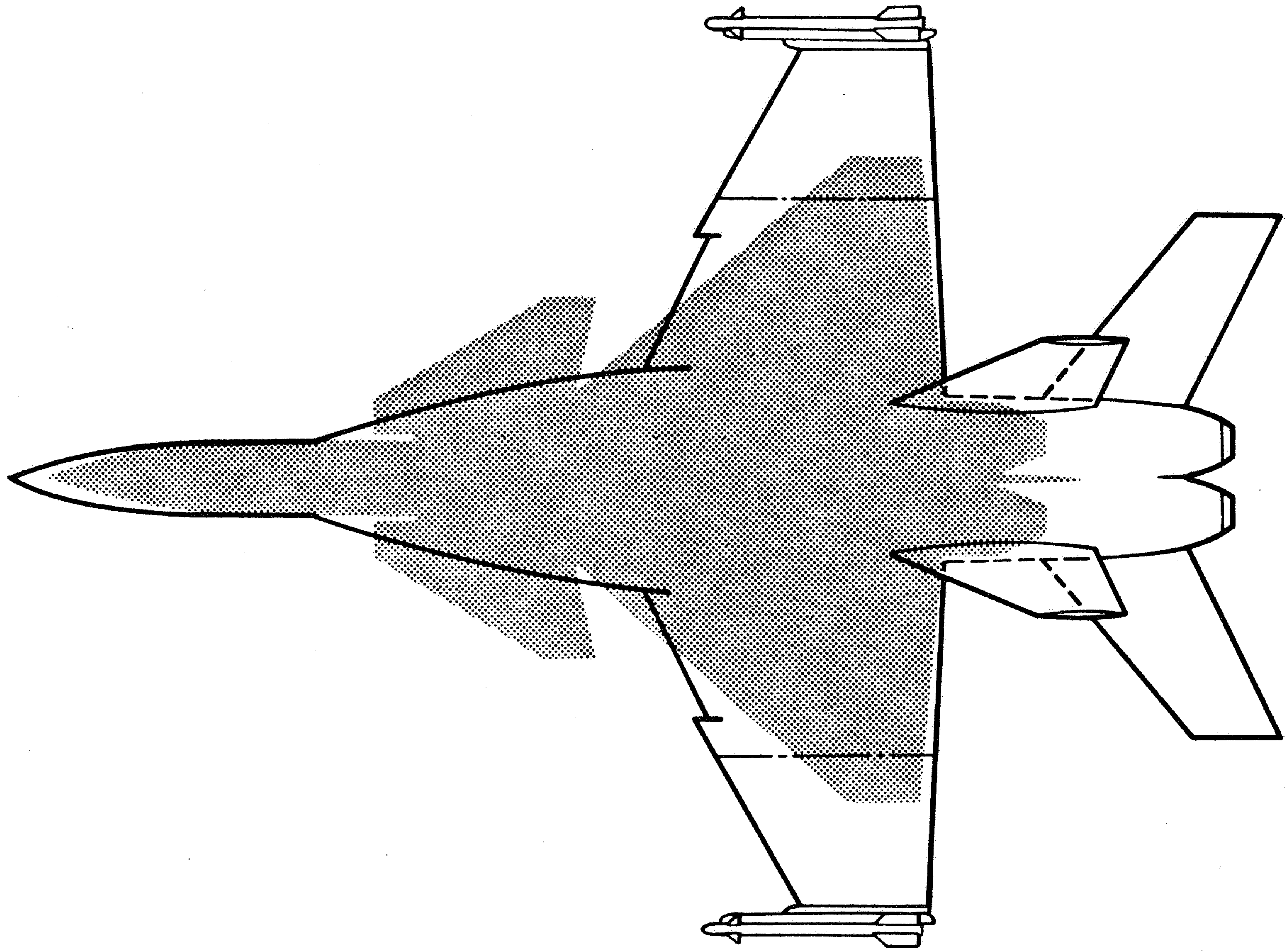


Figure 4-5 - SF-121 Planform Contrasted with F-18 emphasizes the small size of the SF-121 in relation to the F-18. Table 4-1 summarizes the geometric characteristics of the aerodynamic surfaces. Movable surfaces are defined in Table 4-2.

Table 4-1 - SF-121 Aerodynamic Surfaces Geometry

AERODYNAMIC		WING	CANARD (PER SIDE)	FIN
REFERENCE AREA	$FT^2(M^2)$	354(32.89)	-	-
EXPOSED AREA	$FT^2(M^2)$	-	26.3(2.44)	60.0(5.57)
OVERALL SPAN	IN(M)	342.4(8.70)	-	-
EXPOSED SPAN	IN(M)	-	54.9(1.39)	93.0(2.36)
ASPECT RATIO		2.30	0.80	1.00
TAPER RATIO		0.150	0.250	0.300
LEADING EDGE SWEEP -	DEG	50.0	60.0	53.0
THICKNESS RATIO (ROOT/TIP)		0.05	0.05/0.04	0.05/0.04
MEAN GEOMETRIC CHORD	IN(M)	177.0(4.50)	77.0(1.96)	101.9(2.59)
ROOT CHORD	IN(M)	260.4(6.61)	110.0(2.79)	143.0(3.63)
TIP CHORD	IN(M)	39.0(0.99)	27.5(0.70)	42.9(1.09)

Table 4-2 - SF-121 Control Surfaces Geometry

CONTROL SURFACES		WING L.E. FLAP	ELEVON	SPEED BRAKE	CANARD FLAP	RUDDER
AREA, PER SIDE	FT^2 (M^2)	17.1 (1.59)	19.3 (1.79)	9.5 (0.88)	6.9 (0.64)	11.6 (1.08)
SPAN, PER SIDE	IN (M)	129.5 (3.29)	107.3 (2.73)	20.6 (0.52)	53.6 (1.36)	74.4 (1.89)
ROOT CHORD	IN (M)	30.2 (0.77)	32.4 (0.82)	34.4 (0.87)	22.0 (0.56)	27.0 (0.69)
TIP CHORD	IN (M)	7.8 (0.20)	19.5 (0.50)	32.4 (0.82)	15.0 (0.38)	18.0 (0.46)
MAXIMUM DEFLECTION	DEG	- 30	± 60	± 60	± 40	± 40

Figure 4-6 is a cross-sectional area buildup for the SF-121, as defined by the Vought 3-D Area Rule computer program. The area distribution includes two inlet streamtubes of 653 in^2 (0.421 m^2) each, which includes the boundary layer diverter.

Wetted areas for the SF-121 broken down in Table 4-3.

TABLE 4-3 - SF-121 Wetted Area by Component

COMPONENT	WETTED AREA -FT^2 (M^2)	CHARACTERISTICS LENGTH-FT (M)
WING	436.6 (40.56)	11.66 (3.55)
CANARD	103.6 (9.62)	6.29 (1.92)
FIN	122.0 (11.33)	8.49 (2.59)
FUSELAGE	678.0 (62.99)	45.25 (13.79)
TOTAL	1,340.2 (124.51)	-

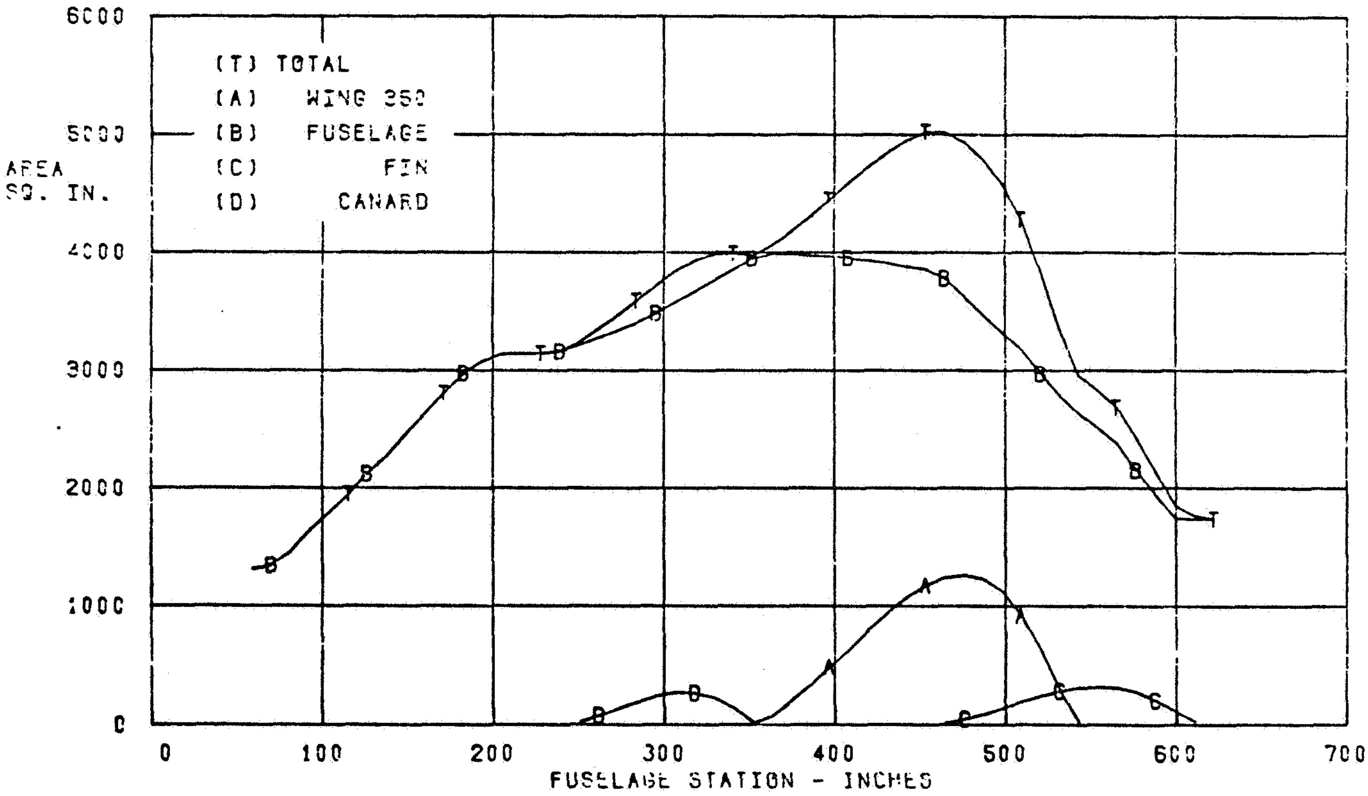


Figure 4-6 - SF-121 Normal Cross-Sectional Area Distribution

5.0 AERODYNAMIC CHARACTERISTICS

The SF-121 was the subject of detailed aerodynamic estimates. Minimum drag and trimmed drag due to lift were prerequisites to sizing a point design, and were determined first. The flying qualities parameters were used in the transition analysis. Except for minimum drag, wind tunnel data was relied upon heavily. Vought conducted high speed wind tunnel tests (Reference 3) on a parametric flow-through model similar to the SF-121. The model differed in several respects which makes it an imperfect data base, particularly for lateral-directional characteristics. Since the Vought tests extended only to α = 35 degrees, a less representative configuration (Reference 4) had to be used for the basis of high angle of attack characteristics. Figure 5-1 diagrams the procedure used (except for minimum drag). The configuration differences are indicated by Figure 5-2.

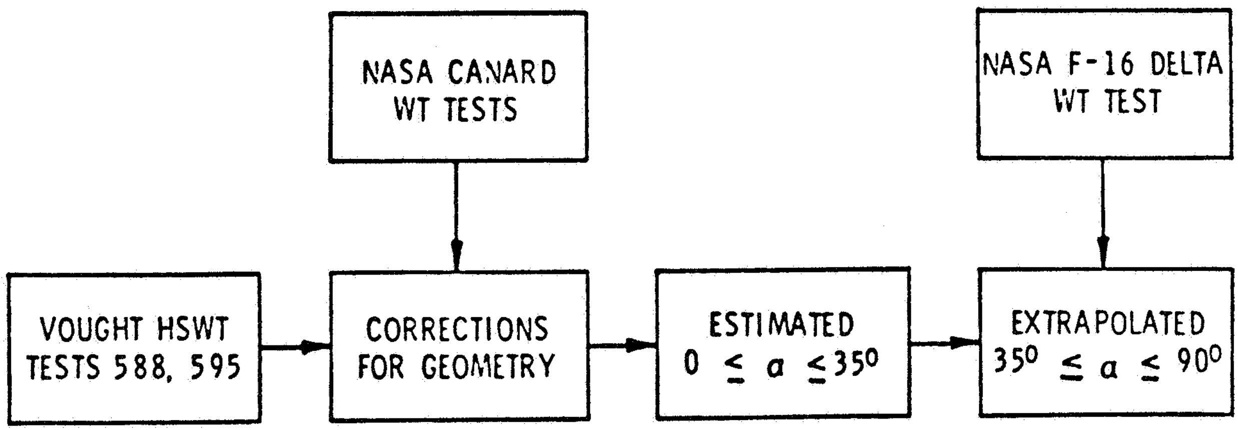


Figure 5-1 - Aerodynamic Estimation Procedure

The resulting analysis presented in the following subsections and detailed in Appendix A is as accurate as was possible with available data and proved valuable in the performance analyses discussed in Section 8. However, the nonlinearities in the coefficients may suggest more precision than is really present; use them, but with caution.

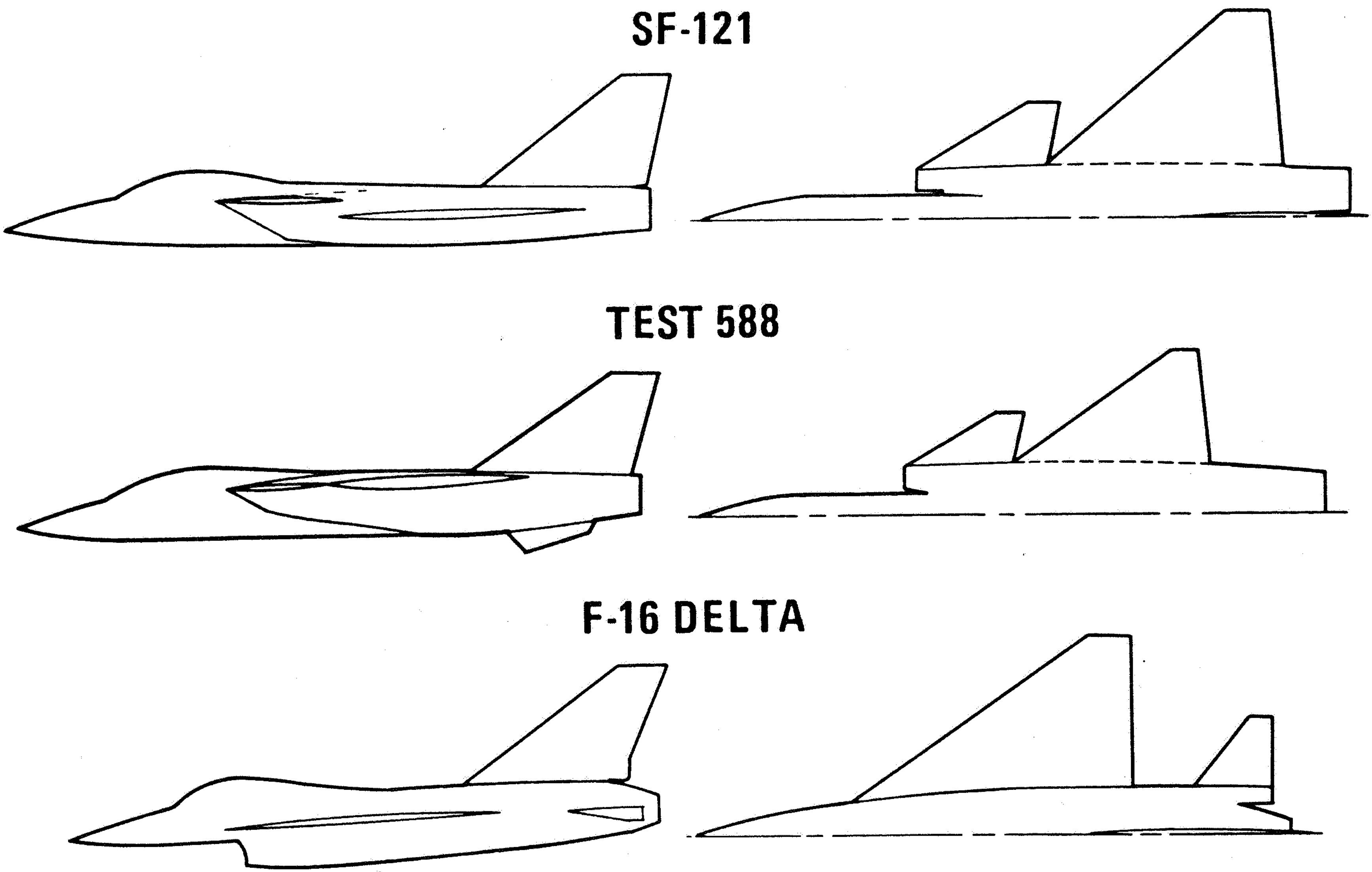


Figure 5-2 - Study and Test Configurations

5.1 LONGITUDINAL CHARACTERISTICS

5.1.1 Minimum Drag

No suitable experimental data base was available for SF-121 minimum drag, so the established buildup procedure diagramed in Figure 5-3 was used to estimate minimum drag as a function of Mach number. The method involves summing the following contributions:

- o Friction and subsonic form drag - Linden-O'Brimski/VAC/DATCOM
- o Transonic drag rise - VAC/Voohrees
- o Wave drag - Vought 3-D Area Rule, plus modified linear theory (Reference 5)
- o Base drag - NASA experiment
- o Miscellaneous - adjusted from Model 1600 Proposal, Performance Data Report

Nozzle/afterbody drag is bookkept with installed thrust. Table 5-1 is a complete minimum drag buildup for the SF-121. The miscellaneous drag is further detailed in Table 5-2. The final clean configuration minimum drag

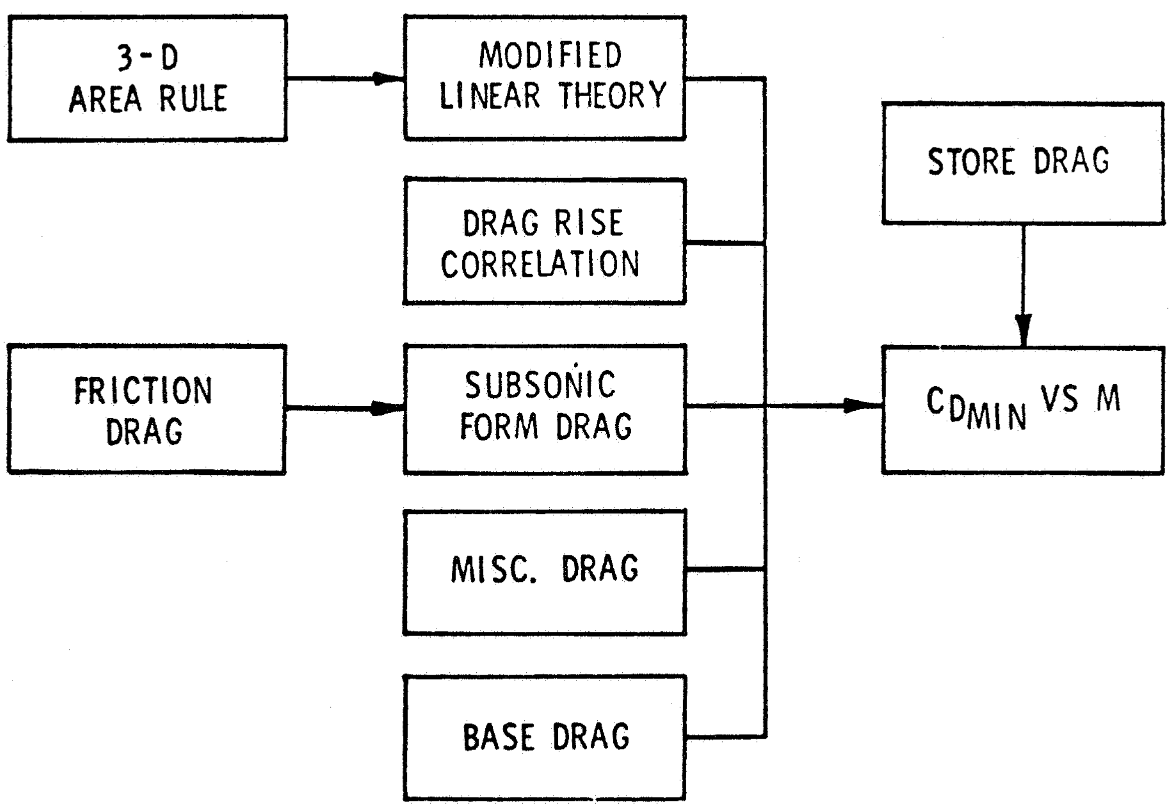


Figure 5-3 - Minimum Drag Buildup Procedure

Table 5-1 - SF-121 Minimum Drag Buildup

S_{WING} = 354 ft^2

h = 36,089 ft

S_{WET} = 1,347 ft^2

MACH	FRICTION	FORM AND INTERFERENCE	WAVE DRAG	BASE DRAG	MISC.	TOTAL (FAIRED)
0.2	.01122	.00165	-	.00042	.00184	.0151
0.4	.00993	.00139	-	.00042	.00191	.0136
0.6	.00917	.00150	-	.00042	.00197	.0130
0.8,0.86	.00856	.00189	-	.00042	.00218	.0130
0.9					.00238	.0135
0.95						.0168
1.0						.0259
1.05						.0306
1.1						.0313
1.2	.00760	-	.0201	.00046	.00380	.0320
1.4	.00730	-	.0208	.00041	.00407	.0325
1.6	.00701	-	.0209	.00035	.00396	.0322
1.8	.00674	-	.0197	.00031	.00377	.0305
2.0	.00648	-	.0183	.00028	.00358	.0286
2.2	.00619	-	.0172	.00025	.00351	.0272
2.4	.00594	-	.0166	.00022	.00343	.0262
2.6	.00571					.0257

NOTES:

(1) Wave drag based on 3-D area rule plus modified linear theory.

(2) Base drag for 1.0 ft^2 base between nozzles with air dumped into base; use 1/2 of X-15 base Cp, NACA TR-100, Figure 3.

(3) Drag rise per VAC/Voorhees method.

(4) Altitude correction is .000017/1,000 ft. below 36,089; .000076/1,000 ft. above 36,089.

(5) Changes in afterbody drag relative to maximum afterburning at 36,089 feet are bookkept in thrust.

Table 5-2 - SF-121 Miscellaneous Drag

MACH	PROTUBERANCE, COOLING, VENT. D/q	ROUGHNESS, WAVINESS, LEAKAGE D/q	B.L. DIVERTER D/q	TOTAL D/q	ΔC_D
0.2	.3821	.2564	.0111	.6496	.00184
0.4	.380	.2574	.0371	.6745	.00191
0.6	.377	.2584	.0632	.6986	.00197
0.8	.401	.2718	.1000	.7728	.00218
0.9	.4347	.2867	.1224	.8438	.00238
1.2	.6591	.4512	.2338	1.3441	.00380
1.4	.603	.3887	.4506	1.4423	.00407
1.6	.5568	.3420	.5031	1.4019	.00396
1.8	.5273	.3191	.4881	1.3345	.00377
2.0	.4978	.2961	.4731	1.2670	.00358
2.2	.4854	.2895	.4664	1.2413	.00351
2.4	.473	.2829	.4598	1.2157	.00343

NOTES:

(1) Based on Model 1600 Proposal, Vol. II, Book 5A, Performance Data Report

(2) Assume boundary layer diverter drag is proportional to capture area (SF-121 A_{CAP} = 1092 in^2)

(3) Assume roughness, waviness, leakage drag is reduced 10 percent due to composites, then scaled proportional to wetted area.

(4) Assume greater use of flush antennas will reduce protuberance drag by 10 percent, also cooling and ventilation drag reduced by 10 percent.

(5) Delete horizontal tail actuator fairing and arresting hook and fairing.

coefficient as a function of Mach number is presented in Figure 5-4. Current external stores were assumed for the SF-121 study, since they can be defined with certainty and wind tunnel drag is available. Future weapons will differ from the present generation, but will have generally similar drag and weights.

AIM-7 (Sparrow) drag for the semi-submerged mounting was the most difficult to estimate. No test configuration was an exact match to the SF-121 installation, and scatter was quite high. A fairing was made of the most relevant configurations.

The AIM-9 Sidewinders are carried on dedicated pylons and launchers at the 82 percent semispan. This location was chosen to keep the missiles clear of the reaction jets and reduce roll axis inertia. Vought wind tunnel data for a similar installation was used without adjustment.

Similar test data for tangential carriage of MK83 LD bombs were applied to the SF-121. Previous estimates for the Harpoon missile were used directly.

Figure 5-5 summarizes the store drag increments used to evaluate SF-121 performance on the five mission profiles (Figures 8-4 and 8-5).

For missions with external fuel, increments for tank and pylon drag were added. The wing pylon and 300 gallon fuel tank drag curves in Figure 5-6 were obtained from Vought wind tunnel tests.

5.1.2 Trimmed Lift and Drag

The trimmed information is for the static margin giving minimum drag at a Mach 0.8 nominal cruise lift coefficient of 0.3. The minimum static margin, at Mach 0.6, is -10.5 percent up to $C_L = 0.5$ and changing to -14.1 percent at $C_L > 0.8$. The trimmed data reflects a scheduled (with angle of attack and Mach No.) wing leading edge flap and canard trailing edge flap deflection as shown in Figure 5-7. Canard incidence is fixed at -5 degrees. The flap schedules and canard incidence were selected to give minimum trimmed drag. The primary longitudinal trim control is the wing trailing edge flap. The estimation of the untrimmed data, which were based primarily on the data of Reference 3 for a configuration similar to the SF-121 configuration, is discussed in Section 5.1.3.

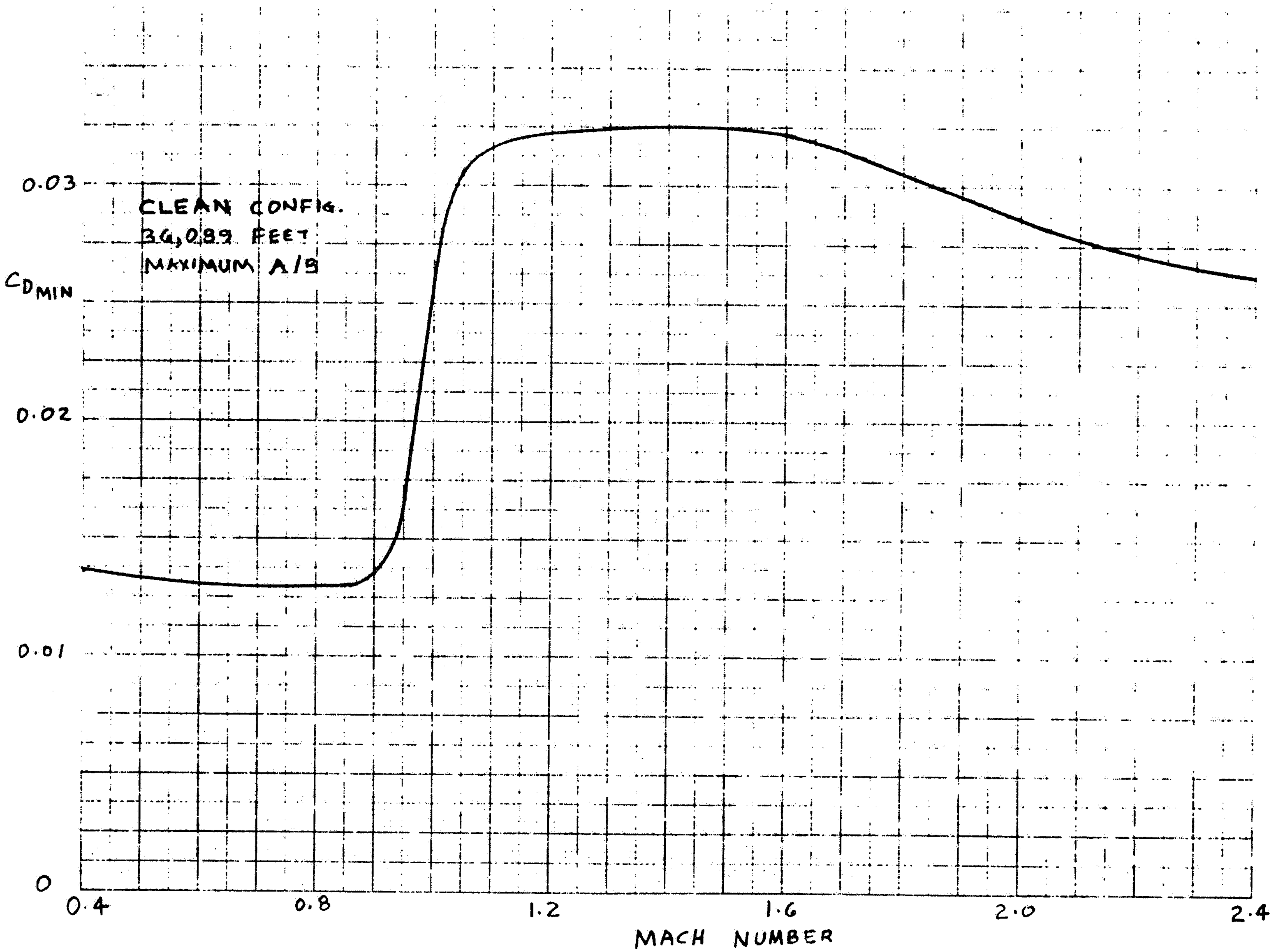


Figure 5-4 - SF-121 Minimum Drag

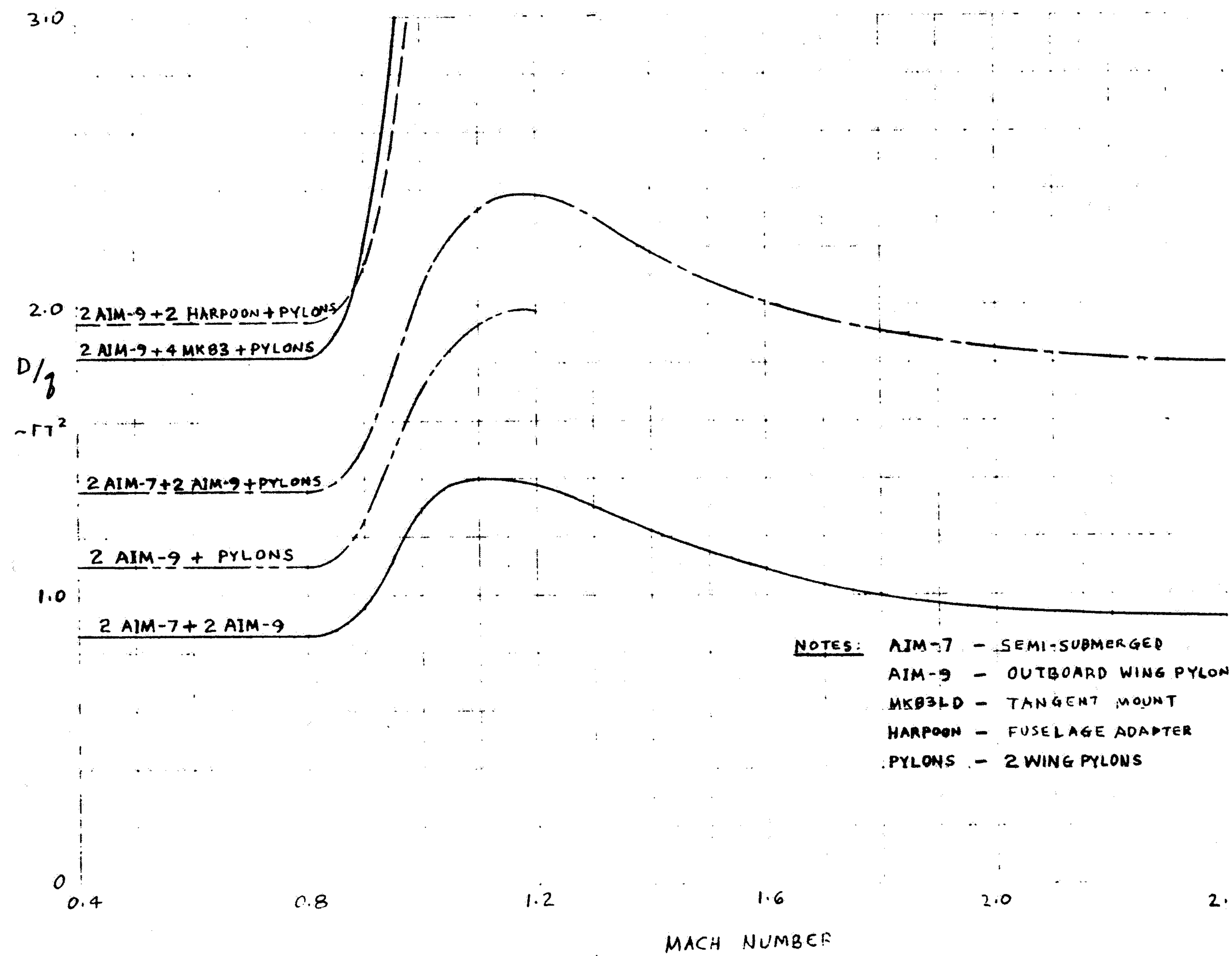


Figure 5-5 - SF-121 Store Drag Increments

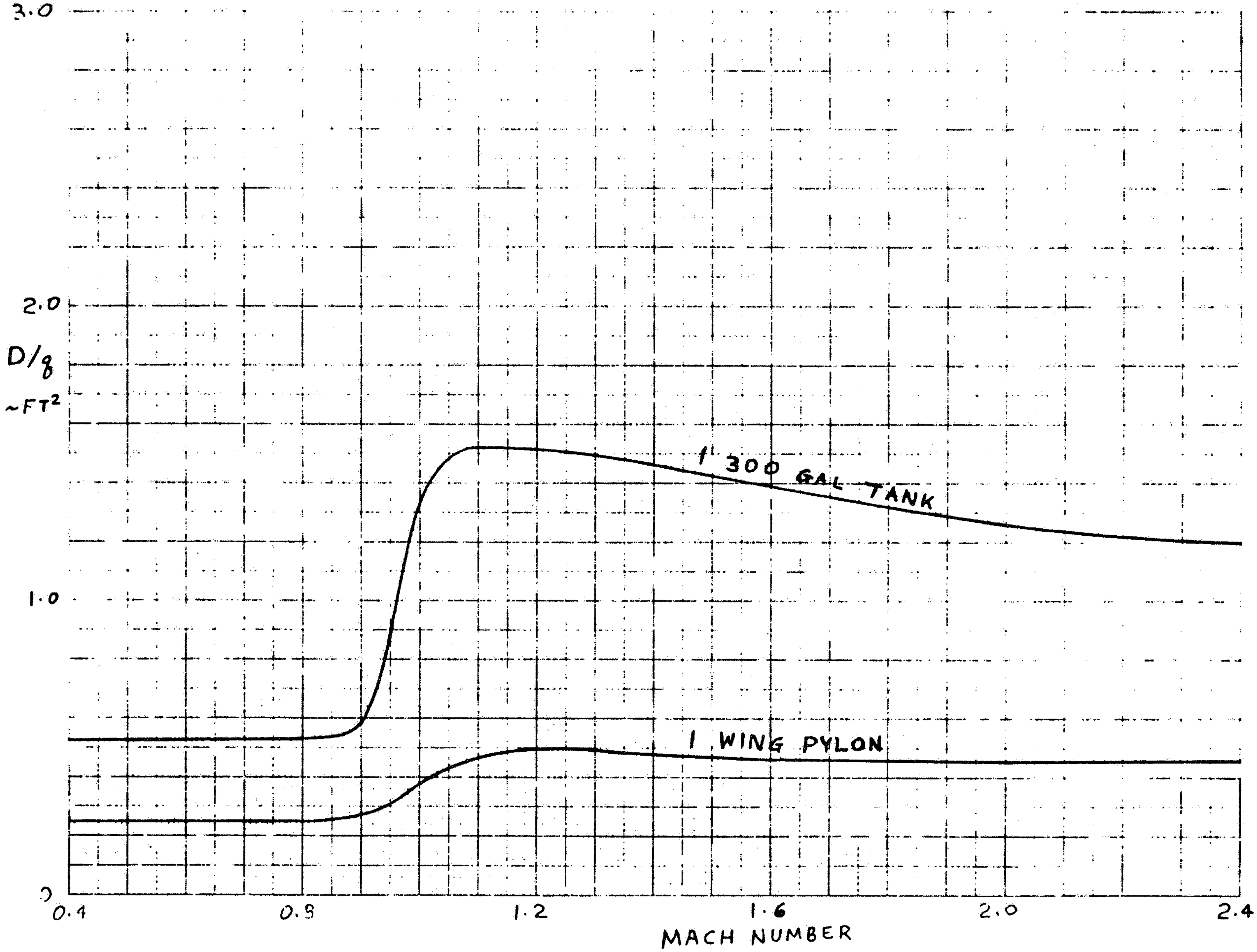


Figure 5-6 - External Tank and Pylon Drag Increments

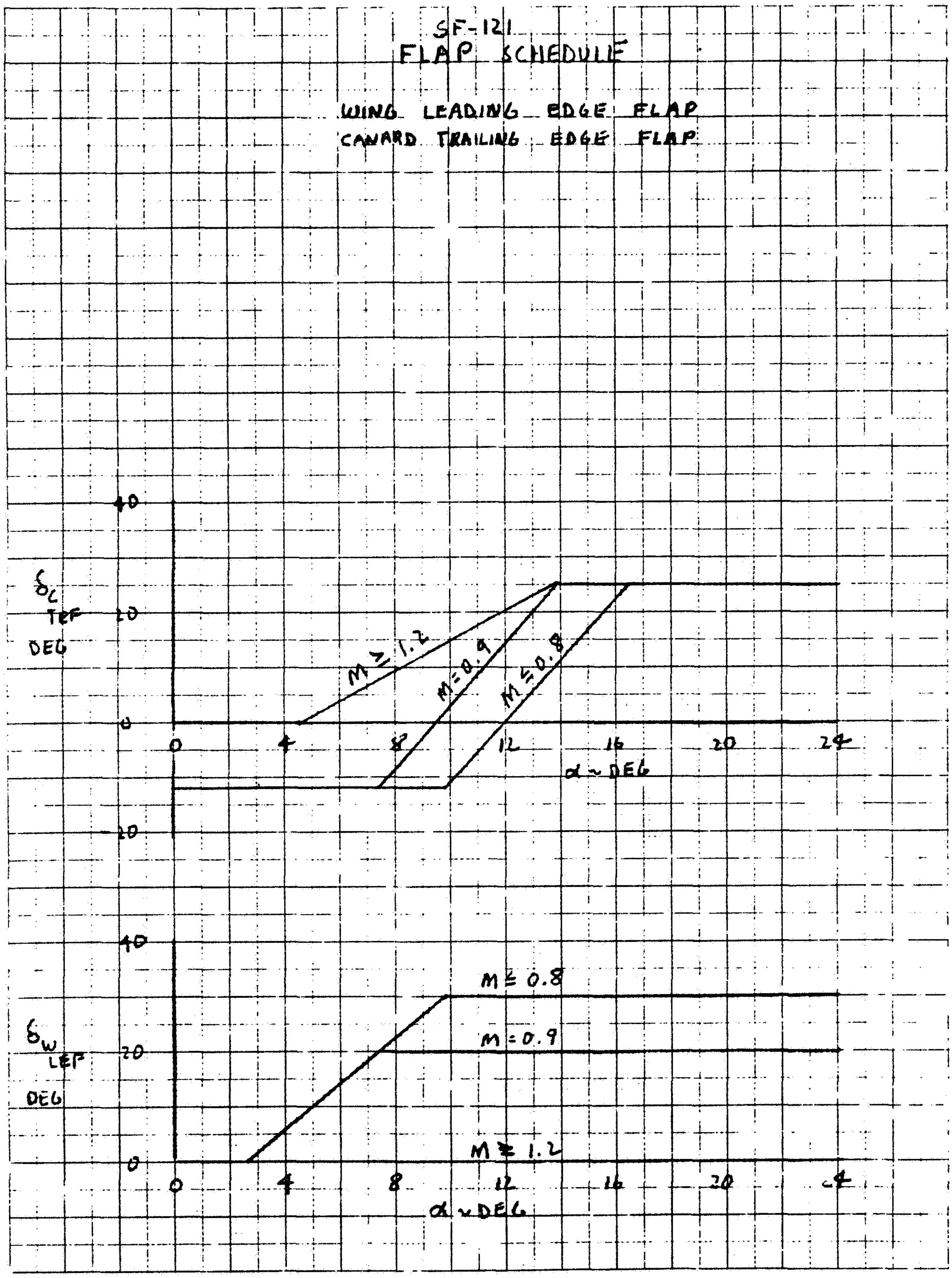


Figure 5-7 - Control Surface Schedule

Figure 5-8 shows the subsonic lift curves up to the first severe lift drop off which defines maximum lift. These maximum values along with the highest C_L shown at Mach 1.6 require sizable wing trailing edge flap deflections for the trim; +30 degrees at subsonic Mach numbers and -30 degrees at Mach 1.6. These high deflections may leave inadequate longitudinal control power remaining, and the angles of attack are such that C_{n_β} is negative. An analysis of the longitudinal control power required at maximum lift and the lateral/directional controllability is necessary to define maximum usable lift coefficient. In the absence of such an analysis, a limit wing trailing edge flap (elevon) deflection of $\pm$ 25 degrees was used to define the maximum usable lift coefficient.

Figure 5-9 shows maximum usable and buffet onset lift coefficient versus Mach number. Buffet onset angles of attack were based on the data in Reference 3 with the canard at zero incidence, the wing trailing edge flap at 10 degrees, and no wing leading edge flap deflection. The average of the angles of attack for the lift curve break and axial force curve break gave:

M	0.6	0.8	0.9
α_{BO}	13.4	9.1	9.6

Data in Reference 4 indicated that the presence of the canard improves the wing lift and that the break in the total (canard on) lift curve indicated canard buffet occurring prior to wing buffet. The data indicated that wing buffet occurs at an angle of attack about 3 degrees higher than that for canard buffet. It is thus assumed that the angles of attack from Reference 3 are for canard buffet onset with the canard at zero incidence. These angles were increased by 5 degrees since the SF-121 canard incidence is -5 degrees. Data in Reference 6 indicated that wing leading edge flap deflection will put the wing buffet onset angles of attack higher than those for canard buffet. There, for SF-121, the buffet onset angles of attack are:

M	0.6	0.8	0.9
α_{BO}	18.0	14.0	15.0

Figure 5-10 shows SF-121 trimmed drag polars. They are derived from test data in Appendix A, with an additional adjustment to the subsonic low lift coefficient drag due to lift. The adjustment consisted of establishing the drag due to lift at $M = 0.9$, $C_L = 0.3$ by the methods of Reference 7

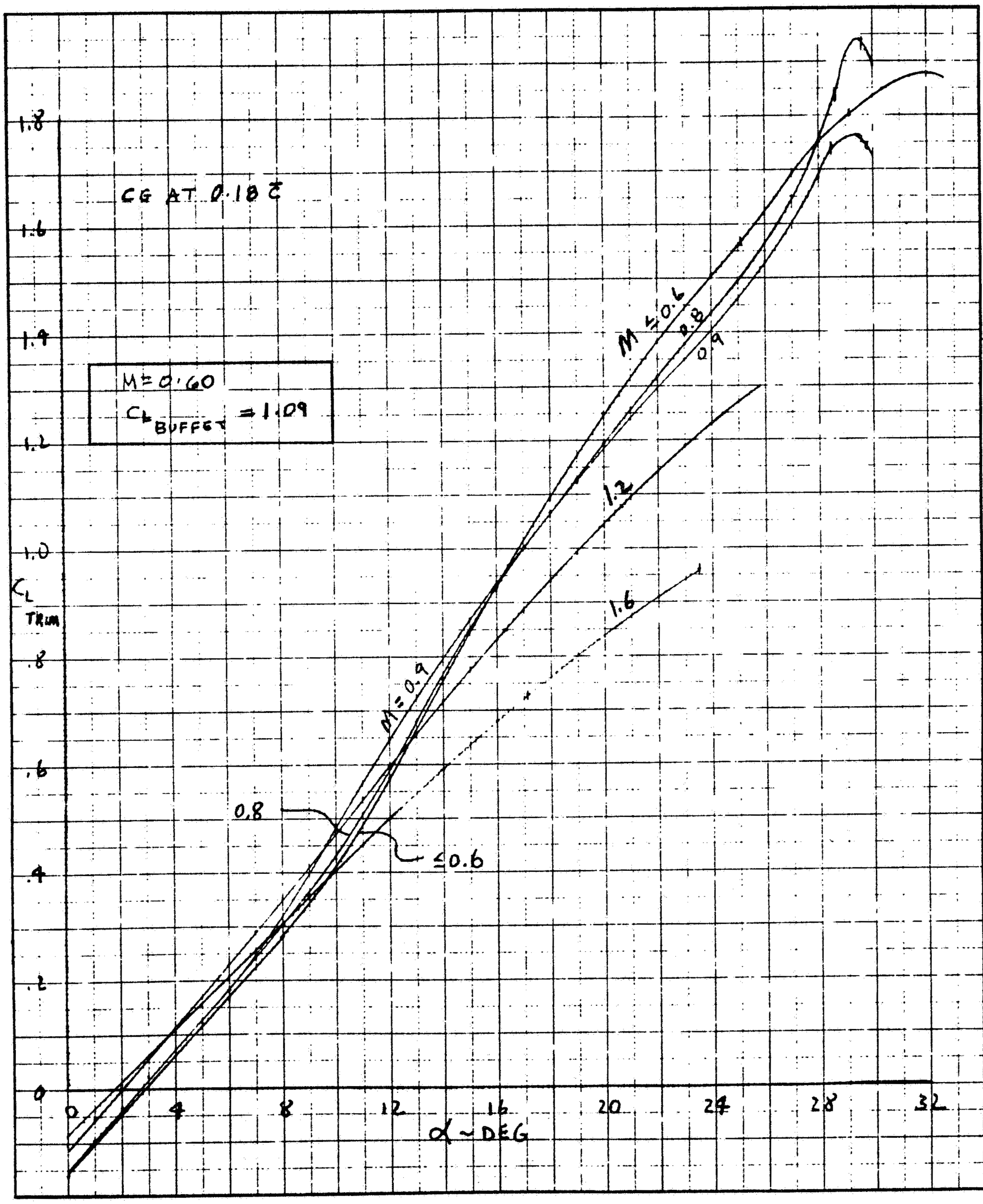


Figure 5-8 - SF-121 Trimmed Lift Coefficient

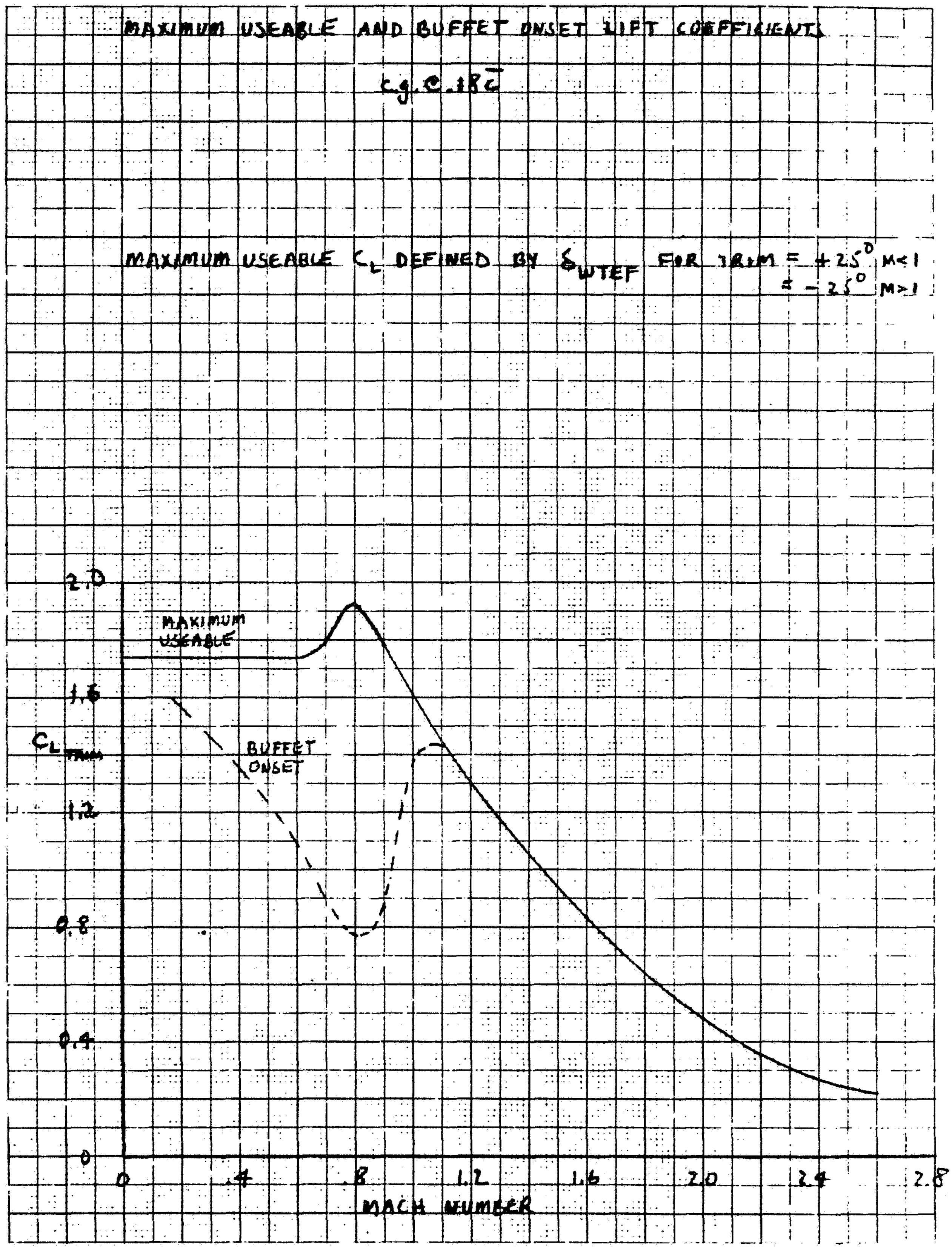


Figure 5-9 - Maximum Usable and Buffet Onset Lift Coefficients

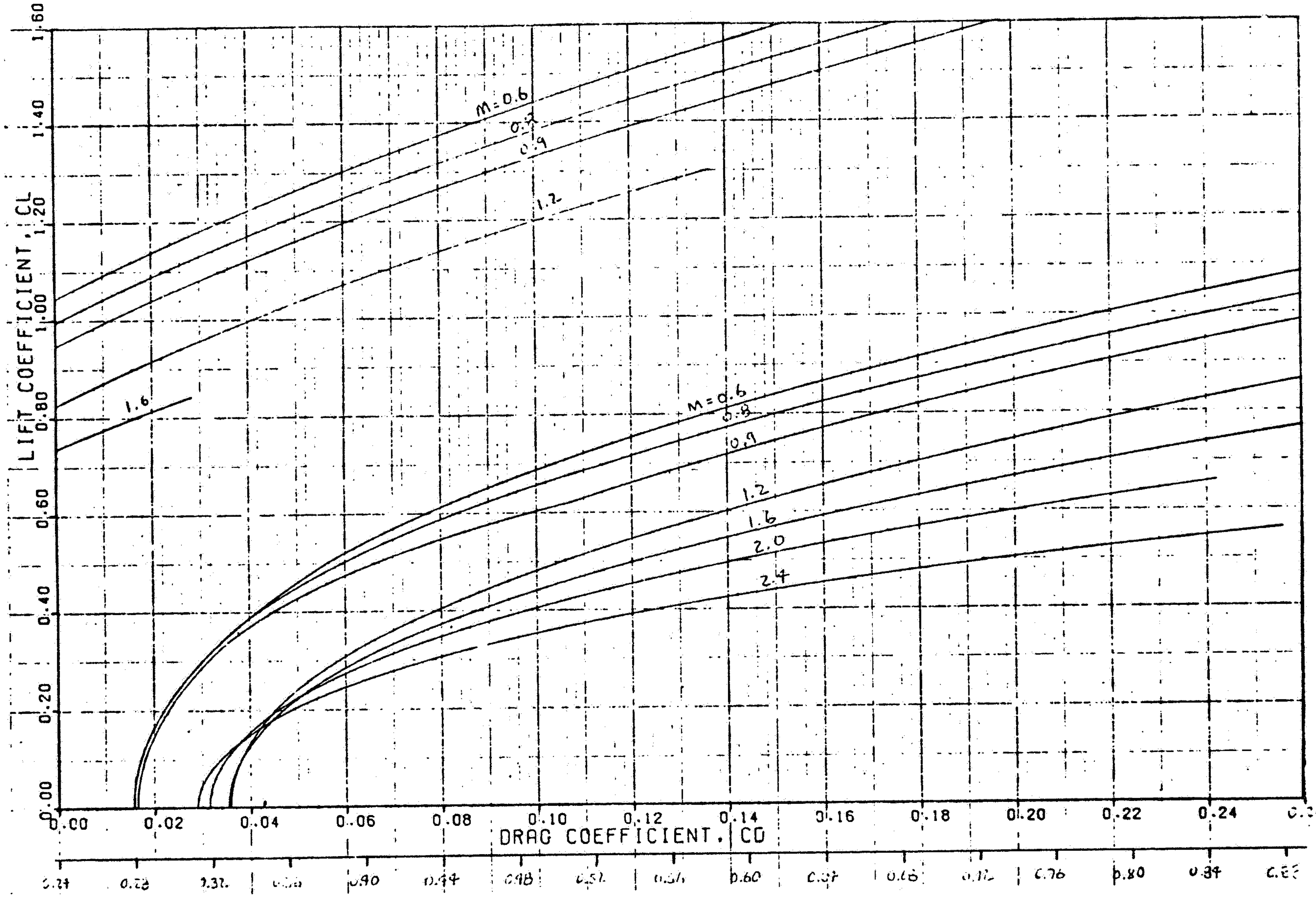


Figure 5-10 - SF-121 Trimmed Drag Polars and fairing from there to the trimmed levels of the data in Appendix A at lift coefficients of 0.5 at M = 0.6, 0.8 at M = 0.8, and 0.65 at M = 0.9. The data of Reference 7 are based on flight tests, and thus reflect levels that are achievable. Figure 5-11 shows the trimmed span efficiency factors used to calculate SF-121 performance. Figure 5-12 shows L/D versus lift coefficient.

5.1.3 Untrimmed Longitudinal Characteristics

The trajectory programs used in the transition analysis required untrimmed lift, drag and pitching moment coefficients to 90 degrees angle of attack. Figures 5-13, 5-14, and 5-15 provide this information for the SF-121 configuration.

The buildup of the untrimmed characteristics from Vought and NASA wind tunnel test data is detailed in Appendix A.

5.2 LATERAL/DIRECTIONAL AERODYNAMICS

5.2.1 Controls Neutral Characteristics

Figures 5-16, 5-17, and 5-18 show C_{n_β}, and C_{Y_β} respectively for the SF-121. The basic data base is results from Reference 3 for a configuration with a canard. Adjustments for configuration are detailed in Appendix A.

Briefly, the lateral/directional characteristics were obtained by first adding a vertical tail contribution, appropriately adjusted to the SF-121 tail size, to the BWC_o^o configuration data at M = 0.6. At M = 1.2, the first steps were to remove the effects of nose strakes and ventral fins, S_{N_1} and V_F, from the $BWC_o^o\ S_{N_1}\ V_C\ V_F$ configuration data and to apply a vertical tail size correction. Final characteristics were obtained by; (1) adding the effects of moving the wing from a high vertical position on the fuselage to a mid vertical position, (2) interpolating for M = 0.9, (3) adding effects due to the deflection of the wing leading edge flaps and the canard trailing edge flaps to the subsonic data, (4) correcting the data to a c.g. position of 0.18 MGC, and (5) extending the M = 0.6 data to α = 90 degrees. Figure 5-19 shows the resulting extrapolated stability derivatives.

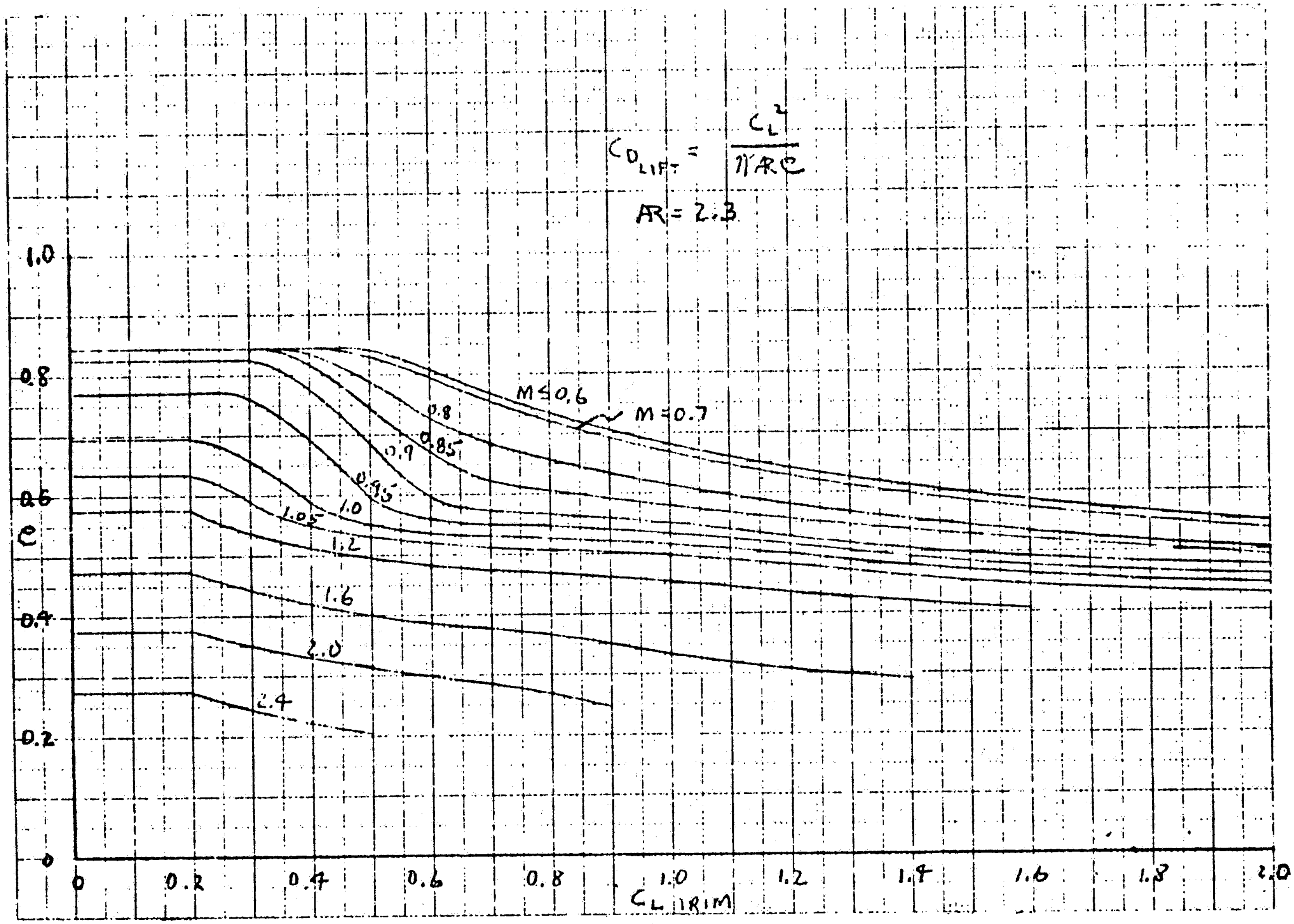


Figure 5-11 - SF-121 Trimmed Span Efficiencies

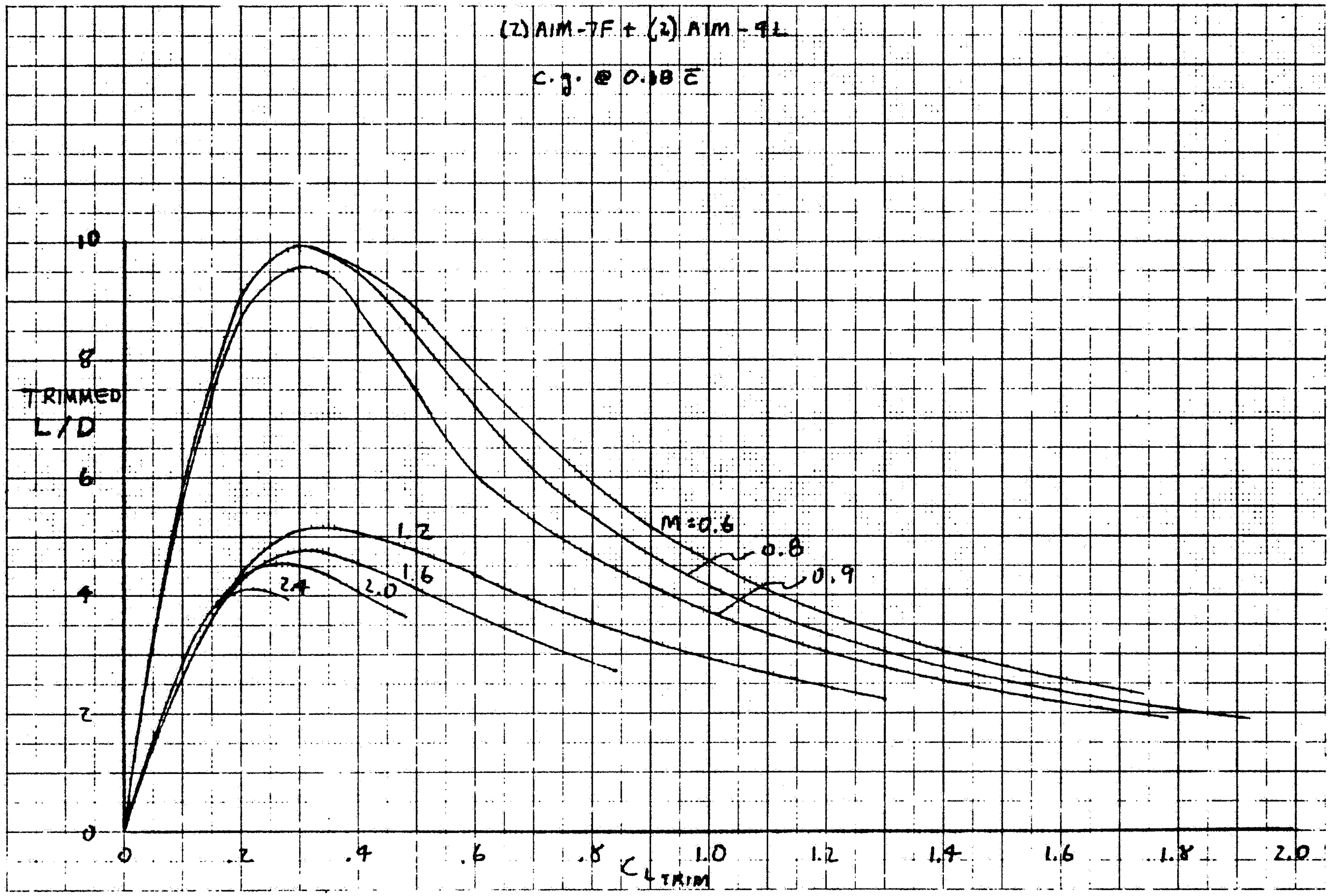


Figure 5-12 - SF-121 Lift/Drag Ratios

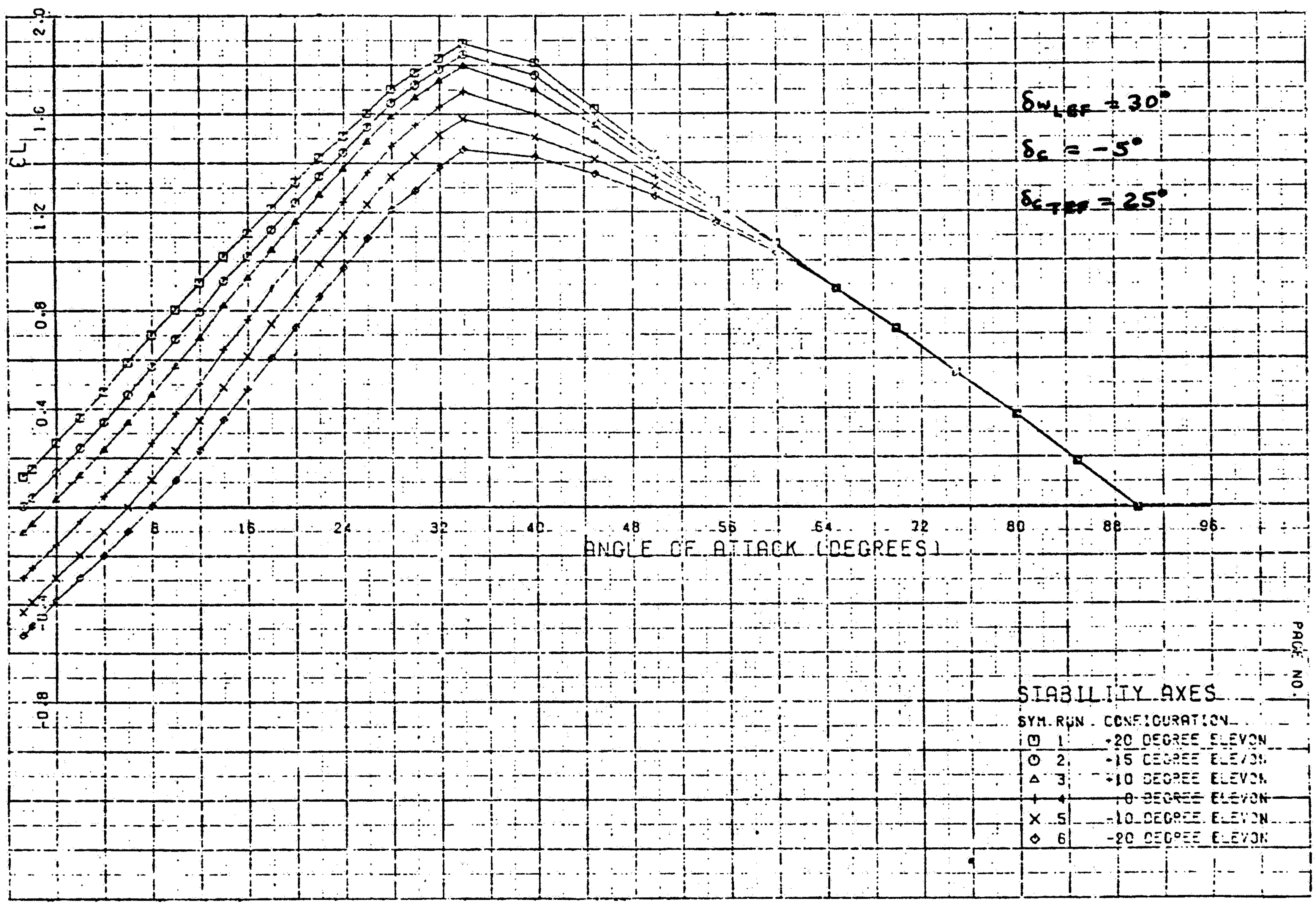


Figure 5-13 - SF-121 Untrimmed Lift Coefficients

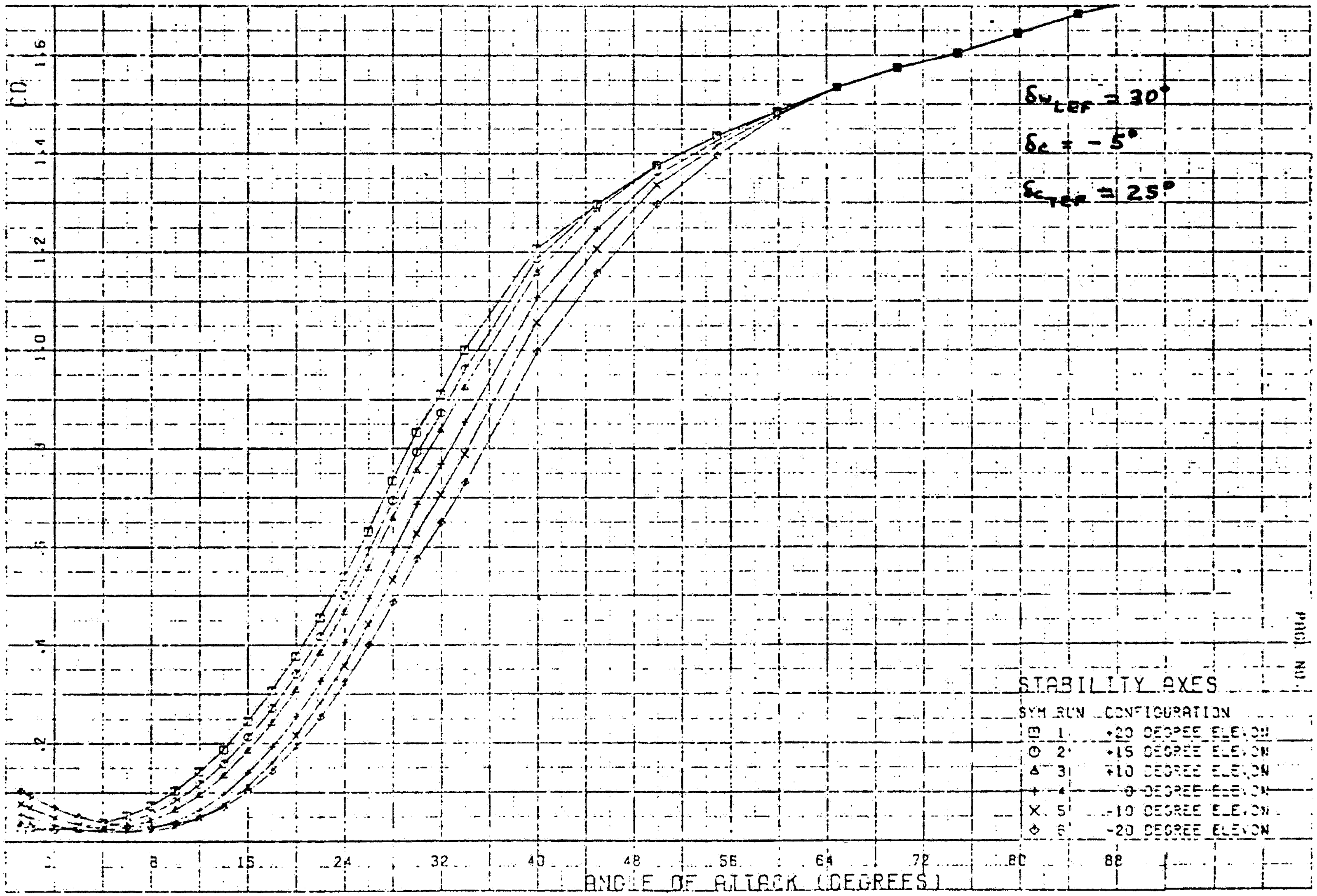


Figure 5-14 - SF-121 Untrimmed Drag Coefficients

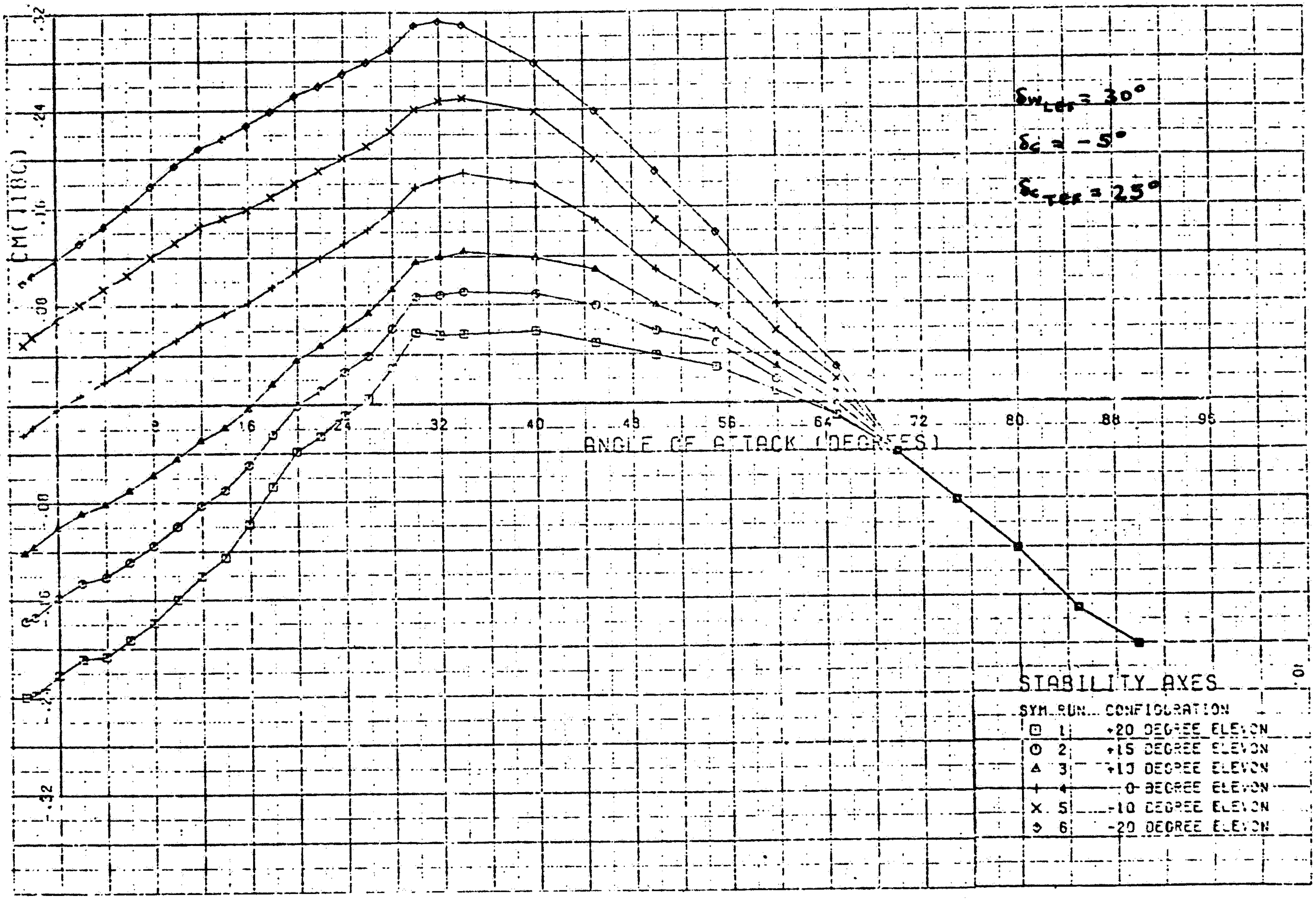


Figure 5-15 - SF-121 Untrimmed Pitching Moment Coefficients

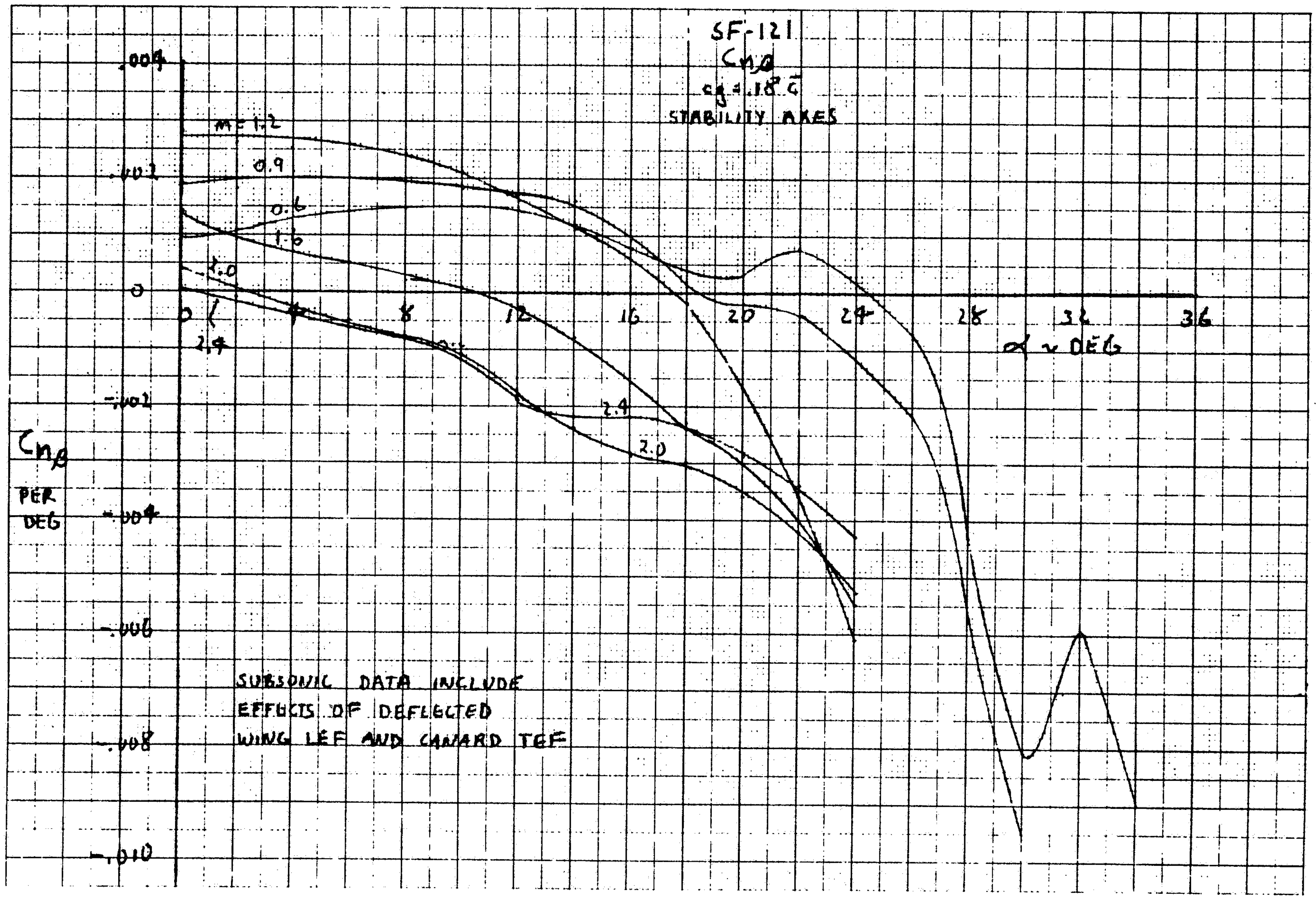


Figure 5-16 - SF-121 Directional Stability Derivatives

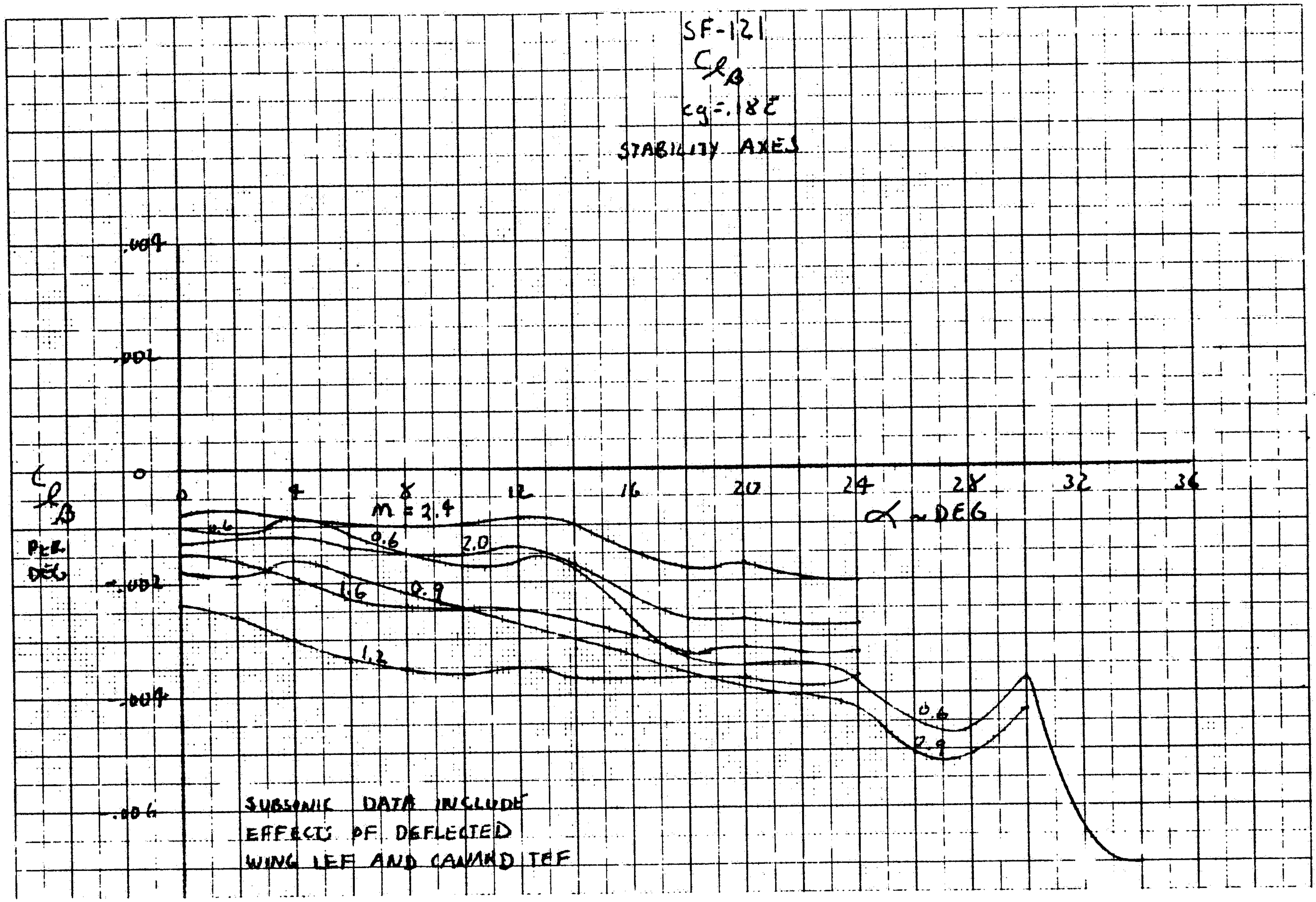


Figure 5-17 - SF-121 Lateral Stability Derivatives

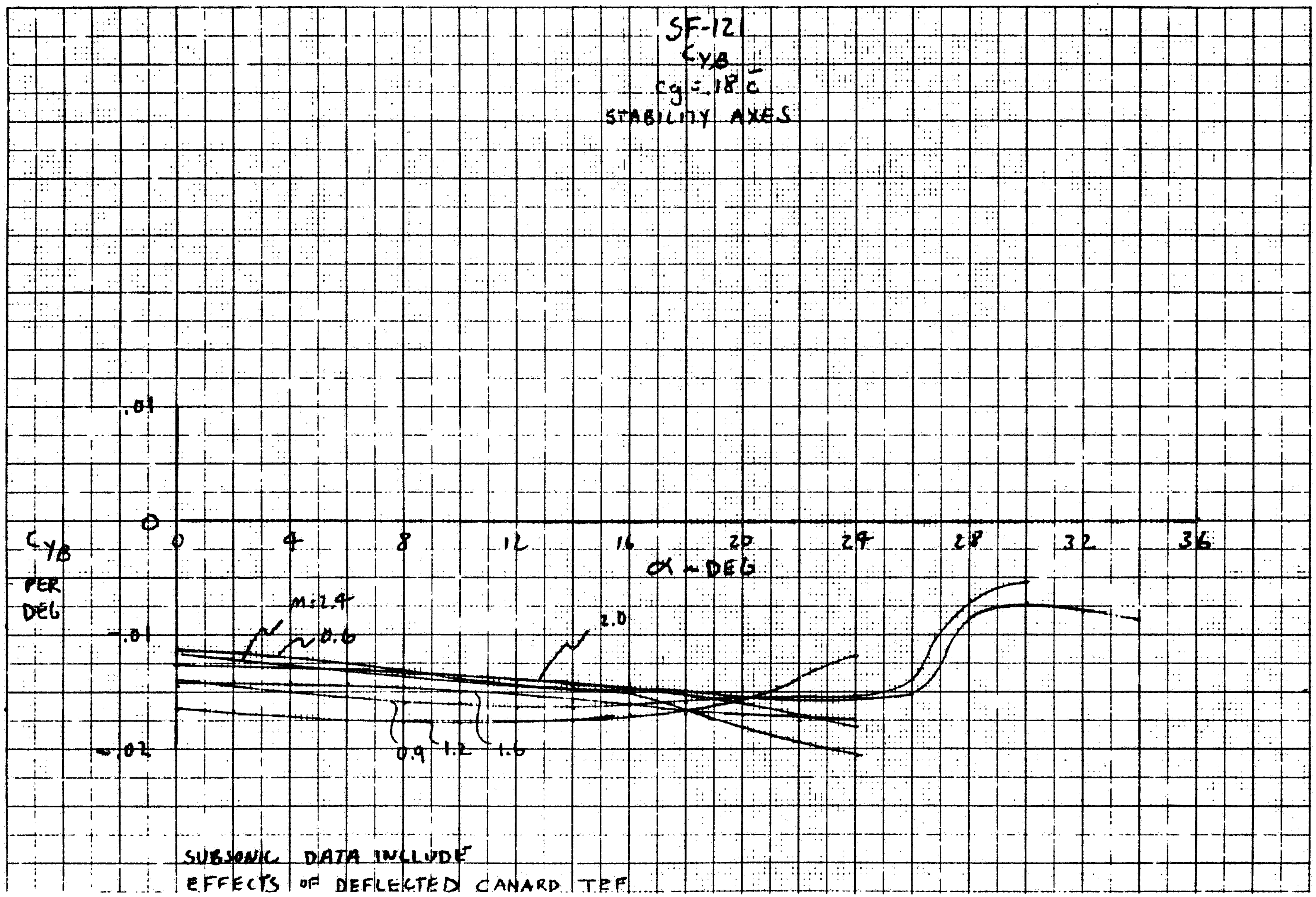


Figure 5-18 - SF-121 Side Force Derivatives

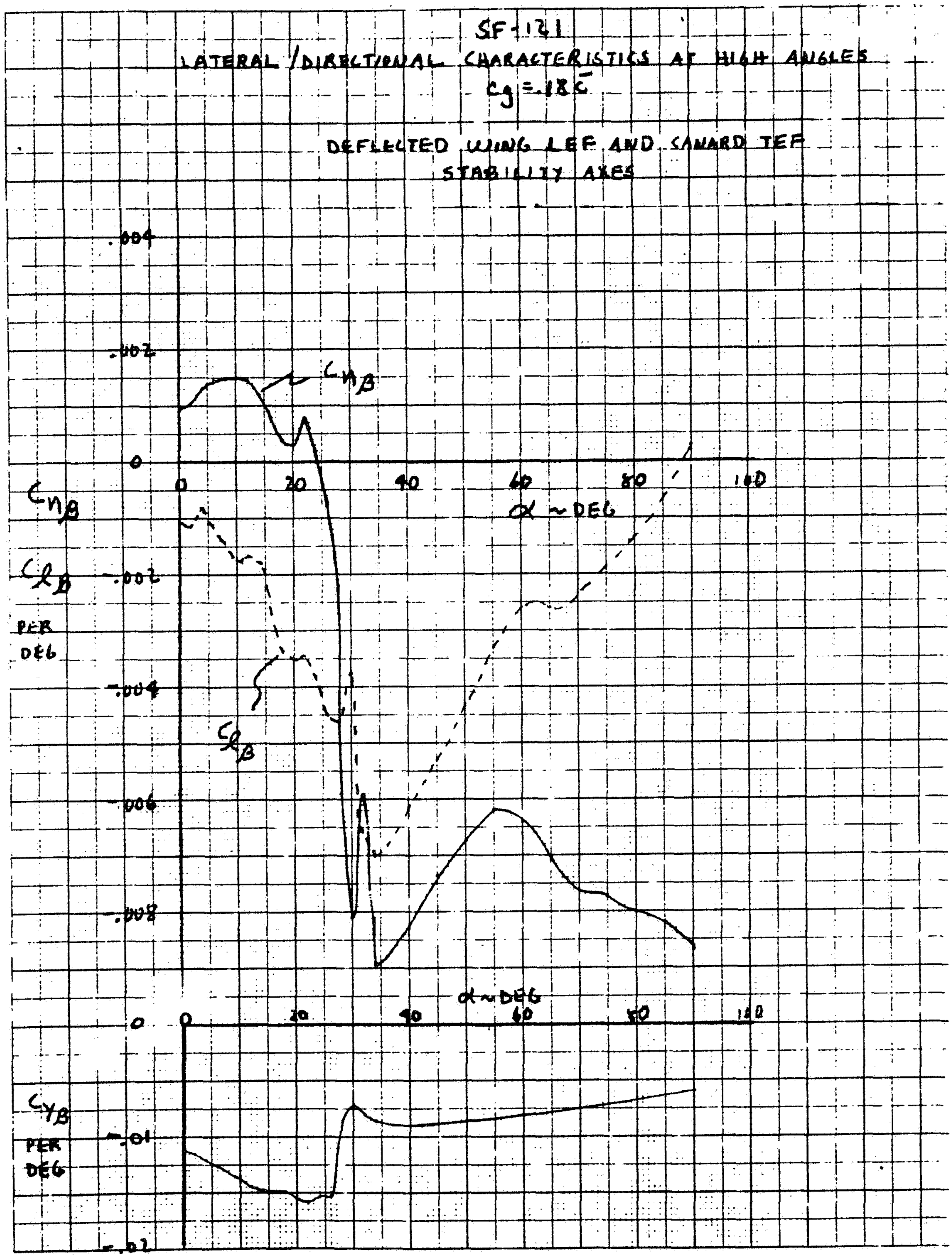


Figure 5-19 - Lateral/Directional Characteristics at High Angle of Attack

5.2.2 Control Surface Effectiveness

Figures 5-20 and 5-21 show SF-121 aileron and rudder control effectiveness. The characteristics are based on the test data from References 3 and 6. The methods of Reference 7 were used to obtain corrections for test model and SF-121 geometry differences and for Mach number effects were necessary. Extension of the $M = 0.6$ data to $\alpha = 90$ degrees (for transition analysis) was made using the trends in Reference 8 for the delta wing configuration, with the results presented in Figure 5-22. The corrections for high control surface deflections (Figure 5-23) are from Reference 9.

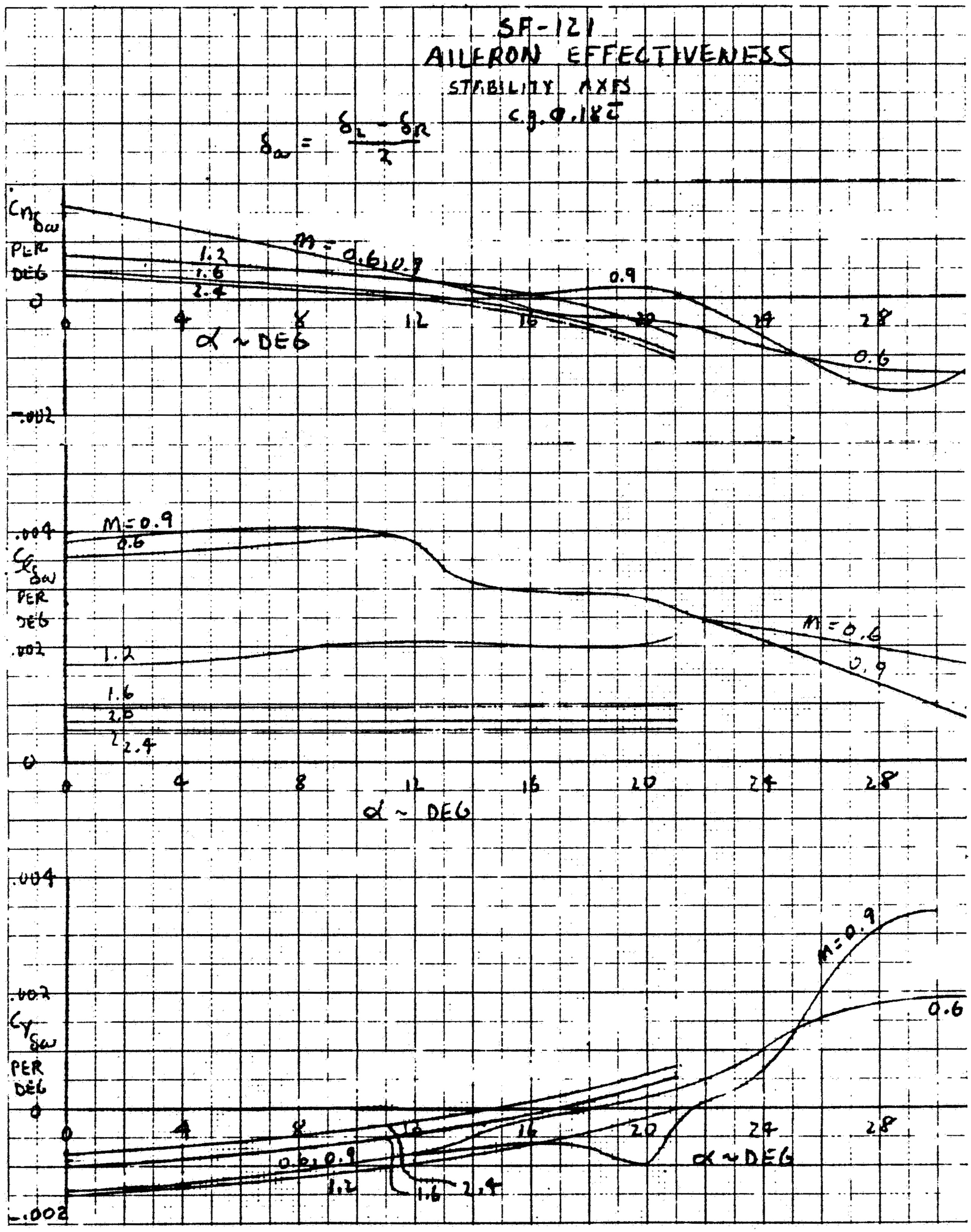


Figure 5-20 - SF-121 Aileron Effectiveness

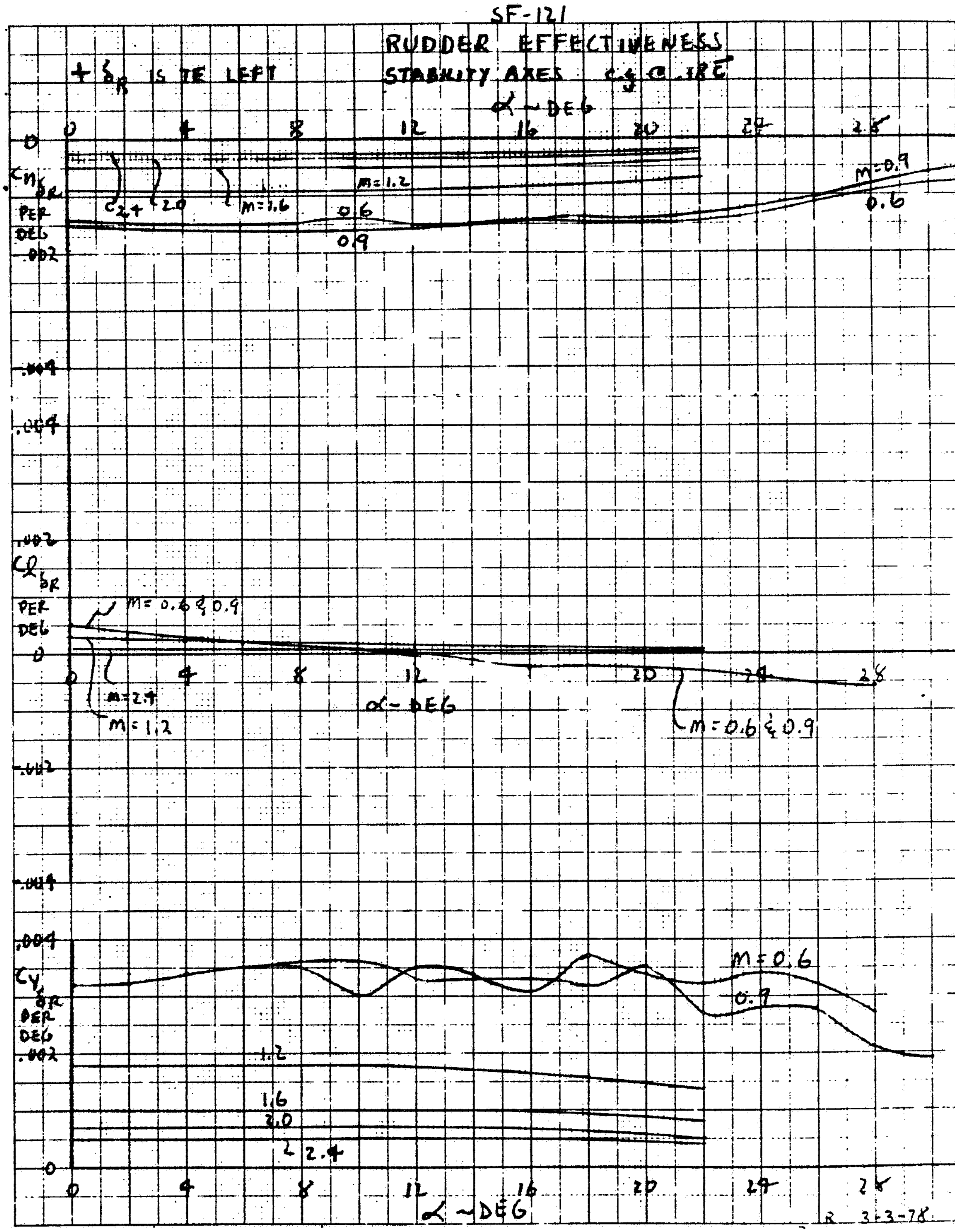


Figure 5-21 - SF-121 Rudder Effectiveness

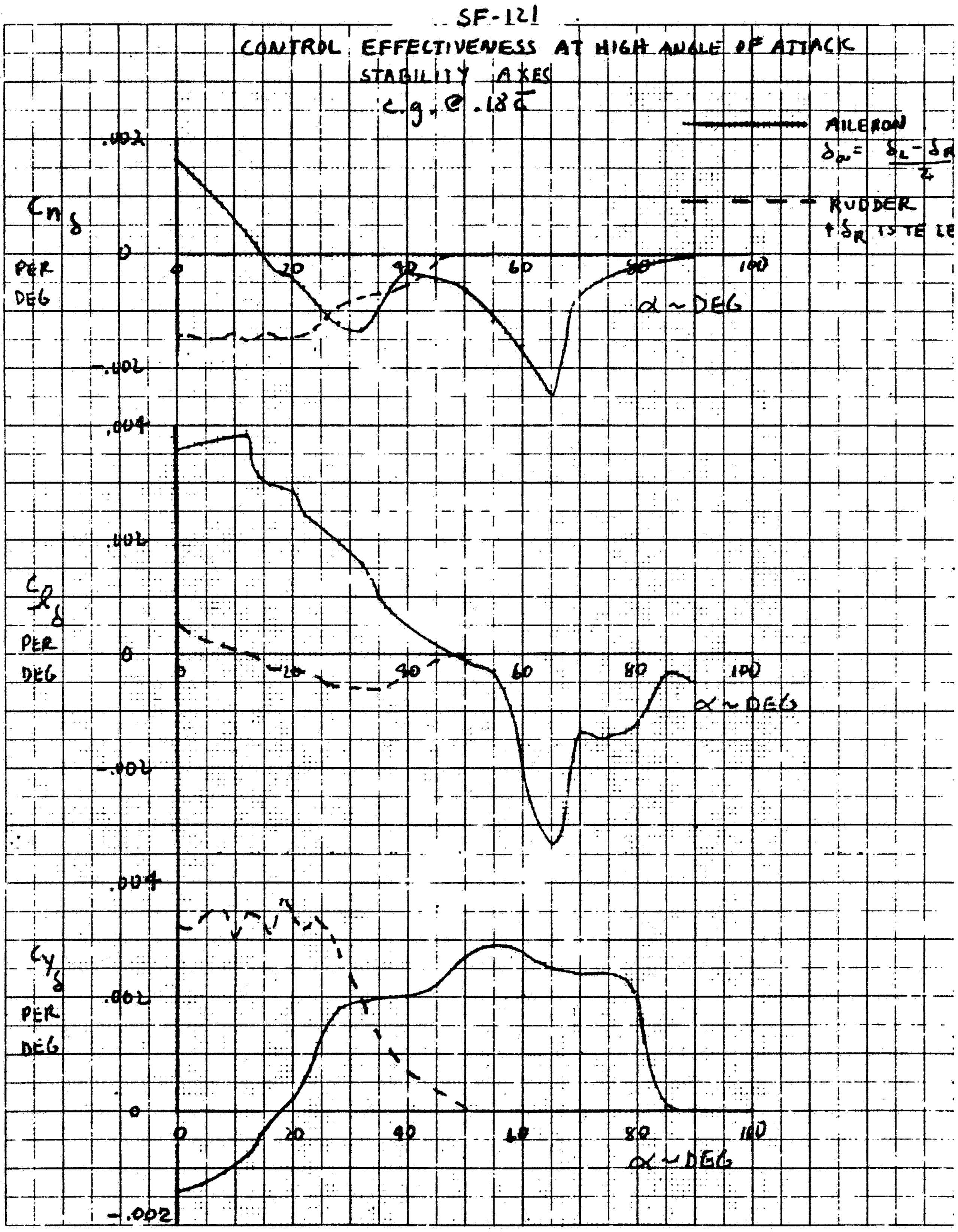


Figure 5-22 - Control Effectiveness at High Angle of Attack

CORRECTION FOR LIFT EFFECTIVENESS OF PLAIN TRAILING-EDGE FLAPS AT HIGH FLAP DEFLECTIONS

DATCOM

SF-121	$(c_f/c)_{AVERAGE}$
AILERON	.34
RUDDER	.28

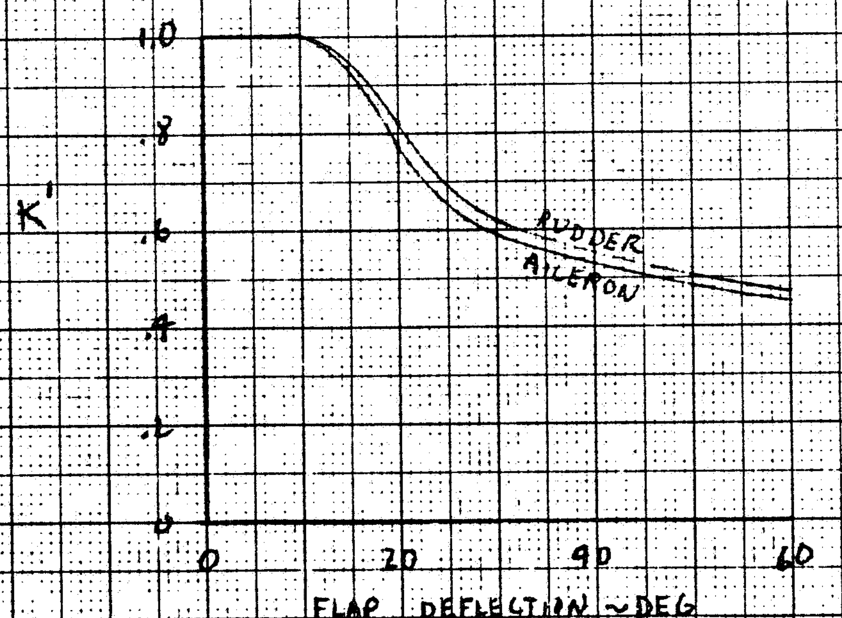


Figure 5-23 - Flap Lift Effectiveness Correction

6.0 PROPULSION

This section omits certain propulsion cycle parameters and performance to protect the proprietary rights of Pratt & Whitney Aircraft Division of United Technologies. Additional information is contained in Appendix B.

6.1 ENGINE DESCRIPTION

The SF-121 is powered by two advanced technology mixed flow augmented turbofan engines. A bypass ratio of 1.0 was selected for several reasons centering around the multimission role of the SF-121, including:

- o Substantially improved subsonic loiter time
- o Moderately improved subsonic radius of action
- o Minimal impairment of Supersonic Intercept radius
- o Reduced IR signature without augmentation
- o Slightly higher thrust to weight
- o Higher augmentation ratio
- o Slightly milder footprint

One distinct disadvantage of the BPR = 1.0 engine is relatively high static thrust loss due to reaction jet compressor bleed.

Installed performance and weight were estimated using a Pratt & Whitney parametric performance computer program, with Vought installation factors. A weight increment was added for the thrust vectoring system. The uninstalled weight of the SF-121 point design engine is 1,749 pounds (793 kg). Figure 6-1 shows the corresponding physical characteristics. Installed afterburner thrust for the single engine vertical landing condition (SLS, Tropical Day) is 15,128 lb (67,312 N).

6.2 AIR INDUCTION SYSTEM

The side inlets are horizontal ramp two dimensional types. They are a three shock fixed geometry configuration with scheduled throat boundary layer bleed and a Mach 1.6 design point. Blow-in doors are provided for low speed operation. Capture area for the SF-121 point design is 555 in^2 (3,580 cm^2) per inlet. Figure 6-2 displays inlet total pressure recovery as a function of Mach number.

The inlet configuration was selected with high angle of attack performance in mind. A recent study of this inlet indicated satisfactory pressure recovery

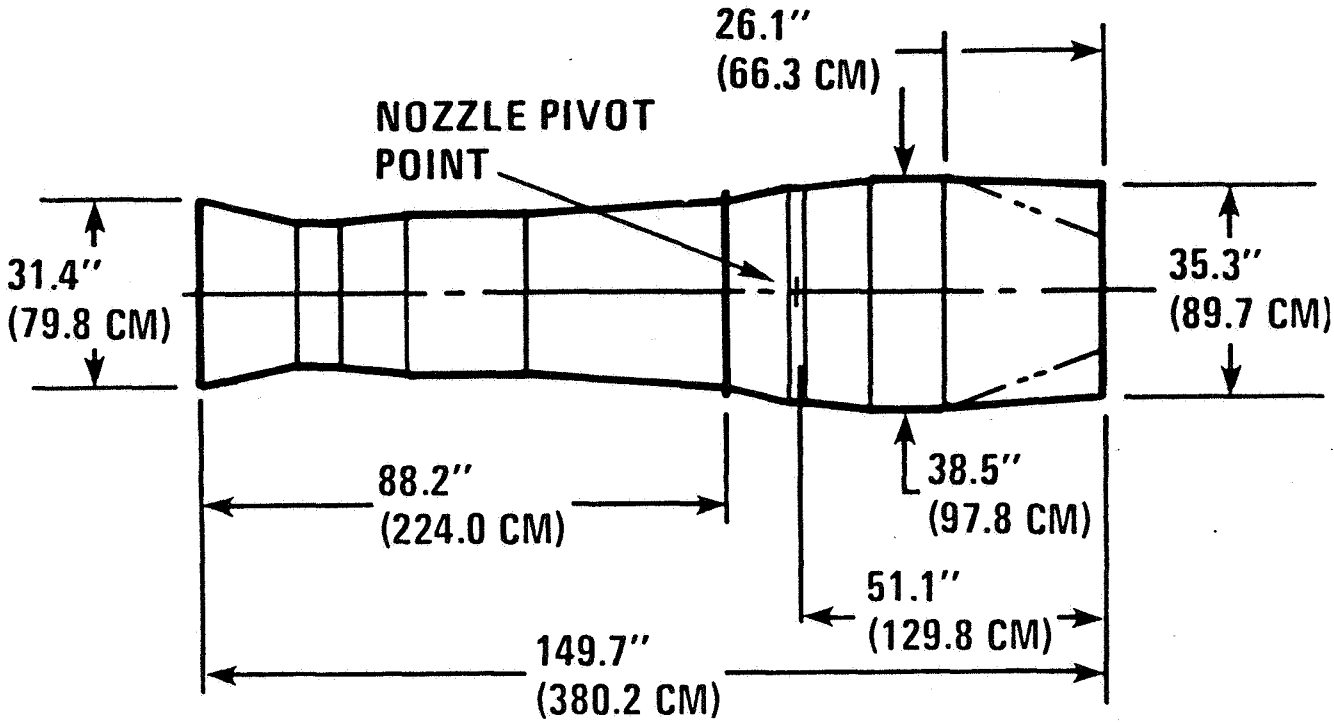


Figure 6-1 - Point Design Engine Dimensions

and distortion index can be achieved through VATOL transition to hover. If required, a simple flap on the lower inlet lip can produce substantially higher recovery and reduce distortion at the compressor face by 50 percent. The effects of sideslip, however, have not been investigated.

6.3 ATTITUDE CONTROL SYSTEM

The high speed flight propulsion system also provides all the thrust required to support the aircraft in vertical attitude hover, without requiring any operating mode change from the high speed flight configuration. However, powerful control moments must be supplied by the propulsion system to balance the aircraft in the vertical attitude and control the flight path during transaction. The required control power is achieved by vectoring the entire efflux of the aft-mounted engines through a small deflection angle. No more than 15 degrees deflection is necessary due to the large gross thrust vector and neglible turning losses, other than the deflection angle term (3.4 percent at 15 degrees).

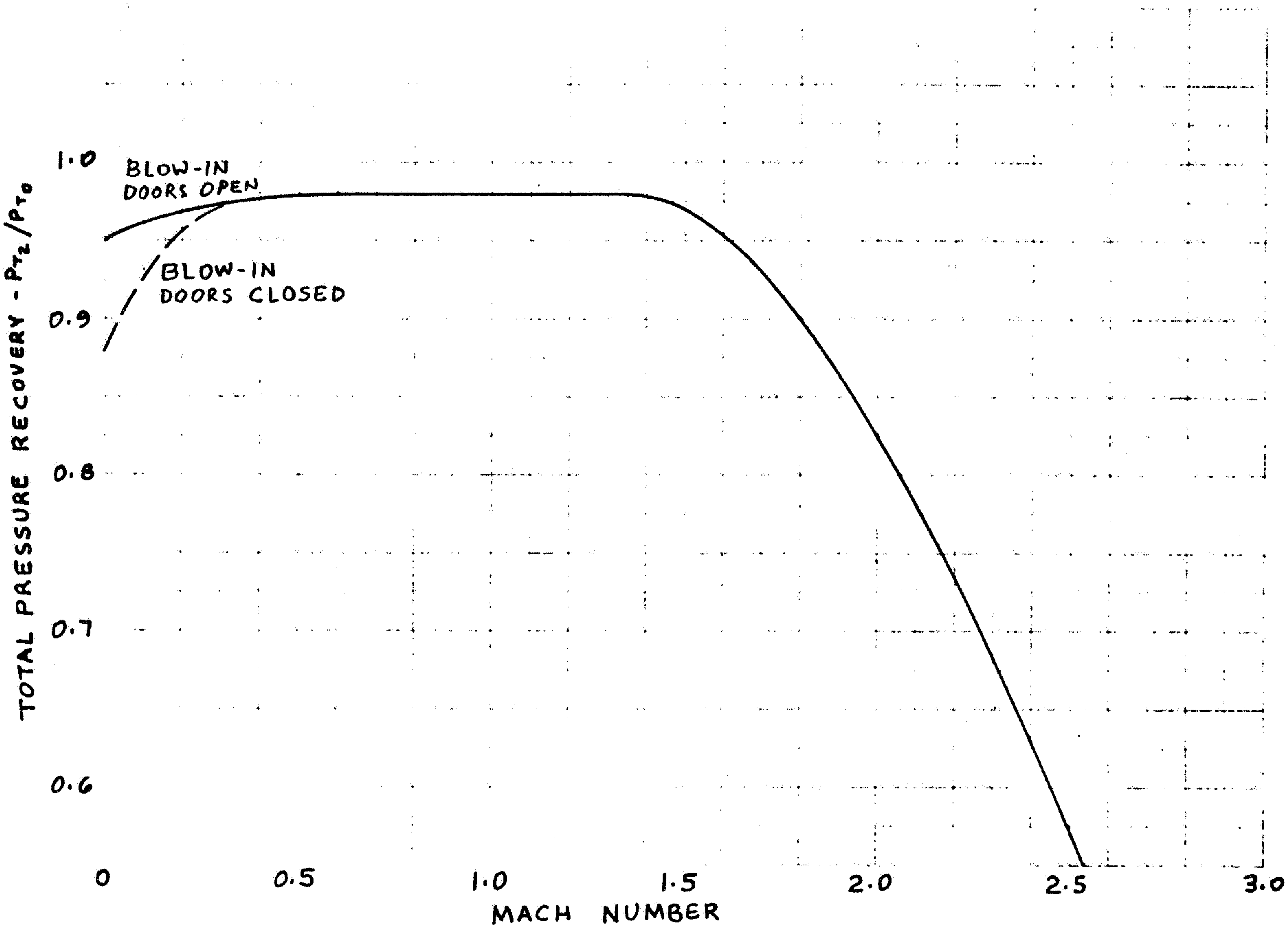


Figure 6-2 - Inlet Total Pressure Recovery

Thrust vectoring in pitch and yaw axes is achieved by a gimbal mechanism between the nozzle assembly and the afterburner casing. The gas flow path is essentially unchanged, and the gimbal mechanism need not increase the total length of the engine. Indeed, the gas seals will receive slightly lower heat input by maintaining constant length. However, the pivot point should be as close to the nozzle exit plane in order to maximize effective moment arm and minimize movable mass and physical travel.

The axisymmetric gimballed (GAX) nozzle was selected for the SF-121 over jet vanes and two-dimensional nozzles. Detailed studies of vectoring nozzles (Reference 10) indicate that the gimballed axisymmetric baseline was lighter than 2-D concepts and had generally better thrust performance (dependent on configuration integration and flight conditions). Development cost is likely to be lower for the GAX, expecially if an existing engine/ nozzle is adapted.

A serious limitation of 2-D vectoring nozzles for VATOL application is the ability to vector in pitch only. Adding a lateral axis would entail additional complexity and weight. Hybrid systems, such as a 2-D pitch nozzle and yaw bleed jets or jet vanes are possible alternatives, but may not provide sufficient control power to cope with an engine failure in a twin engine configuration. In addition, systems which require high compressor bleed airflow restrict the choice of propulsion bypass ratio.

Jet vanes are less efficient than vectoring nozzles and pose several design and operating problems when applied to an afterburning engine:

- o Exposure to afterburner temperatures
- o Thrust loss (drag) at zero vector angle
- o Nozzle area variations changing area in jet or complicating mounting provisions
- o Probable height IR and radar signature.

For these reasons the GAX approach was selected for the SF-121. The two axis gimbal system provides compensation for an engine failure by a lateral deflection which directs the remaining thrust vector through the airplane center of mass. The close spacing of the Superfly engines holds the required deflection to less than eight degrees. The gimbals are installed with an eight degree outward bias so full ± 15 degree yaw control is still available in an engine out situation. For normal twin engine operation the nozzles are deflected inward to cancel the bias and minimize base drag.

The twin engine arrangement can generate all required roll control power in transition and hover by differential nozzle deflections. This action entails minimal thrust loss and is easily harmonized with pitch and yaw commands. Unfortunately, the loss of one engine means a loss of roll control. The SF-121 uses a reaction jet system for roll control. Each engine supplies high pressure compressor bleed air to reaction jets at the wingtips. The reaction system is adequate for all flight conditions yet examined, but does cause a significant thrust loss which is reflected in engine size. For two engines VATOL operation the SF-121 phases differential nozzle deflection and reaction jets for optimum response and flying qualities. This extra control power is used to advantage during vertical takeoff at maximum weight; the presence of external stores can more than triple clean airplane roll inertia. Despite such an inertia increase, takeoff is less constraining on engine size than single engine landing. Section 8.4.1 quantifies VATOL control power requirements.

The VATOL attitude control system can be engaged at any point in the flight envelope, with payoffs in transonic combat agility. There are other benefits, including:

- An independent backup to the entire aerodynamic control system
- Augmented total control power for combat, particularly at extreme angle of attack, where aerodynamic controls may become ineffective and stall departure problems occur.
- Induced lift due to nozzle deflection.

6.4 PERFORMANCE IN TRANSITION

Control power and thrust available in transition are paramount to achieving a satisfactory VATOL aircraft design. Installed gross thrust is degraded by bleed required for roll jet reaction control. Gross thrust available for one and two engines as a function of bleed percentage and Mach number is presented in Figure 6-3. Effective moment arm vs. nozzle deflection is in Figure 6-4. Roll jet reaction thrust available at corresponding conditions is shown in Figure 6-5. All bleed performance shown herein is based on bleed from maximum thrust levels. Percentages shown are not applicable to partial power settings. Attitude control studies reported in Section 8.3 assumed that the reaction jet thrust vs. gross thrust

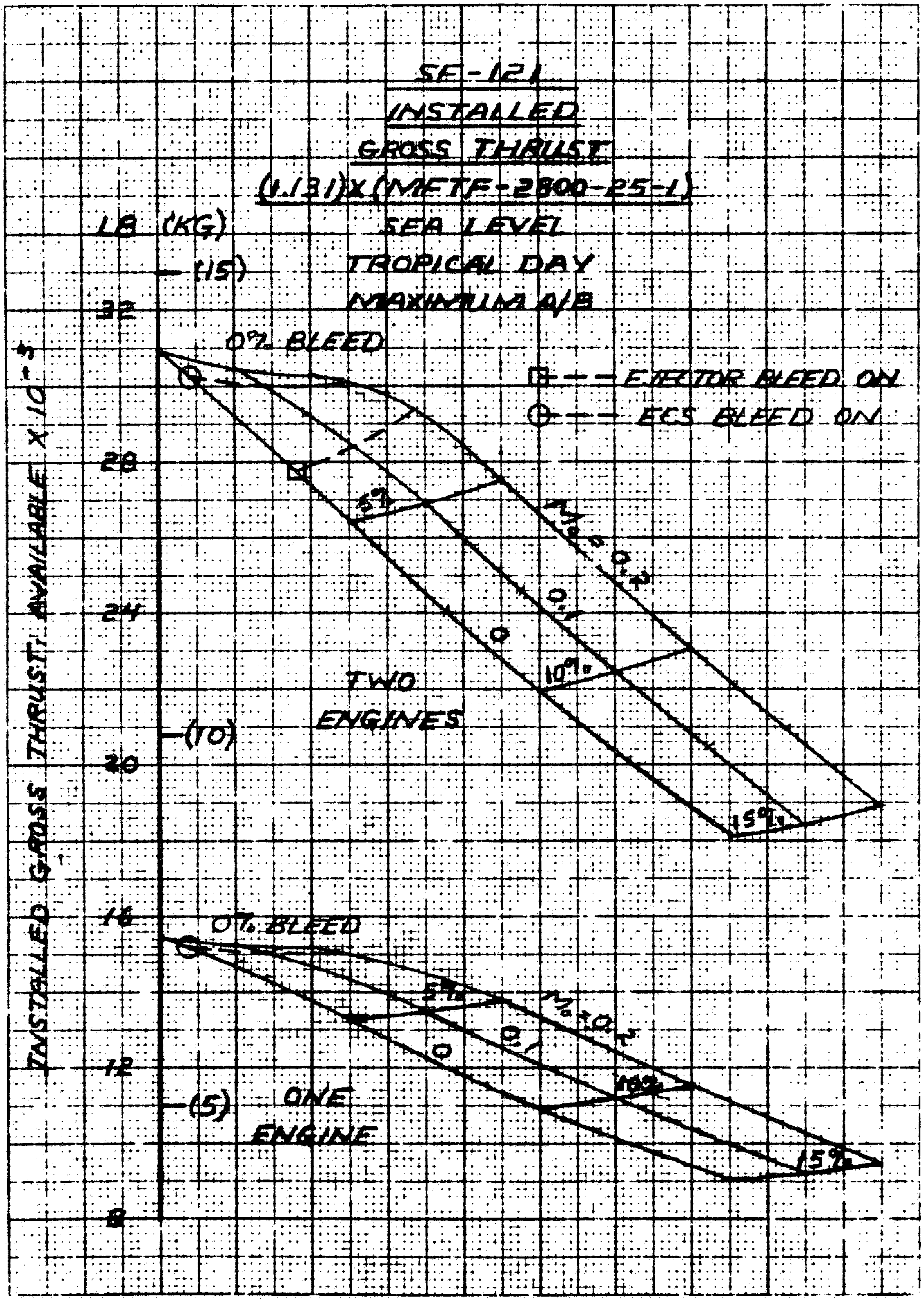


Figure 6-3 - Gross Thrust Available for Transition

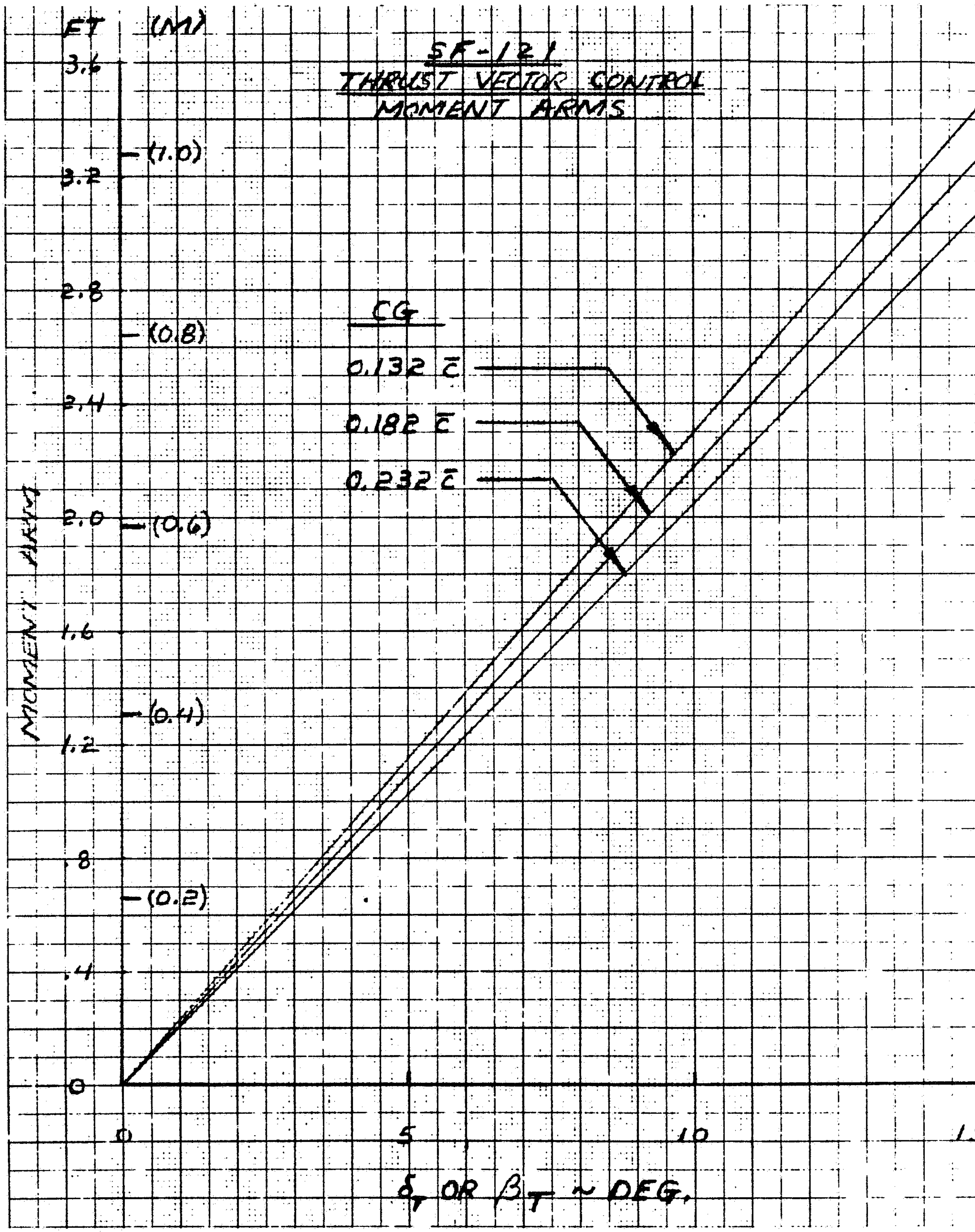


Figure 6-4 - Effective Thrust Vector Moment Arm

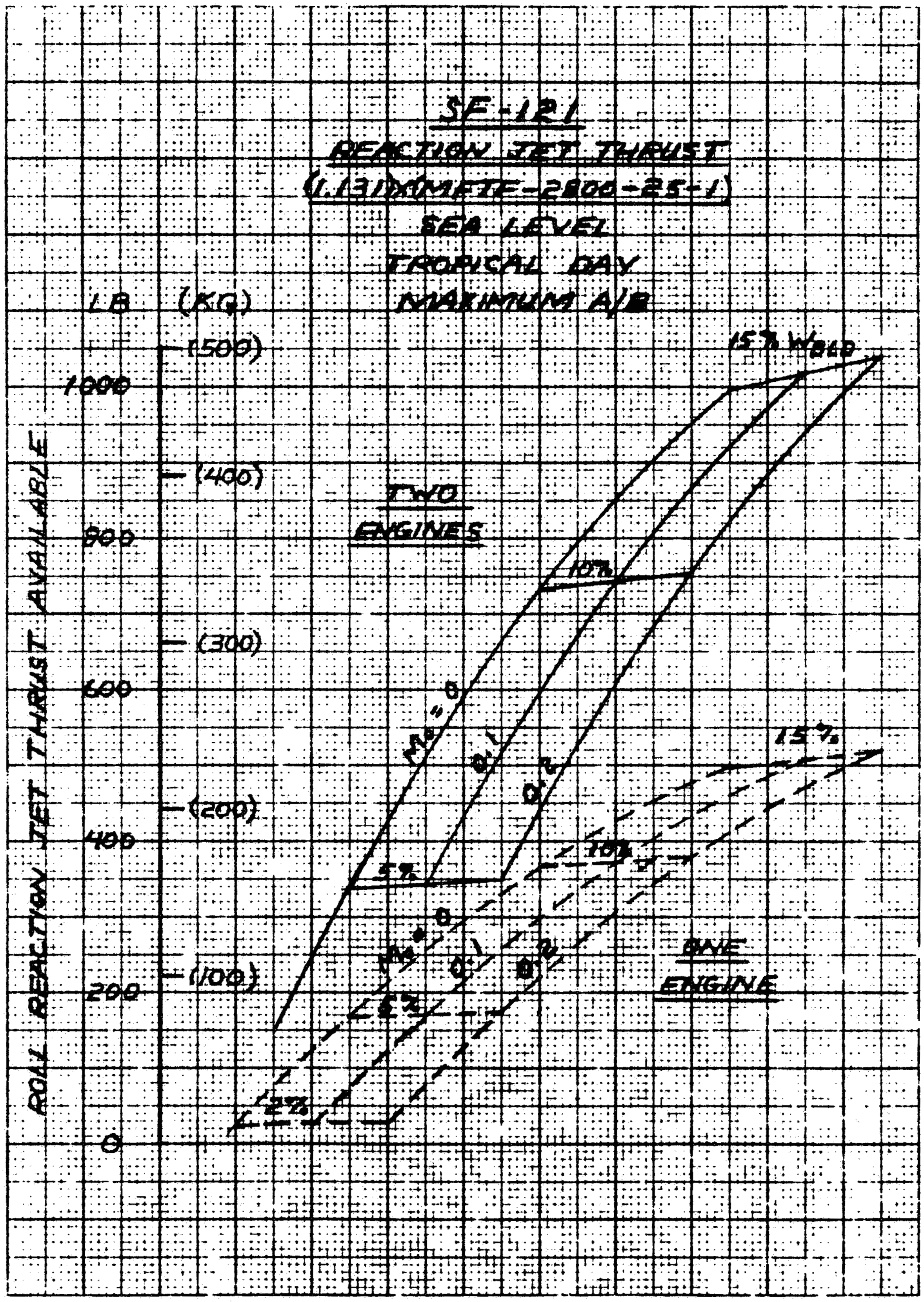


Figure 6-5 - Roll Jet Thrust Available

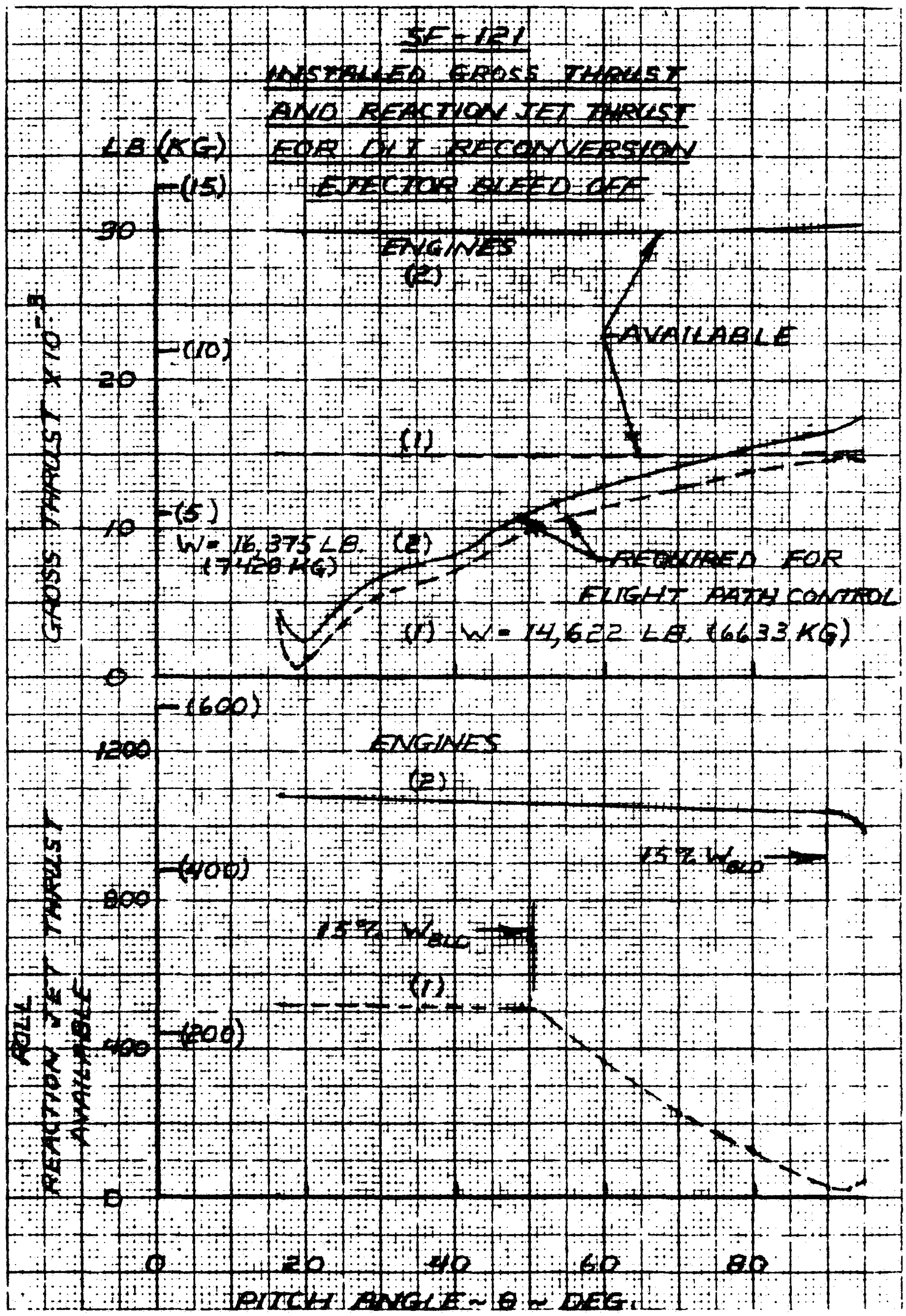


Figure 6-6 - Thrust Available and Required for Reconversion relationships were constant at all power settings. Figure 6-6 illustrates gross thrust available versus required and reaction jet thrust available for design mission normal and single engine reconversions. Ample gross and reaction jet thrust is available for the normal DLI design landing. Gross and reaction jet thrust for single engine hover is inadequate due to the design T/W = 1.03 used for the SF-121 sizing. A higher design T/W margin is recommended. More complete description of suggested design T/W for the SF-121 type aircraft is given in Section 8.3.1.

7.0 AIRCRAFT DESIGN

Section 4.0 contained a detailed description of the SF-121 aerodynamic configuration and geometry. This section focuses on a presentation of the point design mass properties for a range of loading conditions, and briefly describes internal systems which influence weights and inertias. Since the Phase I study philosophy was to concentrate on aerodynamic issues, the SF-121 designers relied on recent Vought IR&D experience for VATOL systems inputs. Appendix C provides additional background abstracted from a recent Vought report (Reference 11).

7.1 FLIGHT CONTROLS

Aerodynamic control is achieved through a quadriplexed digital fly by wire control system. Trailing edge flaps on both canard and wing operate in unison to implement longitudinal and lateral commands, with optimal phasing

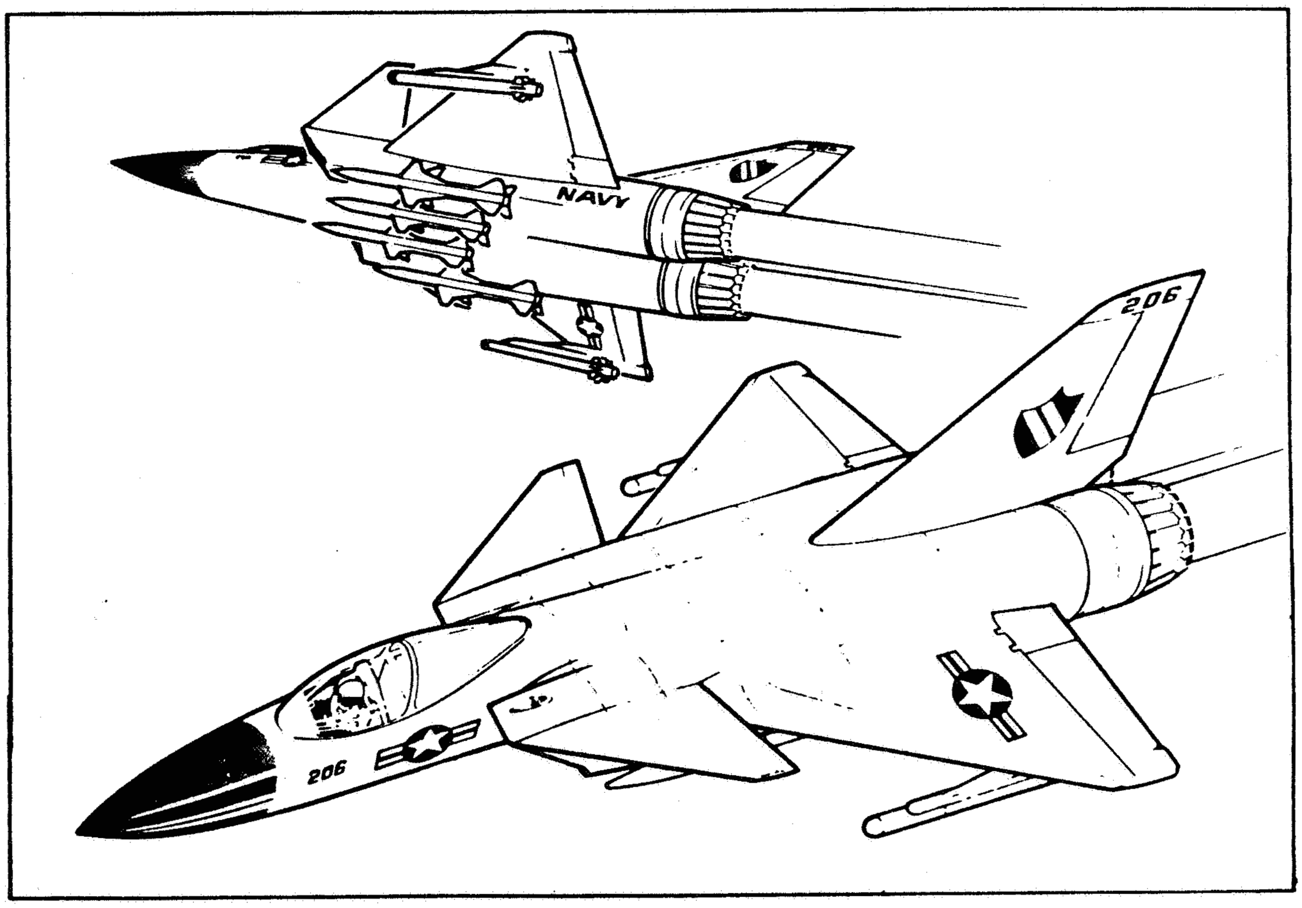


Figure 7-1 - SF-120 Series Superfly VATOL Fighter throughout the flight envelope. Full span leading edge flaps are automatically phased to maintain the optimal camber; the constant chord L.E. flap on the highly tapered wing introduces proportionally greater camber changes to the outboard region to enhance maneuver characteristics. The inboard wing trailing edge forms split flap speedbrakes. The centerline fin has a conventional rudder.

7.2 STRUCTURAL DESIGN

7.2.1 Wing and Empennage

The wings attach to the sides of the fuselage and are made almost entirely of composite materials. Insulation is required around the supply ducts to the wingtip roll jets to protect the aft wing box from high temperature compressor bleed air. (In the original design the air was ducted through the elevons, which required they be made of titanium and stainless steel.) The leading edge flaps are made from polyimide/graphite composite with metal erosion strips on the leading edges.

Canard and fin are generally similar to the wing in construction, but are lightly loaded and contain minimum gauge materials. They also attach directly to the fuselage structure.

7.2.2 Fuselage

Length exclusive of the exposed exhaust nozzles is 42.25 feet. The midsection is a rectangular box structure divided into bays by bulkheads which carry external stores, landing gear and wing bending loads. The exposed wing panels attach to lugs on the fuselage bulkheads. The space behind the cockpit contains the aft avionics bay and environmental control system components. Aft of this section is the weapons installation on the underside and fuel tanks between and above the air induction system. A structural firewall separates the engine compartments and distributes vertical tail and engine loads. A remote accessory package is shaft driven by both engines.

7.2.3 Fuel System

An inflight refueling probe retracts into the topside of the right inlet nacelle. The probe extends up, out and forward with the tip in clear view of the pilot. The wing structural box is an integral fuel cell. Fuselage fuel cells extend from the nose gear bulkhead to the engine ducts. A rear fuel cell can be located above the engines forward of the hot section. This tank is not required to contain the 8,077 pounds of JP-5 required for the SF-121 design mission.

7.2.4 Landing Gear

Conventional tricycle landing gear with wheels and brakes is employed to give the Superfly STO and CTOL capability, as well as to facilitate deck handling. The tires and oleo struts absorb up to 15 feet per second contact in either conventional landing or VATOL modes. The main gear consists of vertical stroke cantilever struts which retract aft to lay flat beneath the engines. The wheels shield the engine from ground fire, and the MLG wells provide access to the engines without requiring additional assess doors. The nose landing gear is integrated with the VATOL capture mechanism. The SF-121 does not have catapult and arresting provisions.

7.2.5 Internal Gun

The M61A1 20 mm six-barrel gun with 600-round capacity drum was selected for the SF-121. The rationale was that a much heavier gun such as the 30 mm GAU-8 imposes too great a performance penalty on a lightweight VSTOL B. Increasing caliber at the expense of muzzle velocity was undesirable for the air superiority function. Thus a new gun for VSTOL B is likely to be a compromise, trading off firepower and weight; a lightweight 25 mm three-barrel gun using caseless ammunition, for example. Such a weapon and ammunition would be similar in size and weight to the M61, and may even be designed for retrofit. The detailed information available on the M61 contributes to a credible installation and facilitate comparisons with other concepts.

The M61A1 weighs 250 pounds. A lightweight 600-round drum and all associated components add another 274 pounds.

7.2.6 Tilting Seat

The single place crew station is provided with a movable ejection seat which tilts forward during vertical attitude operation. The primary purpose is to assist the pilot in holding his head in an upright position to maintain conventional vestibular cues. The decision not to use a complete tilting nose section was based on X-13 flight test experience, which showed that direct forward visibility was not required for repeatable vertical attitude dockings.

7.2.7 Materials

Composite material usage on the Superfly is projected to save 20 percent of the structural weight. Vought has recently completed a detailed analysis of the application of composites for the Type A VSTOL. Most of the materials technology is applicable to this aircraft. The 1995 IOC projected for VSTOL fighter attack will permit an additional five years of materials development beyond Type A technology.

Composite material application is separated into three major levels depending on the state-of-art and the status of supporting R&D efforts.

Level I — Components are composite material applications where production capability and payoff has been proven. No new R&D programs are necessary. Level I components could be incorporated into a near-term Type B prototype (1980 design date).

Level II — Components are composite material applications where proof of concept has not been thoroughly demonstrated, however, necessary R&D efforts are either currently being funded or funding is planned. Level II components will be available for design in the 1985 time period. Some Level II components could be available for a near term Type B prototype.

Level III — Components are potentially high payoff composite material applications for which little or no design experience exists and for which R&D funding is just now being planned. Most Level III components will be available for design in the early 1990's.

Figure 7-2 shows the weight payoff for the three application levels and identifies the components considered for each level.

7.3 MASS PROPERTIES

The component weights for the SF-121 were derived by semi-analytical analyses, statistical equations or vendor quoted values. The effect of technological improvements anticipated by 1990 are discussed in the following paragraphs and are reflected in the group weight summary shown in Table 7-1.

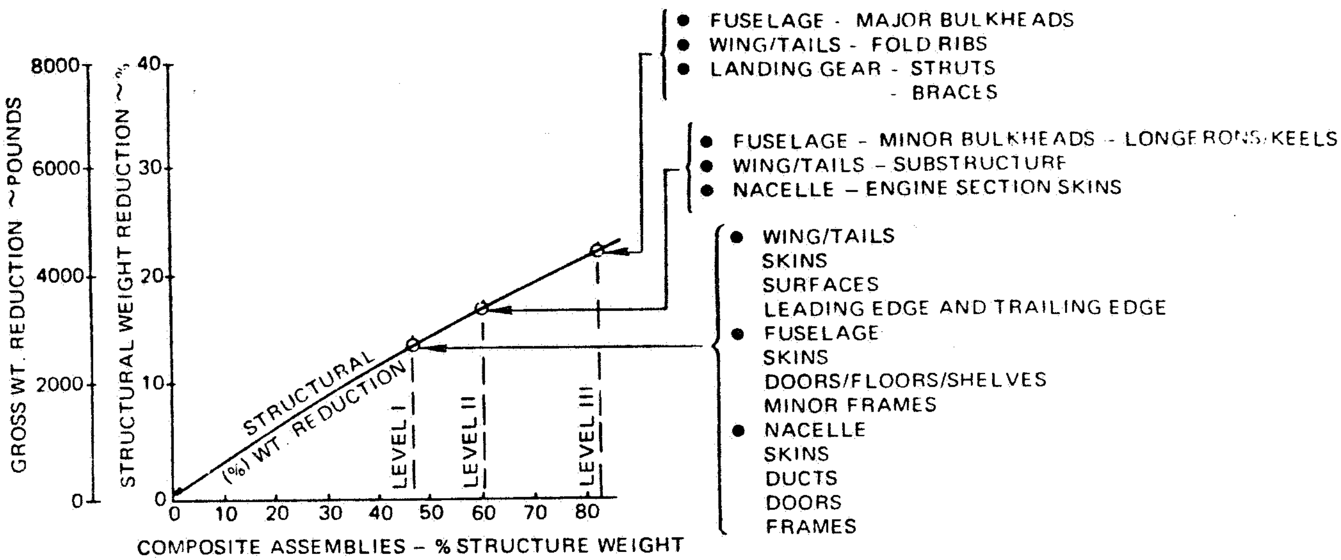


Figure 7-2 - Weight Payoff for Composite Materials

Tables 7-2 through 7-7 detail SF-121 center of gravity and moments of inertia about all axes for a range of external store loadings and fuel states.

Table 7-1 - SF-121 Group Weight Statement

SHORT GROUP WEIGHT STATEMENT
NAVAIR FORM 13060/3 (4-72)

DATE

	MODEL	SF-121 LBS	KG			
1	WING	1,245	565			
2.	ROTOR	-				
3	TAIL	513	233			
4.	BODY	2,154	977			
5.	ALIGHTING GEAR	711	323			
6	ENGINE SECTION	510	231			
7	PROPULSION	4,375	1,984			
8	ENGINE INSTALLATION	(3,497)	(1,586)			
9	ACCESS. GR. BOXES & DRIVE	(100)	(45)			
10	EXHAUST SYSTEM	(-)	(-)			
11.	ENGINE COOLING	(-)	(-)			
12	WATER INJECTION	(-)	(-)			
13.	ENGINE CONTROLS	(34)	(15)			
14	STARTING SYSTEM	(60)	(27)			
15.	PROPELLER INSTAL.	(-)	(-)			
16	SMOKE ABATEMENT	(-)	(-)			
17	LUBRICATION SYSTEM	(-)	(-)			
18	FUEL SYSTEM	(684)	(310)			
19	DRIVE SYSTEM	(-)	(-)			
20	JET DRIVE	(-)	(-)			
21	FLIGHT CONTROLS	710	322			
22	AUX POWER PLANT	100	45			
23	INSTRUMENTS	134	61			
24	HYDRAULICS & PNEUMATICS	266	121			
25	ELECTRICAL	279	127			
26	AVIONICS	1,000	454			
27	ARMAMENT	352	160			
28	FURNISHINGS & EQUIPMENT	215	98			
29	AIR CONDITIONING	141	64			
30	ANTI-ICING	(-)	(-)			
31	PHOTOGRAPHIC	(-)	(-)			
32	LOAD & HANDLING	25	11			
33	MANUFACTURING VARIATION	-	-			
34	WEIGHT EMPTY	12,730	5,774			
35	CREW (NO.)	200	91			
36	PASSENGERS (NO)	-	(-)			
37	FUEL-UNUSABLE	82	37			
38	FUEL-INTERNAL	8,077	3,664			
39	FUEL-EXTERNAL	-	-			
40.	OIL	60	27			
41.	FUEL TANKS AUX.	-	-			
42	BAGGAGE	-	-			
43	CARGO. TROOPS	-	-			
44	GUNS	181	82			
45	AMMUNITION - 400 RDS 20MM	224	102			
46	EQUIPMENT (O_2, SURVIVAL KITS)	42	19			
47	WEAPONS INSTALLATION					
48	BOMBS					
49.	ROCKETS, MISSILES					
50.	-AIM-9L (2)	620	281			
51.	-AIM-7F (2)	1,160	526			
52.						
53.	PHOTOGRAPHIC	-	-			
54	MISCELLANEOUS	-	-			
55	USEFUL LOAD	10,646	4,829			
56.	GROSS WEIGHT	23,376	10,603			

Table 7-2 - SF-121 Balance and Inertial Data
Design Mission Stores, Gear Up

FUEL STATE	WEIGHT	CENTER OF GRAVITY			MOMENT OF INERTIA SLUGS FT^2			"θ" RADS
		F.S.	% MAC	W.L.	ROLL	PITCH	YAW	
Full	23,375	401.0	17.9	101.6	10,388	57,289	65,693	-.009
7,000 lb.	22,299	401.1	18.0	100.9	9,195	56,194	63,515	-.005
6,000 lb.	21,299	401.1	18.0	100.8	8,538	55,210	61,905	-.003
COMBAT 4,845 lb.	20,144	401.2	18.0	100.8	8,032	54,091	60,283	-.003
4,000 lb.	19,299	401.3	18.1	100.0	7,840	53,615	59,718	-.004
3,000 lb.	18,229	401.3	18.1	99.7	7,686	53,184	59,185	-.006
2,000 lb.	17,299	401.4	18.1	99.8	7,550	52,772	58,652	-.006
1,000 lb.	16,229	401.5	18.2	99.9	7,383	52,382	58,167	-.006
EMPTY	15,229	401.6	18.2	100.1	7,188	52,127	57,826	-.005

Table 7-3 - SF-121 Balance and Inertial Data
Design Mission Stores, Gear Down

FUEL STATE	WEIGHT	CENTER OF GRAVITY			MOMENT OF INERTIA SLUGS FT^2			"θ" RADS
		F.S.	% MAC	W.L.	ROLL	PITCH	YAW	
FULL	23,375	400.5	17.6	101.0	10,744	57,526	65,862	-.007
7,000 lb.	22,299	400.5	17.6	100.2	9,545	56,426	63,684	-.003
6,000 lb.	21,299	400.5	17.6	100.0	8,886	55,441	62,075	-.001
COMBAT	20,144	400.5	17.6	100.0	8,392	54,355	60,497	-.001
4,000 lb.	19,299	400.5	17.6	99.2	8,184	53,842	59,888	-.002
3,000 lb.	18,299	400.6	17.7	98.9	8,028	53,410	59,355	-.003
2,000 lb.	17,299	400.6	17.7	98.9	7,892	52,998	58,823	-.004
1,000 lb.	16,299	400.6	17.7	99.0	7,726	52,609	58,338	-.004
EMPTY	15,299	400.7	17.7	99.1	7,532	52,356	57,998	-.002

Table 7-4 - SF-121 Balance and Inertial Data
Design Mission Stores Off, Gear Up

(AIM-9L Launchers and Pylons On)

FUEL STATE	WEIGHT	CENTER OF GRAVITY			MOMENT OF INERTIA SLUGS FT^2			"θ" RADS
		F.S.	% MAC	W.L.	ROLL	PITCH	YAW	
FULL	21,738	401.7	18.3	102.8	8,559	55,909	62,802	-.012
7,000 lb.	20,662	401.8	18.4	102.1	7,373	54,821	60,624	-.008
6,000 lb.	19,662	401.9	18.4	101.9	6,718	53,838	59,014	-.006
COMBAT 4,845 lb.	18,506	402.0	18.5	102.0	6,212	52,719	57,392	-.006
4,000 lb.	17,662	402.1	18.5	101.3	6,028	52,250	56,825	-.007
3,000 lb.	16,662	402.2	18.6	101.0	5,876	51,821	56,292	-.009
2,000 lb.	15,662	402.4	18.7	101.2	5,739	51,407	55,759	-.009
1,000 lb.	14,662	402.5	18.8	101.4	5,571	51,015	55,273	-.009
EMPTY	13,662	402.7	18.9	101.8	5,373	50,756	54,932	-.008

Table 7-5 - SF-121 Balance and Inertial Data
Design Mission Stores Off, Gear Down

(AIM-9 Launchers and Pylons On)

FUEL STATE	WEIGHT	CENTER OF GRAVITY			MOMENT OF INERTIA SLUGS FT^2			"θ" RADS
		F.S.	% MAC	W.L.	ROLL	PITCH	YAW	
FULL	21,738	401.1	18.0	102.1	8,922	56,158	62,975	-.010
7,000 lb.	20,662	401.1	18.0	101.3	7,731	55,065	60,797	-.006
6,000 lb.	19,662	401.2	18.0	101.2	7,075	54,082	59,188	-.004
COMBAT 4,845 lb.	18,506	401.2	18.0	101.2	6,569	52,964	57,566	-.003
4,000 lb.	17,662	401.3	18.1	100.4	6,381	52,490	57,000	-.005
3,000 lb.	16,662	401.4	18.1	100.1	6,227	52,060	56,467	-.006
2,000 lb.	15,662	401.5	18.2	100.2	6,090	51,647	55,935	-.007
1,000 lb.	14,662	401.6	18.2	100.4	5,923	51,257	55,450	-.007
EMPTY	13,662	401.7	18.3	100.6	5,728	51,001	55,109	-.005

Table 7-6 - SF-121 Balance and Inertial Data Maximum Overload

DLI WEIGHT PLUS 10,000 LB. (GEAR DOWN)

WEIGHT	CENTER OF GRAVITY			MOMENT OF INERTIA ~ SLUGS FT^2		
	F.S.	% MAC	W.L.	ROLL	PITCH	YAW
33,375	399.1	17.1	92.7	20,106	65,862	81,123

DLI LOADING -	(2) AIM-7 AND LAUNCHERS	-1,160
+	(4) FUSELAGE RACKS	+ 100
+	(2) MK84 LGB	+4,160
+	(2) MK83LDB AND RETARDER	+2,020
+	(2) 300 GALLON TANKS	+4,880
	ΔW =	+10,000 LB.

Table 7-7 - Maximum VTO Weight (T/W = 1.0)

DLI MISSION AND EXTERNAL FUEL (GEAR DOWN)

WEIGHT	CENTER OF GRAVITY			MOMENT OF INERTIA SLUGS FT^2		
	F.S.	% MAC	W.L.	ROLL	PITCH	YAW
27,500	401.4	18.4	96.6	17,438	62,153	75,838

DLI LOADING + (2) 300 GALLON TANKS FILLED TO 245 GALLONS EACH

8.0 SF-121 PERFORMANCE

8.1 POINT DESIGN

Vought studies of the VATOL concept prior to the Phase I contract and the wing optimization described in Section 8.2.1 provided an excellent basis for synthesizing a point design. The study approach was to complete this task at an early date so that the major goals of a complete aerodynamic description and transition analysis could be performed in depth. Figure 8-1 shows schematically how the SF-121 point design was achieved.

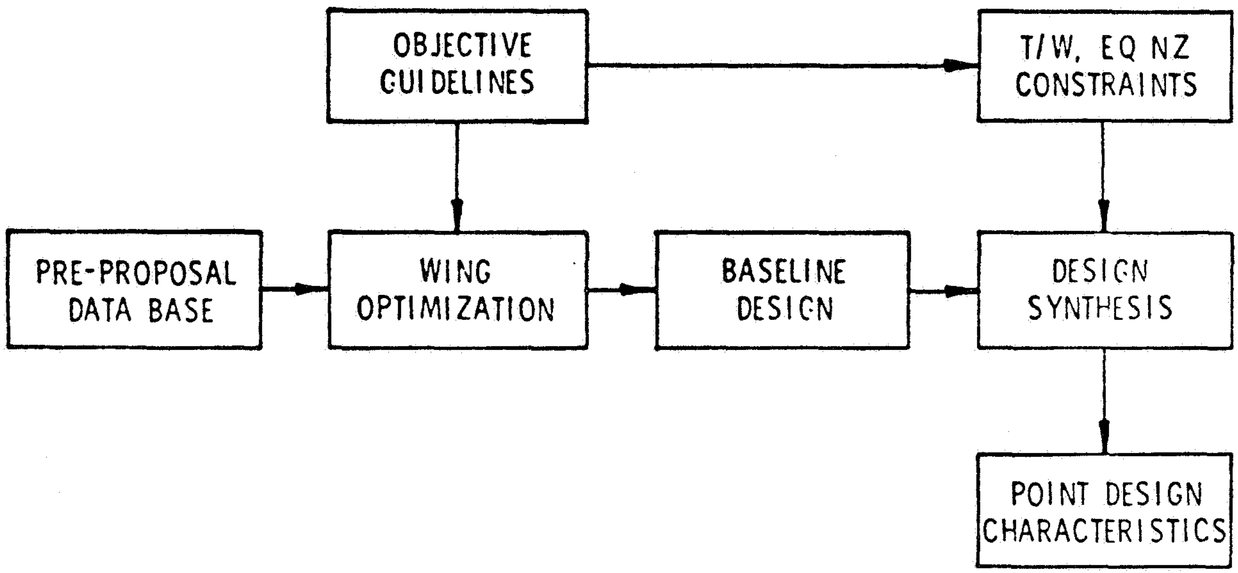


Figure 8-1 - Design Synthesis Procedure

8.1.1 Configuration Synthesis

VSTOL fighters can be uniquely defined by meeting three interacting but distinct conditions, typically:

- o Design mission radius → Internal fuel capacity
- o Maneuver load factor → Wing area
- o Hover thrust to weight → Engine size

The mission which best exploits the inherent capabilities of VATOL is the

Supersonic Intercept (Deck Launched Intercept). Dispersed basing has the effect of reducing required radius of action and dash Mach number. For this study the Supersonic Intercept design mission profile was specified to have a 150 NM radius with a Mach 1.6 40,000 foot outbound dash. These values have been used in other VSTOL studies in recent years and will facilitate comparisons with other concepts.

The objective guidelines in the Request for Proposal included a sustained 6.2 g maneuver constraint at Mach 0.6 10,000 feet. Vought selected a thrust to weight of 1.03 for single engine hover as the third sizing condition. It was imposed at a weight corresponding to 1,000 pounds fuel, 400 rounds of 20 mm ammunition, but no external stores. Thrust rating was maximum afterburner, 89.6°F, with minimum cooling air bleed but no contingency rating. The 1.03 value was recommended by the Navy for Type A VSTOL studies. Later analysis addressed the suitability of this criterion.

The SF-121 point design which meets the three sizing criteria was determined using the Vought Aircraft Synthesis Analysis Program (ASAP), Figure 8-2 ASAP interfaces the technical disciplines (weights, propulsion, aerodynamics and performance) and creates a design space for a specified matrix of configuration variables. The CDC 6600 interactive computer graphics facility displays the results. The minimum weight airplane which satisfies all missions and constraints within the design space can then be selected by the designer and machine plotted. Figure 8-3 shows the weight carpet for the SF-121. All nine combinations of wing area and engine scale factor are fuel balanced to a 150 NM radius on the design mission. It is seen that only designs with a wing area of 350 square feet or greater satisfy the 6.2 g maneuver constraint. The selected point design is defined by the intersection of the maneuver and thrust/weight = 1.03 boundaries, yielding:

- o Takeoff gross weight = 23,375 lb. (10,603 kg)
- o Wing reference area = 350 ft^2 (32.56 m^2)
- o Engine scale = 1.131 Ratio thrust per engine = 16,965 lbt (75,464 N)

The capabilities of this point design will be explored in the following sections.

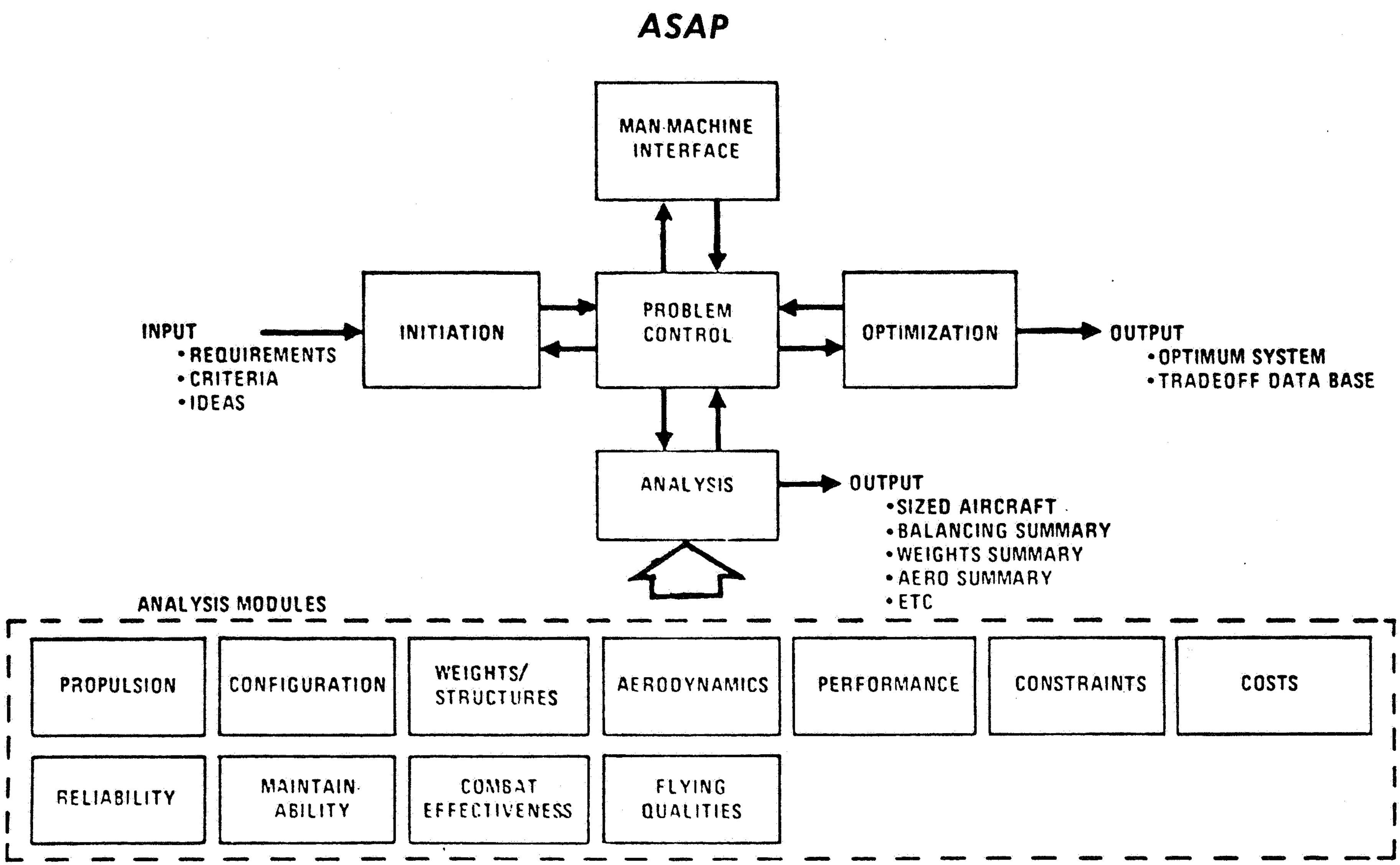


Figure 8-2 - Aircraft Synthesis Analysis Program [ASAP] Architecture

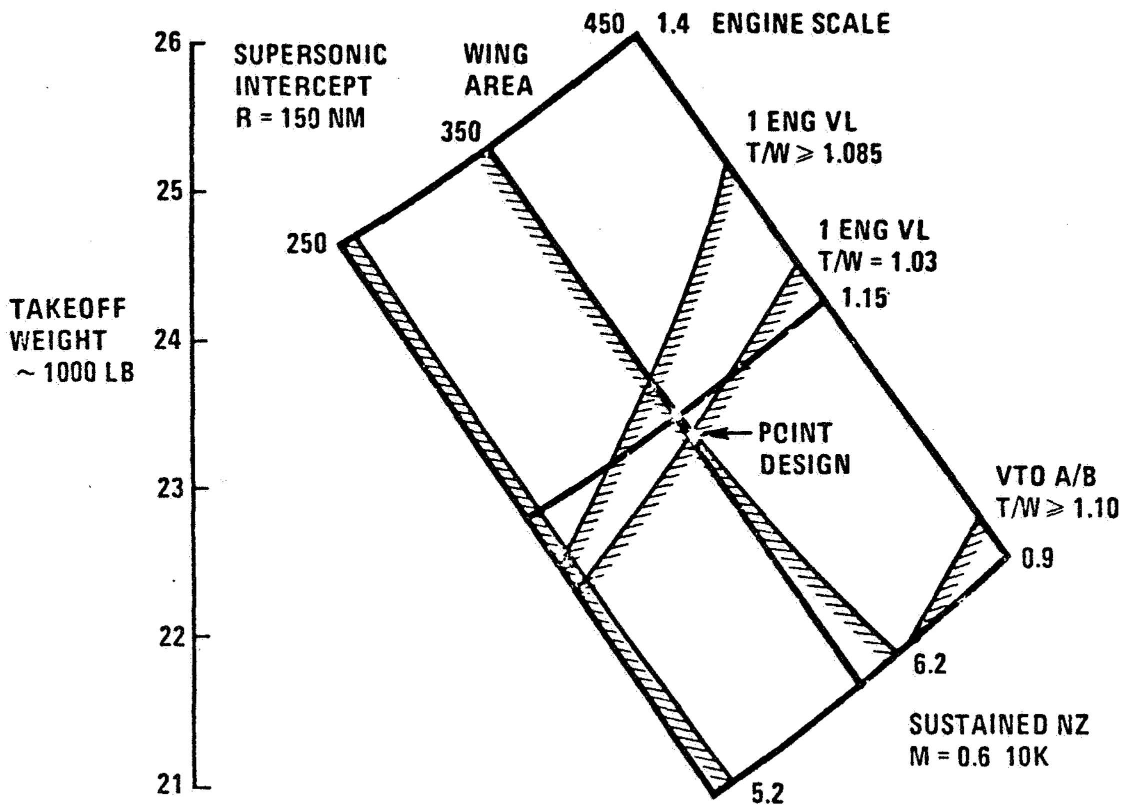


Figure 8-3 - SF-121 Parametric Sizing Carpet

8.1.2 Mission Capability

The SF-121 was evaluated on a total of five missions. Figure 8-4 diagrams the design mission profile as well as alternate Fighter Escort and Combat Air Patrol (CAP) missions. Two strike missions, Surface Strike and Interdiction, are described in Figure 8-5. Internal fuel capacity is set by the Supersonic Intercept design mission. Two 300 gallon external fuel tasks are carried on all the alternates except Fighter Escort. Tanks are dropped when empty. No specific radius of action or time on station goals were set for the alternate missions.

Mission performance is summarized in Table 8-1. The efficiency of the SF-121 on the design mission is reflected in the rather small fuel capacity of 8,077 pounds. This is consistent with the operating philosophy of dispersed or forward basing, which tends to reduce range requirements. The SF-121 responds well to external fuel, as indicated by the last three missions of Table 8-1. Note that the high takeoff thrust/weight on the fighter missions could enable higher internal fuel loads than assumed with minimal effect on the airplane itself.

Table 8-1 - SF-121 Mission Performance

Supersonic Intercept (Design Mission) M = 1.6, 40K	VTO W = 23,375 lb T/W = 1.29	Radius = 150 NM
Fighter Escort	VTO W = 23,375 lb T/W = 1.29	Radius = 278 NM
Combat Air Patrol R = 150 NM	STO W = 28,255 lb T/W = 1.07	TOS = 2.25 hr
Surface Strike R = 300 NM	STO W = 29,549 lb T/W = 1.02	TOS = 1.89 hr
Interdiction 50 NM SL Dash	STO W = 31,135 lb T.W = 0.97	Radius - 528 NM

8.1.3 Combat Performance

The operational envelope for the SF-121 at combat weight is shown in Figure 8-6. The placard limits are a dynamic pressure of 2,133 psf below 34,000 feet and a Mach 2.4 aerodynamic heating limit above 34,000 feet. The large engines overcome the rapidly decaying pressure recovery of the fixed three-shock inlets out to Mach 2.57. The ability to fly supersonically without afterburner indicates the low degree of augmentation required on the Supersonic Intercept mission.

Figures 8-7, 8-8, and 8-9 map specific excess power versus sustained load factor, at 10, 20, and 30,000 feet, respectively. Structural design load factor is 7.5 g with 60 percent internal fuel (7.35 g at 0.88 VTO weight).

Table 8-2 lists other combat performance capabilities. The inherently high energy maneuverability of the SF-122 is apparent.

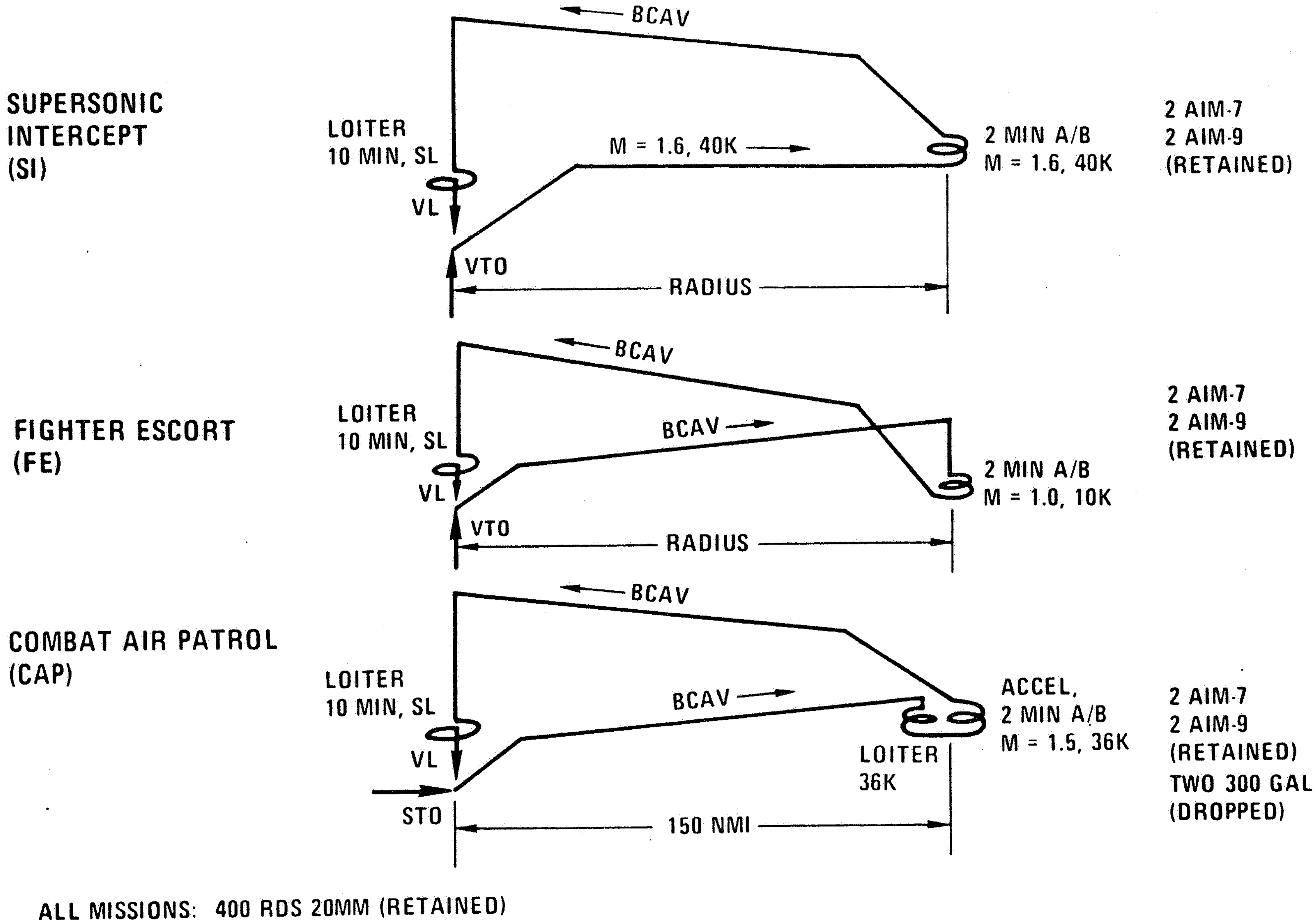


Figure 8-4 - SF-121 Notional Fighter Missions

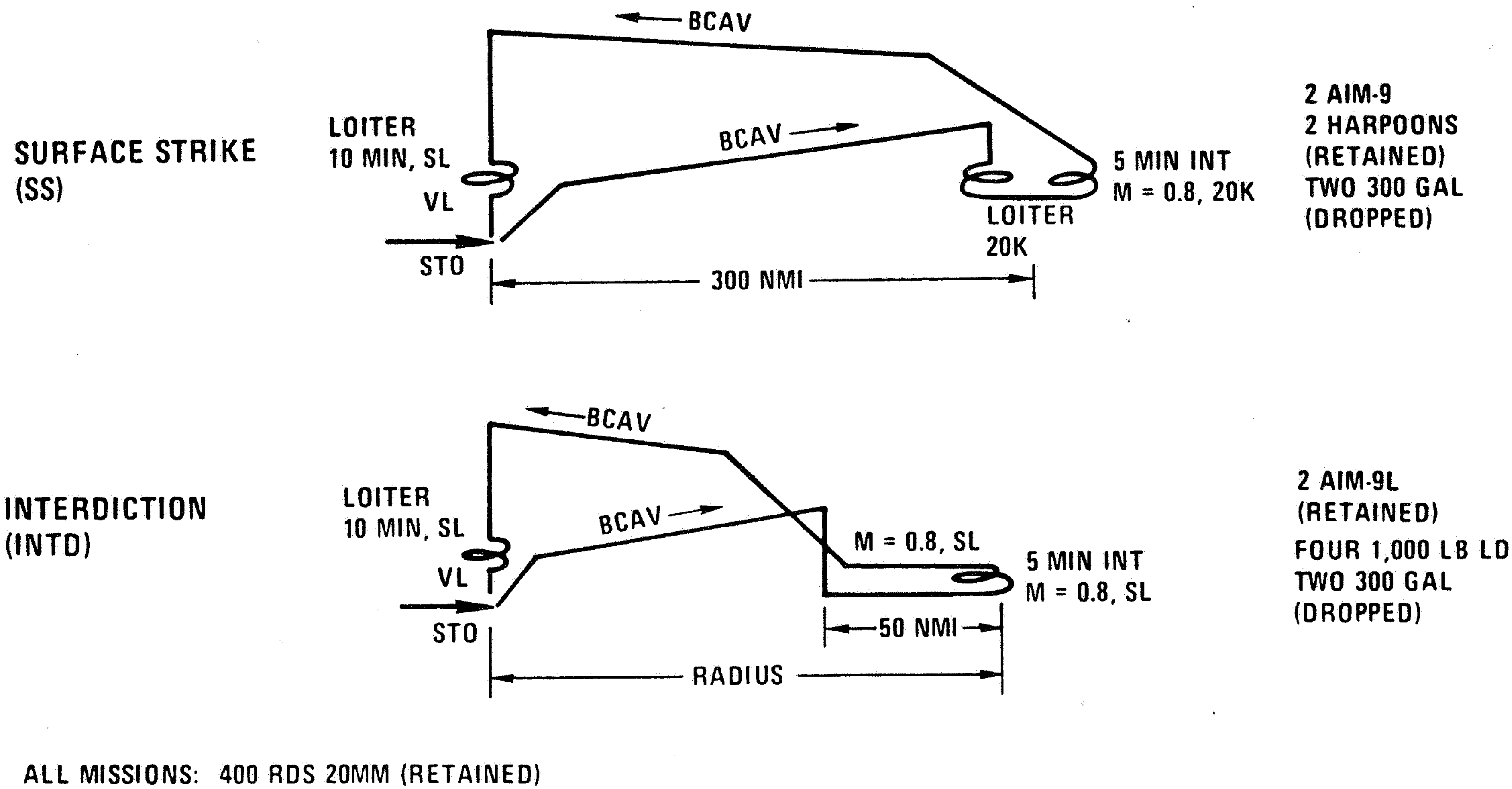


Figure 8-5 - SF-122 Notional Strike Missions

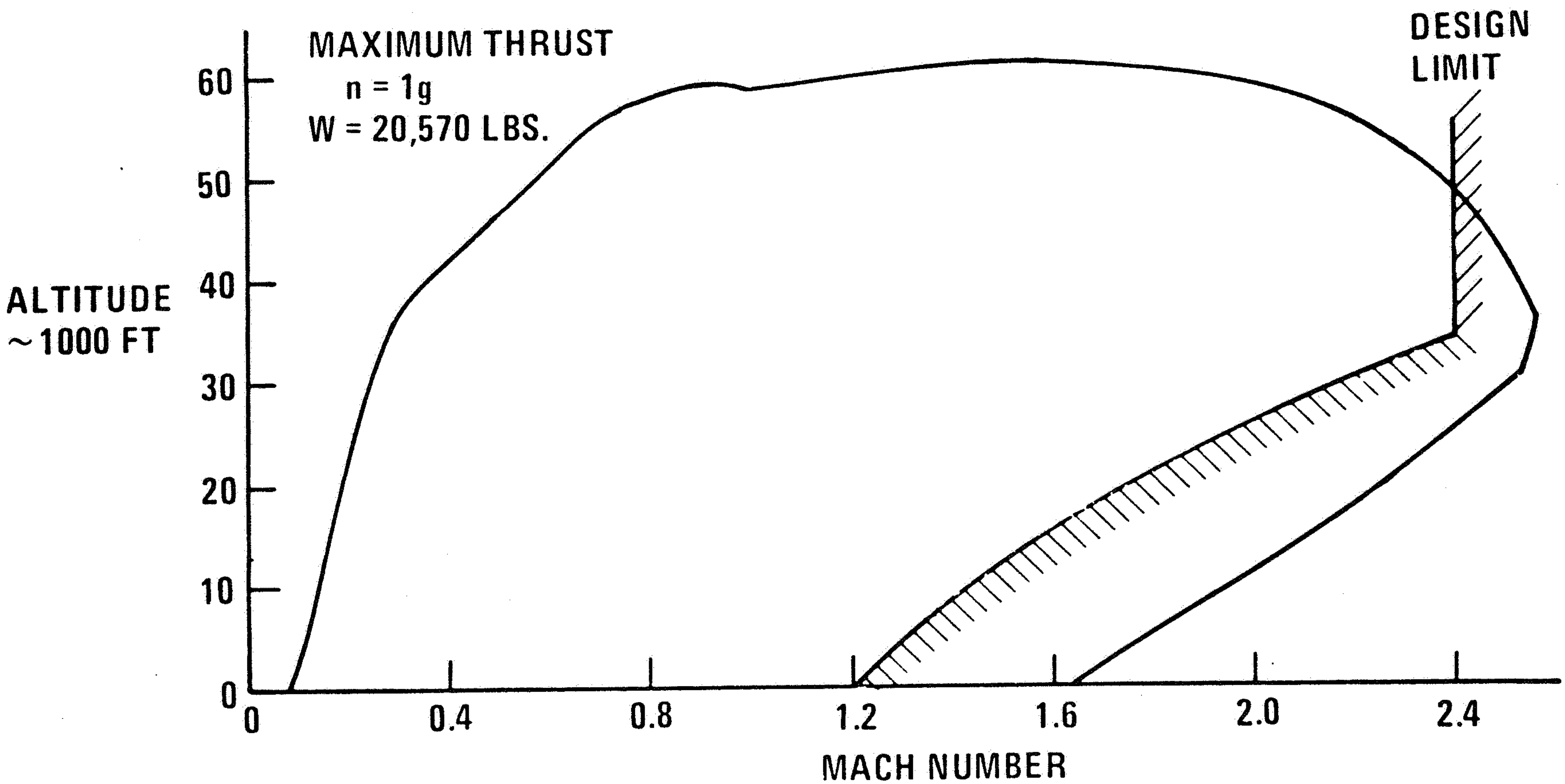


Figure 8-6 - SF-121 Flight Envelope

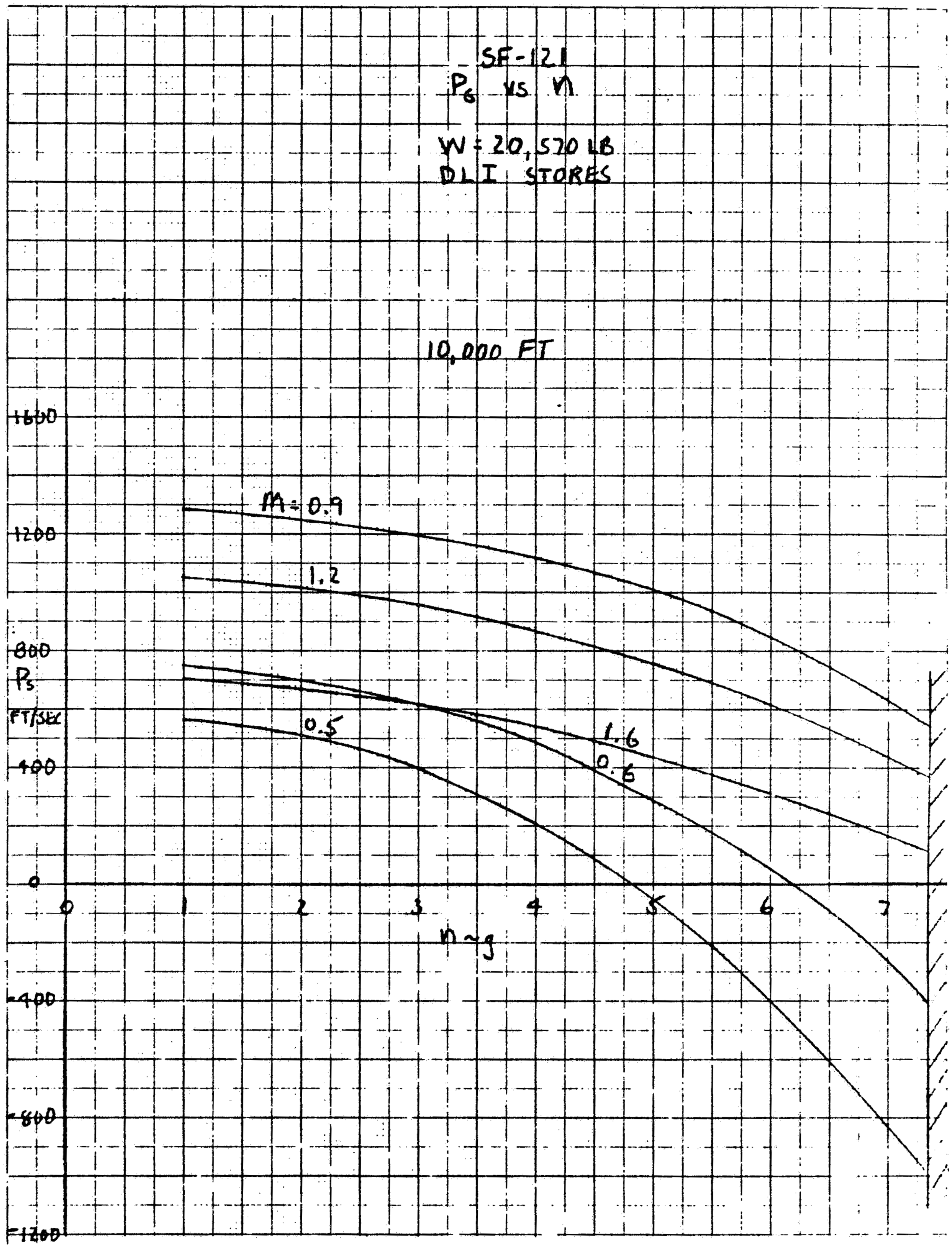


Figure 8-7 - SF-121 Energy Maneuverability - 10,000 ft

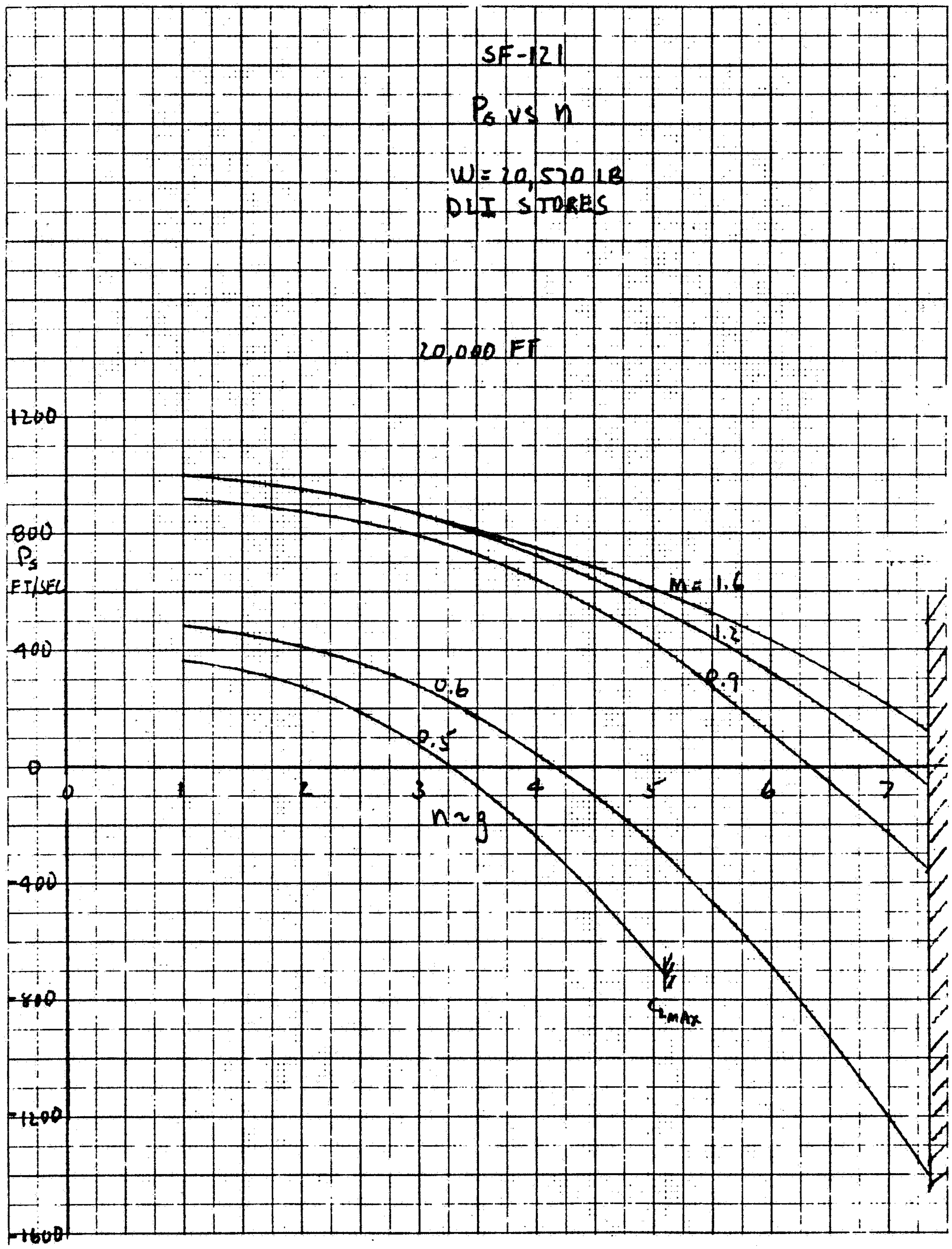


Figure 8-6 - SF-121 Energy Maneuverability - 20,000 ft

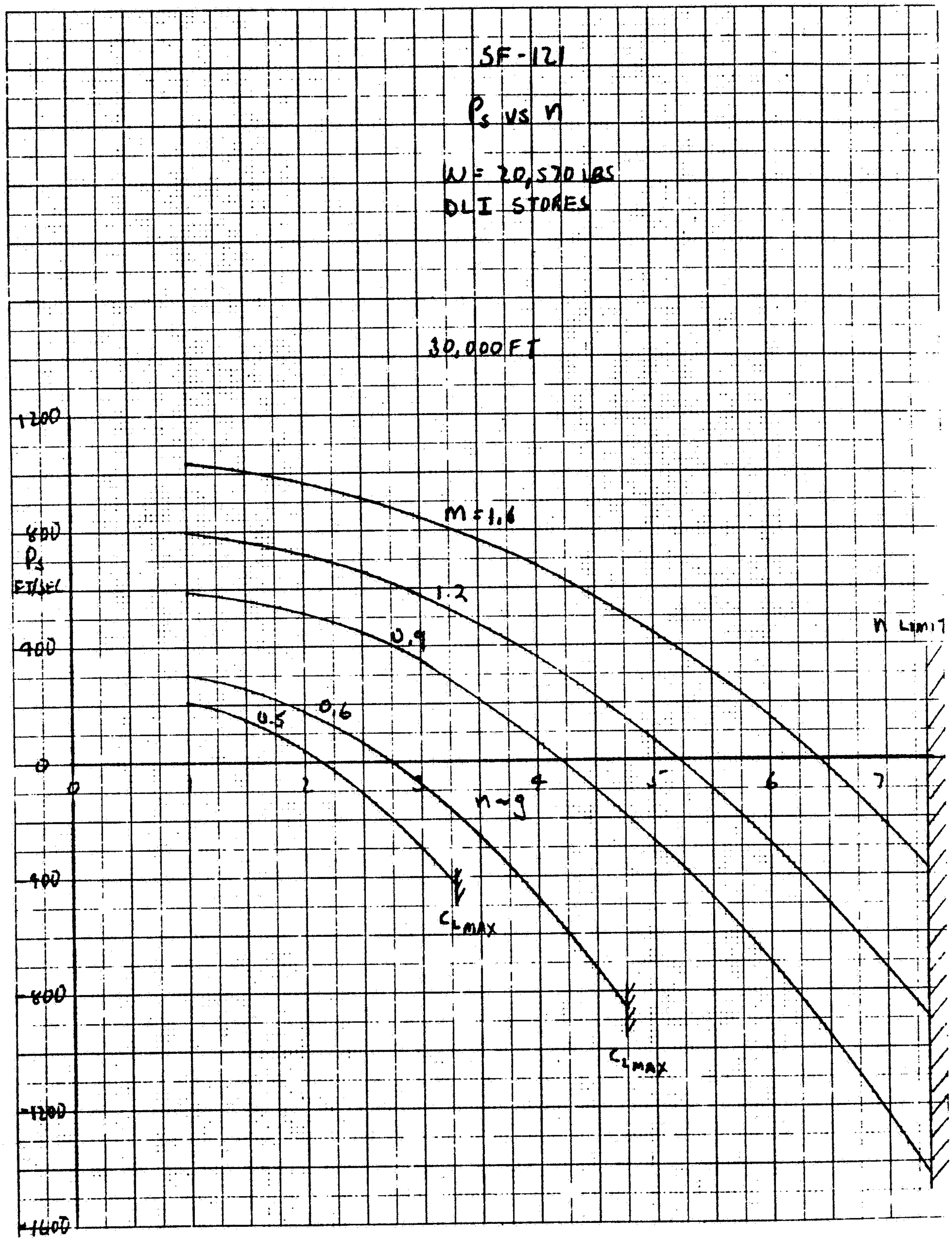


Figure 8-9 - SF-121 Energy Maneuverability - 30,000 ft

Table 8-2 - SF-121 Combat Performance

PARAMETER	VALUE
Maximum Mach Number	2.57 (Limit 2.4)
Max A/B, 36K	2.57 (Limit 2.4)
Intermediate, 36K	1.20
Intermediate, SL	1.00
Combat Ceiling - Max A/B	60,650 feet
Combat Ceiling - Intermediate	51,500 feet
Acceleration Time, Mach 0.8 → 1.6, 36K	44.6 seconds
Specific Excess Power, M = 0.9, 10K	1,286 feet/sec
Equilibrium Load Factor	
Mach 0.6, 10K	6.20 g
Mach 0.9, 30K	4.84 g
Mach 1.6, 40K	4.02 g
Equilibrium Turn Rate	
Mach 0.6, 10K	17.42 deg/sec.
Mach 0.9, 30K	9.75 deg/sec.
Mach 1.6, 40K	4.63 deg/sec.

8.2 SENSITIVITIES

8.2.1 Wing Optimization

Definition of the SF-121 itself was preceded by a wing planform study performed on the SF-120 proposal configuration. The purpose was to ensure that the planform chosen for in-depth aerodynamic analysis was compatible with good mission performance. Three wing/canard variations were sized using ASAP to the performance guidelines discussed in Section 8.1.1 Figure 8-10 compares the resulting planforms. As aspect ratio is increased and leading edge sweep simultaneously decreased, required wing area becomes less. However, the greater span of the increased aspect ratio overrides the weight saving from lower area. The resulting weight comparison is presented

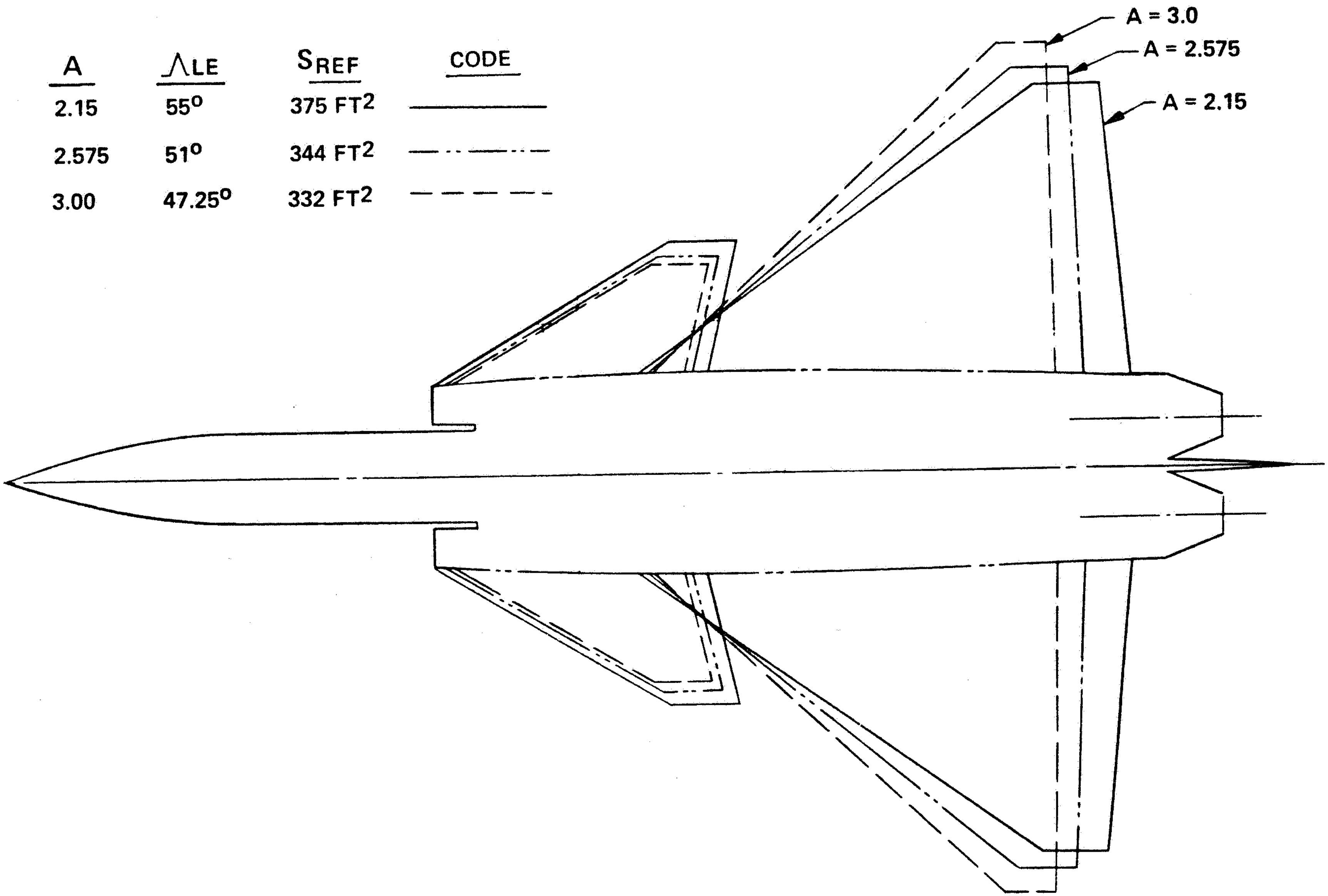


Figure 8-10 - SF-120 Planform Variants

in Figure 8-11. It is seen that the optimal aspect ratio is about 2.3, and that "high" values such as 3.0 yield higher takeoff weight.

This is a significant result. CTOL fighters usually benefit from aspect ratios as high as 4.0. But the premium placed on low empty (or landing) weight makes a light wing more valuable than one with lower drag due to lift. On the Supersonic Intercept mission the aspect ratio 2.3 wing also has low supersonic drag, which reinforces its superiority.

Previous wing studies have shown taper ratio to be a second order parameter. The original value of 0.1 was increased to 0.15 for the SF-121 to increase outer panel elevon chord and provide more space for hover reaction jet roll controls. Another study showed a wing thickness ratio of either 5 or 6 percent to give equal performance. The thinner wing was chosen to permit supersonic dash at slightly lower augmentation to reduce infrared signature.

8.2.2 Constraint Variations

The standard ASAP synthesis procedure yields a wealth of performance sensitivity data relating the primary design variables and constraints. Appendix D contains this backup data for the SF-121 and explains how to use it to determine the weight and performance consequences of alternative sizing criteria.

8.3 TRANSITION PERFORMANCE

Transitions from hover to conventional flight (conversions) and conventional to hover flight (reconversions) have been simulated for the SF-121 point design. Variables evaluated include weight, flight path angle, deceleration rate and aircraft static margin. Time histories show: horizontal and vertical position and angle; aircraft angles of attack and pitch; aerodynamic forces and moment; thrust required; and trim thrust deflection. Conversion time to Mach 0.3 was rapid, but refinements are needed for flight path control. Reconversion time and thrust required evidenced much variation due to technique and static margin. Single engine reconversions were possible only over a very narrow band of operating conditions determined by thrust available. A thin aerodynamic data base precluded evaluation of configuration effects which could reduce time in transition.

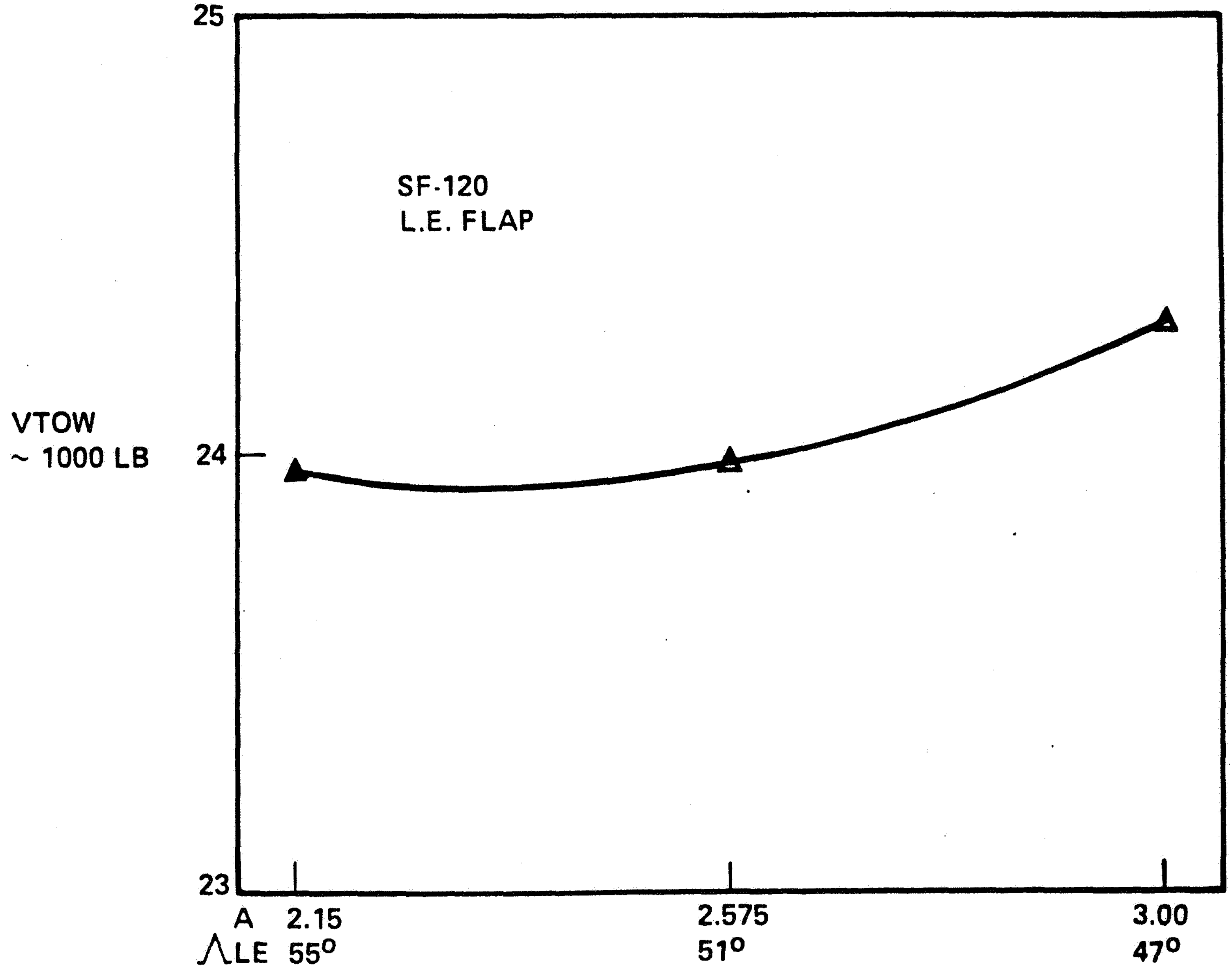


Figure 8-11 - Planform Study Results

The analysis clearly showed that the engine sizing criterion of thrust/weight = 1.03 for the single engine vertical landing made transitions to hover difficult and marginal. Either a larger engine or a short term rating giving a $T/W \geq 1.086$ is necessary to meet MIL-F-83300 Level 2 control powers. Most of the problems discussed in this section apply to the T/W = 1.03 sizing constraint and can be alleviated by increased thrust available.

Maximum available control power was determined for two types of attitude control systems. The basic system included thrust vectoring control for pitch and yaw and reaction jet roll control. An alternate system using reaction control about all axes was compared to the basic system. Results showed the basic system to be distinctly superior. Maximum control power and control sensitivity compared favorably to MIL-F-83300 and AGARD 577 requirements. Revised design thrust to weight margins have been postulated as a result of these studies. Neither system studied provided enough control power to trim out 15 degrees sideslip at $\alpha > 26$ degrees in transition. This was due to highly unstable directional stability estimated for the basic SF-121 configuration. Configuration development testing to reduce or eliminate this problem is indicated.

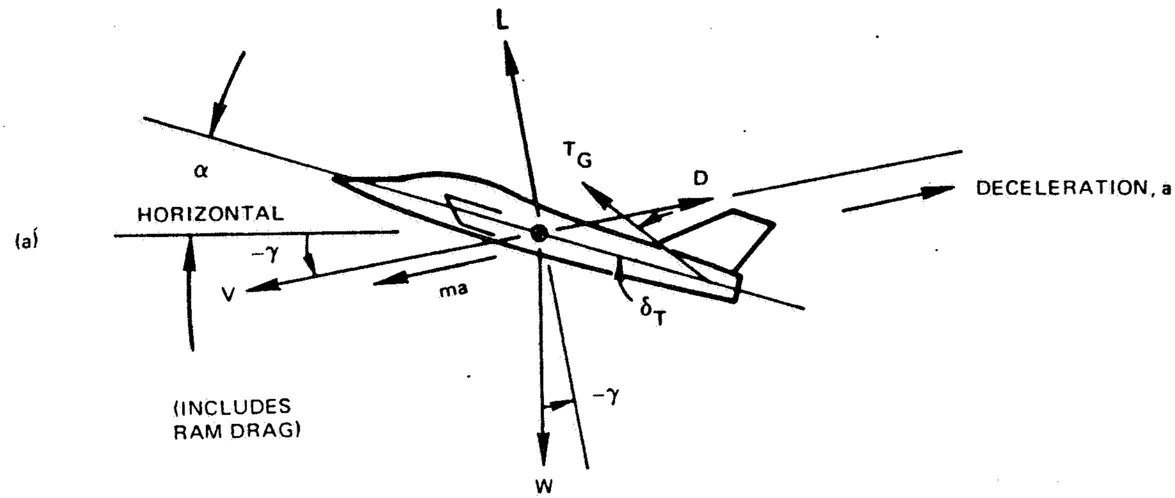
Transition roll/yaw control phasing of roll reaction jet thrust and yaw control thrust vector deflection with aircraft pitch attitude was evaluated. Opposite axis coupling was negated for this study. The phasing schedule was developed to keep the pilot's conventional flight cockpit controls inputs and orientation with the horizon compatible through transition into hover. Thus, pilot workload and training time would be reduced. In conventional flight rudder provides yaw and stick provides roll. During vertical attitude hover stick provides yaw and rudder provides roll. To determine the transition phasing, intermediate inertias were computed, control requirements were established, and required control input phasing was calculated. Results indicated that roll and yaw coupling was favorable at intermediate pitch angles (i.e., nose right yaw control induced nose right yaw). Proper blending of these controls could significantly reduce the maximum single axis control power required (e.g., reaction jet thrust and/or thrust vector deflection).

Further study is needed to set minimum requirements for control power and consequent design thrust to weight levels. Thrust vectoring appeared attractive for two engine operation roll control but a scheme for single-engine roll control sans reaction jet was not readily apparent. Proper assessment of the emergency landing is critical to assure a satisfactory design solution.

8.3.1 Landing Transition

The objective of the transition analyses was to evaluate VATOL reconversion and conversion flight paths. Selected flight paths required adequate height and attitude control with minimum impact on engine size, VTO gross weight and pilot workload. Reconversion was much more difficult because of throttle excursions combined with pitch and sink rate control during deceleration and descent. Profiles were based on X-13 Vertijet flight test experience (References 12 and 13). Relatively straightforward profiles including a constant pitch rate level deceleration to stall, a higher constant pitch rate level deceleration to near vertical attitude, and a descending or level deceleration to an intercept were developed. Control variables were pitch rate and rate of sink as a function of velocity. Reconversions were assessed for the basic unstable and stable two engine landings and the unstable single-engine emergency landing. Normal two-engine reconversions from approach speed (1.3 V_{SPA}) to a landing intercept (5 ft/sec forward velocity and 3 ft/sec sink rate) were achieved in 37.6 to 46.8 seconds. Similar results were shown for level decelerations ending at 5 ft/sec forward velocity. Thrust required and attitude calculated for the reconversions were used as a basis for the control power and phasing studies discussed in Sections 8.3.3 and 8.3.4.

All reconversions were calculated on a quasi-steady (i.e., no Z axis acceleration forces) basis using conventional longitudinal three degree of freedom equations of motion (see Figure 8-12). To effect solution, an initial angle of attack, velocity, pitch rate, and rate of sink profile and maximum pitch angle were specified. Vertical force and moment balance was required for each point (time interval) calculated. Net deficiency in horizontal force was output as a deceleration and was integrated to give velocity and position along the flight path. Vertical position was integrated from rate of sink. The force and moment balance resulted in thrust required and thrust deflection needed to trim. Thus, thrust deflection required to



Σ

$$\Sigma M_{CG} = 0 = M_{AERO} + F_R \ell_R \sin\alpha + F_G \ell_{F_G} \sin\delta_{F_G}$$

$$\Sigma F_X = 0 = D_{AERO} - F_R + F_G \cos(\alpha - \delta_{F_G}) - W \sin\delta + m\, a_X$$

$$\Sigma F_Z = 0 = L_{AERO} + F_G \sin(\alpha - \delta_{F_G}) - W \cos\gamma + m\, a_z$$

where: $M_{AERO} = C_M\, q\, S\bar{C}$

$L_{AERO} = C_L\, q\, S$

$D_{AERO} = C_D\, q\, S$

Figure 8-12 - Longitudinal Equations of Motion maintain or establish pitch rate was not included in the results. Hand calculations indicated that a 3 degree thrust deflection would be required to initiate the maximum 5 deg/sec pitch rate used for these calculations. Much less would be required to overcome aerodynamic damping, which was not estimated.

Aerodynamic data used for transition analyses is presented in Figures 5-13, 5-14 and 5-15. The derivation of this data appears in Appendix A. For the transition analyses, aerodynamic elevon trim was assumed to be available up to +20 degrees elevon deflection. Maximum elevon was limited to $\pm$ 20 degrees until the end of the transition to assure an adequate margin of elevon deflection for roll control. Landing gear drag ($\Delta C_{D_{GEAR}} = 0.0200$) and store drag ($\Delta C_{D_{STORES}} = 0.0024$) were added to the trimmed drag. Store drag was deleted for the single engine vertical landing.

Reconversions were patterned after X-13 Vertijet flight test results (Reference 13). Reconversion was initiated at 1.3 V_{SPA} ($\alpha \geq 16.4$ degrees) with a slow pitch rate of 0.8 - 1.2 deg/sec. Vertical force balance required to maintain level flight limited pitch rate to that which would yield thrust reductions to idle. Average deceleration to stall was 2 to 3 kts/sec. At stall, a pitch rate of 5 deg/sec was commanded and held until a specified maximum pitch angle (Θ) was reached. Pitch rates less than five degrees per second caused significant increases in deceleration time. Sustained pitch rates exceeding 5 deg/sec would reduce transition time but could be quite uncomfortable for the pilot. All descending profiles initiated sink rate at 80 ft/sec forward velocity. Maximum sink rate was 12 ft/sec for two-engine descents and 3 ft/sec for one-engine descents. A summary of reconversion performance is presented in Table 8-3. Thrust available for reconversion and conversion is tabulated in Table 8-4. Reconversion time histories for the SF-121 point design are presented in Figure 8-13 (normal two engine), and in Figure 8-14 for the single engine case.

Two engine reconversions were generally uncomplicated. Time and distance to landing intercept was reduced by increasing the maximum allowable angle of attack beyond 90 degrees. This maneuver increased braking thrust required near the end of transition. The higher braking thrust required still left ample margin attitude and height control (See Table 8-3). Fuel

Table 8.3-1

SF-121

Reconversion Performance Summary

LANDING CONFIGURATION (1)	$\dot{\alpha}$ (2) (DEG/SEC)	Θ_{MAX} (DEG)	R/S_{MAX} (3) (FT/SEC)	MINIMUM EXCESS T/W (4)	TIME (5) (SEC)	FUEL USED (LB) (5)	HORIZONTAL DISTANCE ΔX (FT) (5)	VERTICAL DISTANCE ΔZ (FT) (5)
BASIC DESIGN W = 16,375 LB TWO ENGINES	0.8/5	95	12	0.77	45.9	257	5,000	-134
	0.8/5	100	12	0.63	42.1	214	4,910	- 96
	0.8/5	95	0	0.81	46.8	265	4,990	0
	0.8/5	100	0	0.79	41.6	197	4,854	0
STABLE DESIGN W = 16,375 LB TWO ENGINES	1.0/5	95	12	0.82	41.2	323	4,238	-163
BASIC, STORES OFF W = 14,622 LB ONE ENGINE	1.2/5	92.5	3	0.004	45.2	325	3,770	-50
	1.2/5	95	0	0.004	38.8	253	3,615	0

(1) INCLUDES 1,000 LB. FUEL + STORES UNLESS NOTED
(2) PITCH RATE: PRE-/POST- STALL
(3) INITIATED AT 80 FT/SEC FORWARD VELOCITY

(4) $\left(\frac{F_g \text{ avail.} - F_g \text{ req'd.}}{W}\right)_{MIN}$ Ejector Bleed Off

(5) From 1.3 $V_{S_{PA}}$ to 3 ft/sec R/S and 5 feet/sec forward velocity

Table 8.3-2

SF-121

Conversion/Reconversions

Gross Thrust

Sea Level - Tropical Day
MFTF-2800-25-1 Engine
(1.131 Size Factor)

M	V~ (FT/SEC.)	IDLE	GROSS THRUST (LBS)			
			MAXIMUM INTERMEDIATE	MINIMUM A/B	MAXIMUM A/B	MAXIMUM A/B*
0	0	566	8,369	8,641	13,866	15,009
.05	57.3	622	8,460	8,709	14,002	15,042
.10	114.6	679	8,562	8,810	14,149	14,985
.15	171.9	792	8,674	8,935	14,364	14,985
.20	229.2	905	8,800	9,048	14,590	14,985
.30	343.8	1,188	9,104	9,387	15,099	15,099

* ECS BLEED ONLY

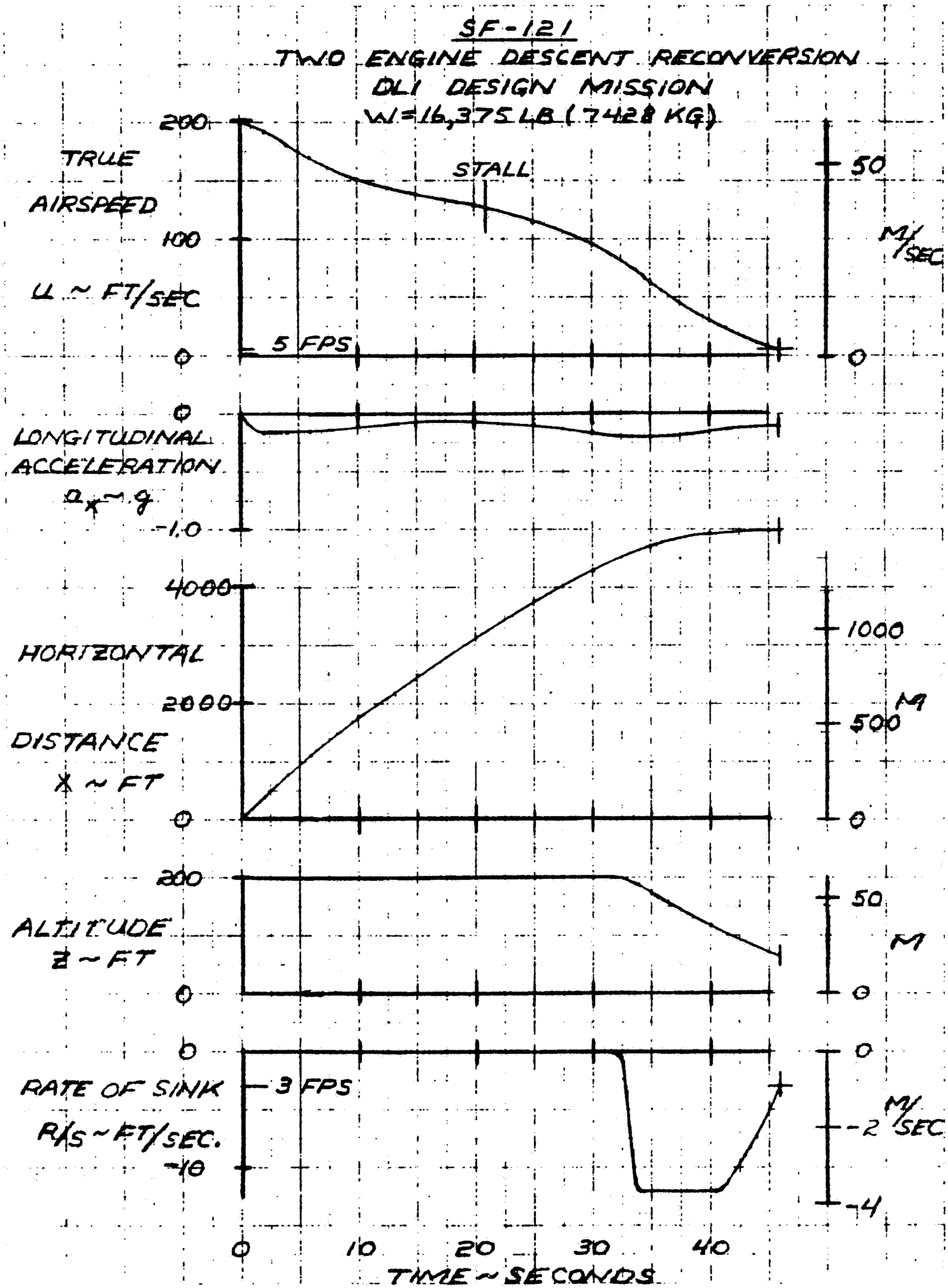


Figure 8-13a - Two Engine Descent Reconversion

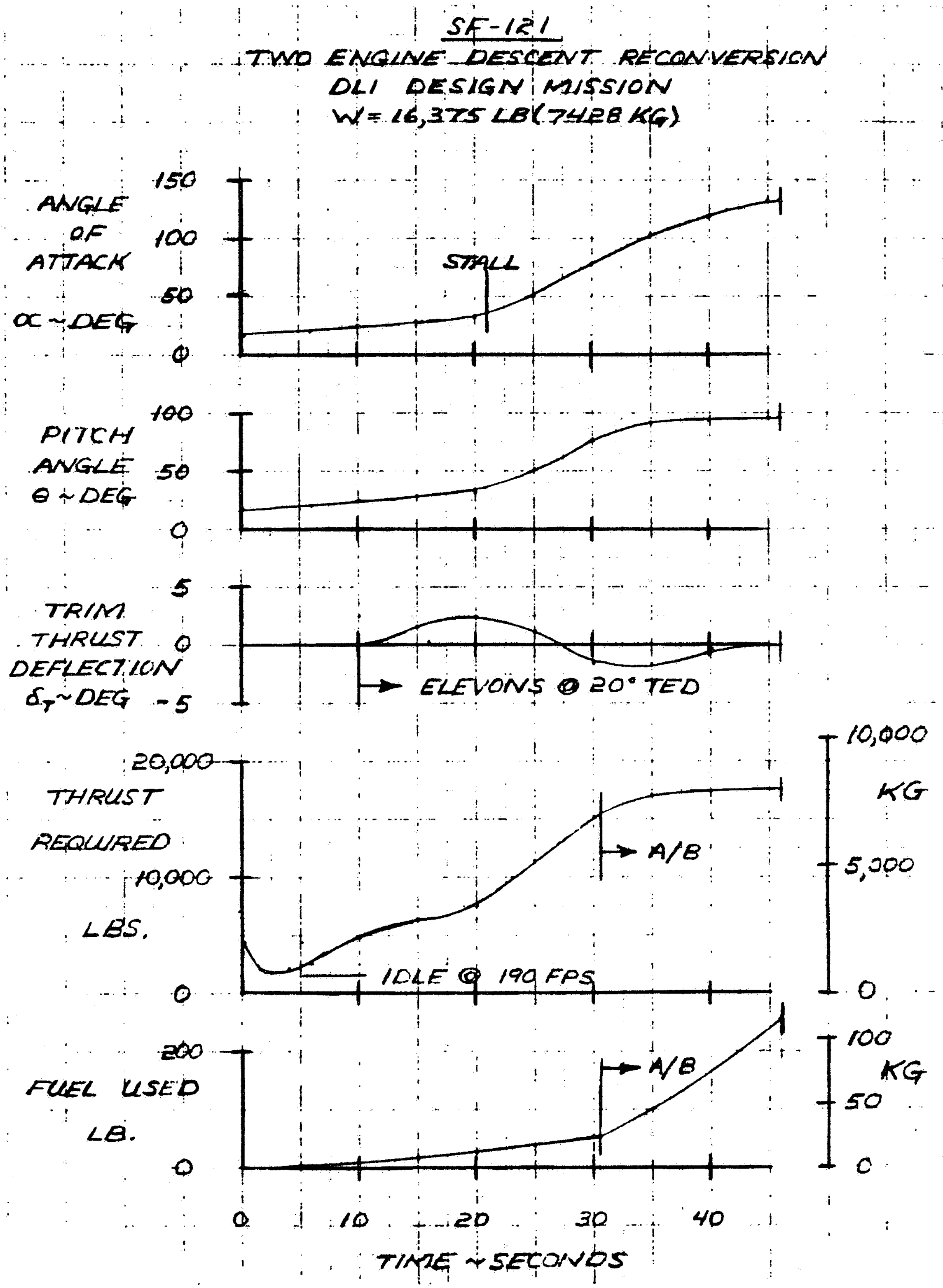


Figure 8-13b - Two Engine Descent Reconversion

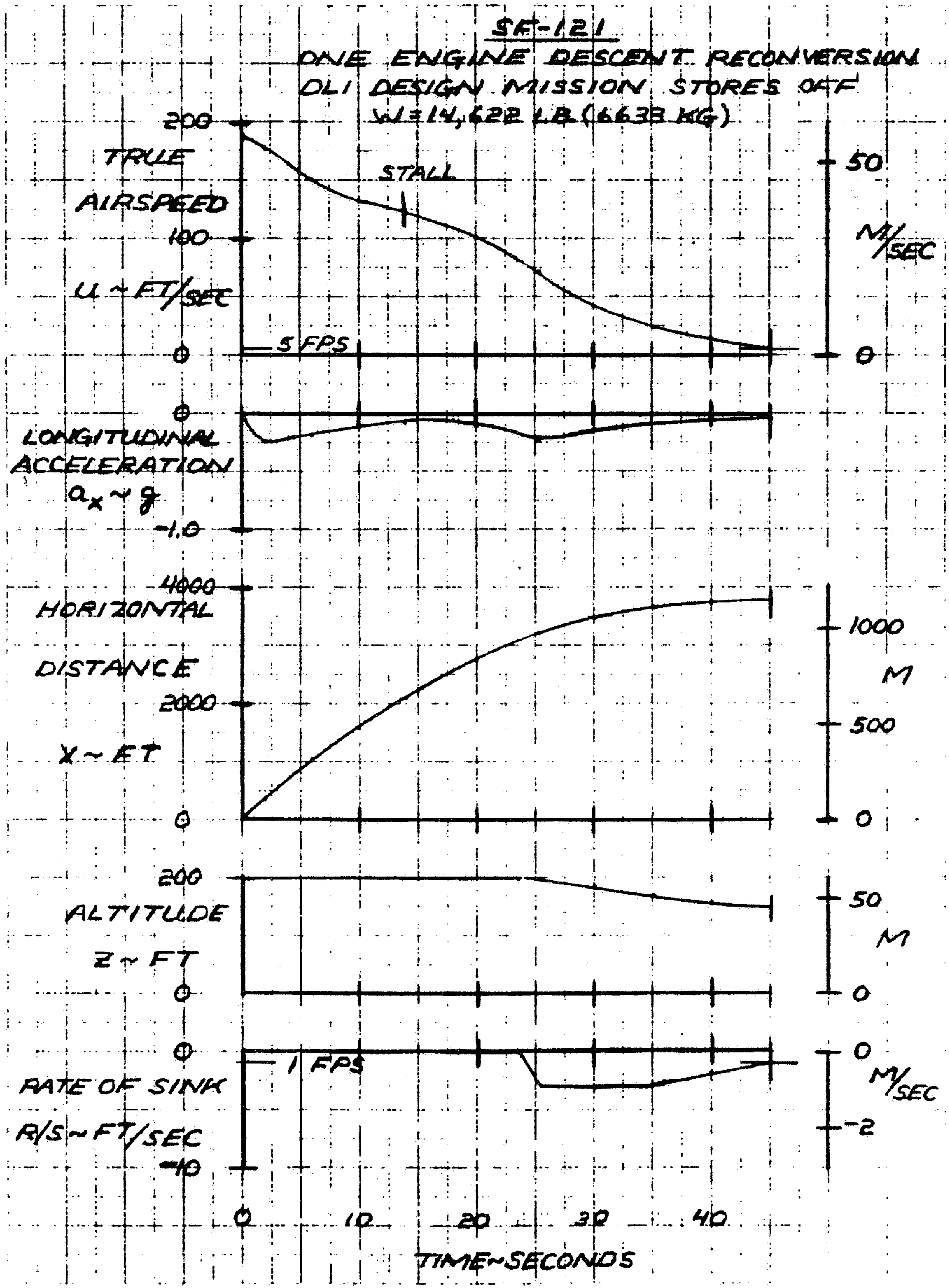


Figure 8-14a - One Engine Descent Reconversion

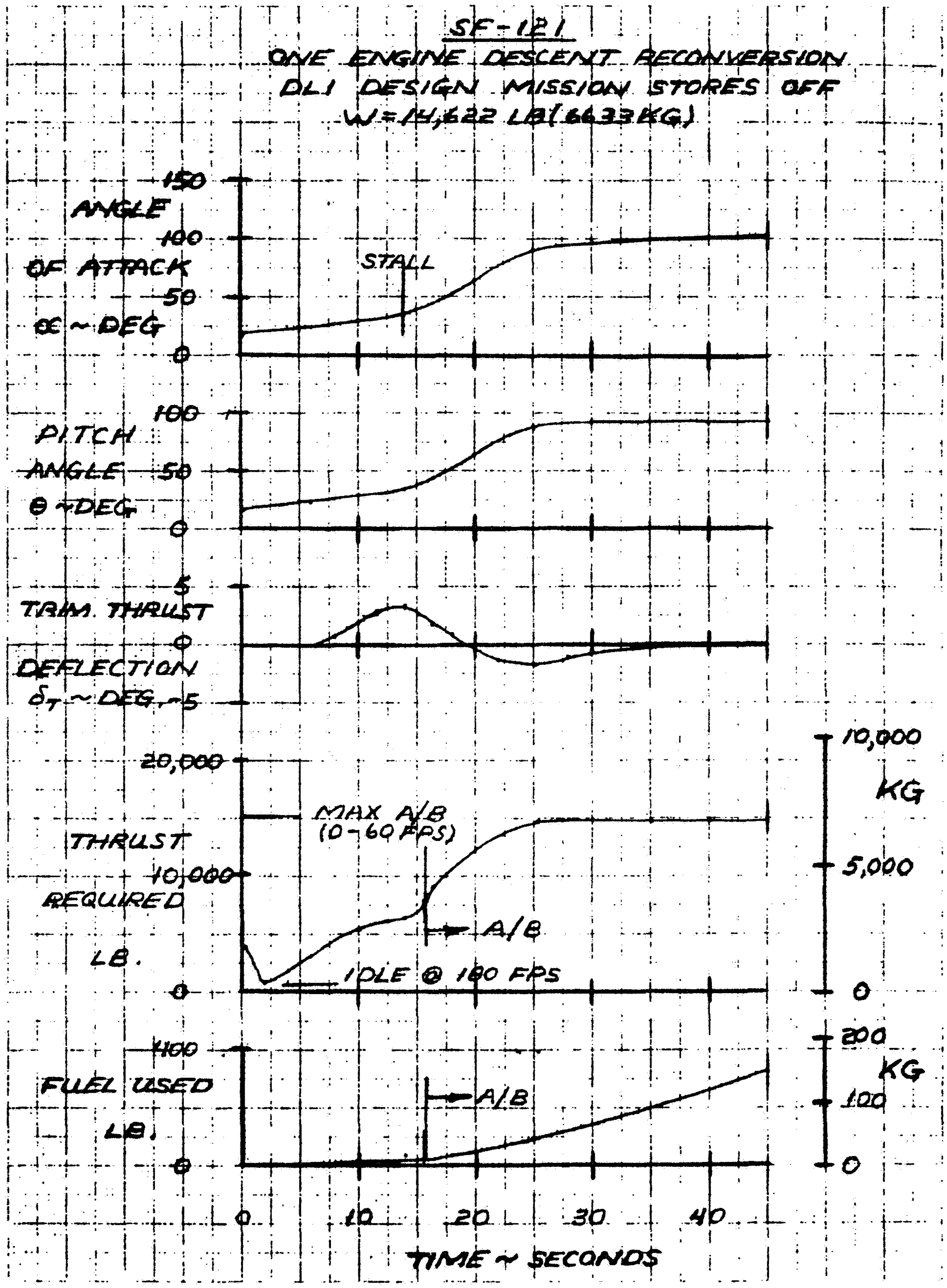


Figure 8-14b - One Engine Descent Reconversion consumption decreased as expected when transitions were performed rapidly. However, fuel flow was essentially the same whether transition was descending or level. Vertical distance in transition was also decreased with the higher maximum pitch angle because of increased thrust required for braking. Operationally, a descending transition with a maximum 95 degree pitch angle appears preferable because of the smaller thrust excursions. Descent to landing would be necessary after a level deceleration with a consequent increase in fuel required for landing.

A stable aircraft (+3 percent vs. basic -9.5 percent static margin) was also evaluted. Several significant differences were noted in the transition time histories. Reconversion was more rapid for the stable aircraft despite a higher initial speed. Lift, drag and moment characteristics are shown in Figures 5-1 thru 5-3. These indicate a lower C_L for 1.3 V_{SPA}, lower $C_{L_{MAX}}$, and a considerable increase in pitching moment past maximum lift. Deceleration time to maximum lift was less for the stable aircraft due to the smaller angle of attack range to be covered and the higher pitch rate required to achieve near-idle thrust. Consequently, high post-stall drag was reached quicker and at a higher speed. Therefore, the post-stall deceleration/descent was more rapid for the stable aircraft. Trim thrust deflection required exceeded the 15 degrees maximum throw available. This result is predicated upon accurate assessment of post-stall aerodynamics for the SF-121. Wind tunnel test results are critical to assure acceptable pitching moments in the post-stall regime, particularly if stable or near-stable configurations are desired.

Using degraded maximum lift to reduce transition time and distance is possible with both the unstable and stable aircraft. The significant factors are buffet intensity at the higher stall speed and thrust vector pitch control required both pre- and post-stall. The question of how much maximum lift is actually desirable warrants further study. The wing and canard variable camber schedules were optimized for cruise and maximum sustained load factor. The adaptive fly-by-wire control system could easily be programmed to tailor the transition aerodynamics.

Single engine reconversions revealed several control power limitations (See Figure 8-14). Maximum pitch angle was limited by thrust available for deceleration and/or descent. The minimum excess T/W (See Table 8-3) clearly left no margin for attitude or height control. The SF-121 was sized at 1.03 T/W for hover which was found to be inadequate. Discussions in Sections 8.3.3 and 8.3.4 indicate that a minimum 1.086 T/W is necessary to meet MIL-F-83300 Level 2 hover flying qualities requirements. Careful assessment of mid-transition control requirements will also be essential. Fuel consumption in transition is higher because A/B light-off occurs much earlier.

In summary:

- Two-engine reconversions appear to be relatively problem free
- High angle of attack aerodynamics could be critical, especially if there are large pitching moment excursions
- Automatic flight path and throttle control may be desirable, although pilot controlled reconversions were performed well on the X-13
- A variety of flight path options and landing configuration aerodynamics may be needed to minimize buffet in reconversion
- Reconversion fuel usage leaves ample reserve for final docking
- Ample excess T/W is available for height and attitude control for a normal two-engine reconversion
- Single-engine T/W margins and flight paths will have to be established very carefully; T/W = 1.03 is unrealistically low

8.3.2 Takeoff Conversion

Conversion performance (transition from hover to conventional flight) was evaluated at the design mission takeoff weight of 23,375 pounds. Level flight was achieved within 10.4 seconds after reaching full throttle. Initial climb speed was reached in 17.0 seconds with less than 400 pounds fuel burned. The minimum excess thrust to weight was 1.29 at hover which provided substantial margin for height and attitude control. The flight path was selected to minimize pilot exposure to non-recoverable engine failure (See Figure 8-14).

Conversions were simulated using a modified version of the computer routine used to calculate reconversions. The program was modified to maintain level flight once it was reached. Conversion began with a pitchover to angle of attack for 0.9 $C_{L_{max}}$. Angle of attack was held constant until rate of climb peaked and then decreased to intercept level flight. Maximum thrust was used until the end of conversion. Fuel use calculated for this conversion was conservative because a pilot would normally reduce thrust to intermediate power at a lower level flight speed. The time history shown in Figure 8-15 shows very smooth variations of all variables. This profile should create minimal pilot workload with increasing buffet due to airspeed used to cue the pilot to pushover. An automatically controlled conversion is possible with pitch angle and climb rate sensing, but considerable development will be needed to assure compatibility over a wide range of operating conditions.

In summary, the outbound transition maneuver does not appear to present any serious problems, but the details warrant further study to develop pilot procedures to minimize time and fuel without approaching the stall region.

8.3.3 Attitude Control System

Final determination of VATOL control power requirements will necessitate manned simulation including effects of ship motion. For this study, maximum available control power was evaluated against MIL F-83300 and AGARD 577 criteria revised to reflect VTOL flight test experience and VATOL operating characteristics. Maximum available control power was determined for two types of attitude control systems. The basic SF-121 system comprises thrust vectoring control (TVC) for pitch and yaw and reaction jet roll control. An alternate system using reaction jet control about all axes was compared to the basic system. Results showed the basic system to be distinctly superior. As a result of these studies, revised design thrust to weight criteria were developed. These increase minimum thrust to weight used to design the SF-121. An apparent problem uncovered was serious directional instability at above 26 degrees angle of attack. Configuration development wind tunnel tests to reduce or eliminate the instability are clearly indicated.

The two types of attitude control systems analyzed are illustrated in Figure 8-16. The basic SF-121 system uses engine nozzles gimballed in pitch and yaw, with compressor bleed jets at the wingtips for roll control. For single engine transition, the operating nozzle is biased through the

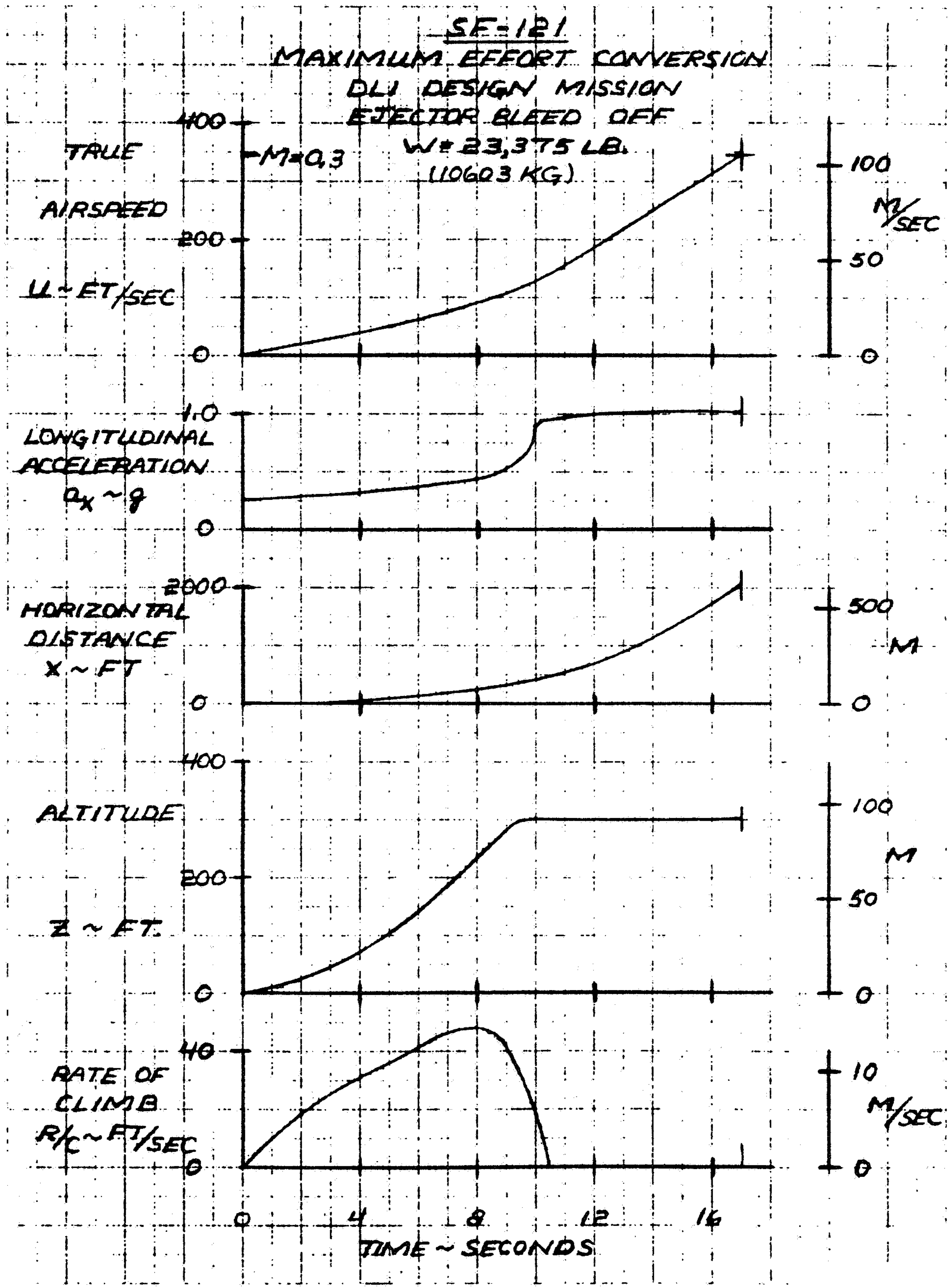


Figure 8-15a - Maximum Effort Conversion

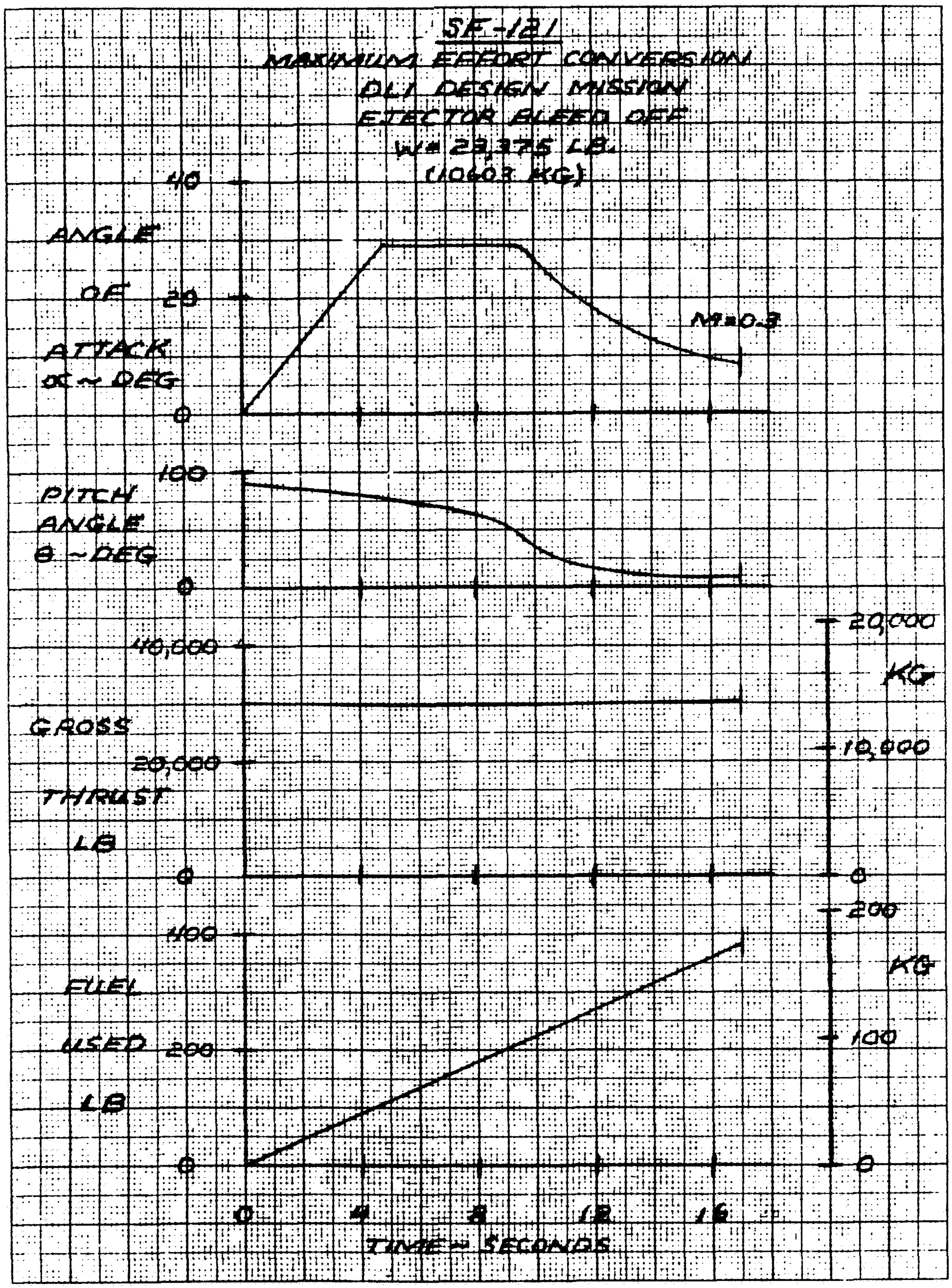


Figure 8-15b - Maximum Effort Conversion

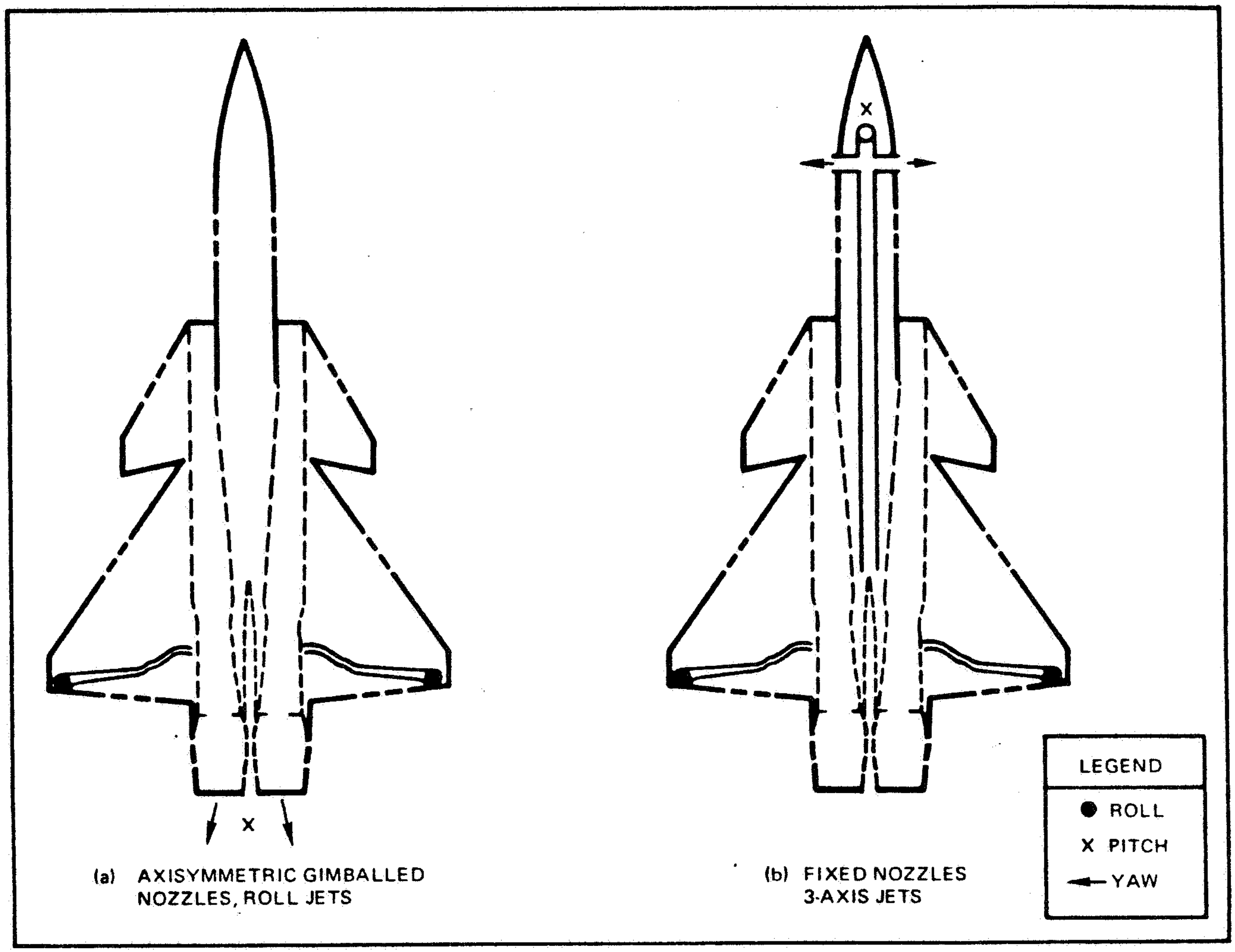


Figure 8-16 - Candidate Attitude Control Systems

center of gravity to maintain trim. The original SF-120 proposed in Reference 14 included a jet flap elevon for roll control. Aerodynamic predictions indicated a reversal in aileron effectiveness at high angles of attack. Thus, the jet flap was dropped in favor of wing tip reaction jets. Differential thrust deflection was also considered for roll control. However, there was no apparent way to apply it for engine out conditions. Its effect on control phasing is discussed in the following Section. Dynamic response is similar to an airplane with a horizontal tail, in that a pitch or yaw control input initially acts in a direction opposite to the desired motion. In contrast, the three axis bleed system is analogous to a canard control, where a pitch or yaw control force acts in the desired direction of translation. The all-bleed system suffers high thrust losses, but has the advantage of a fixed main engine nozzle. A comparison of the two systems is presented in the following paragraphs.

Before proceeding with a comparison of the basic and alternate control systems the attitude control systems, the hover attitude control requirements had to be revised. Changes were needed to reflect NAVAIR revisions of MIL-F-83300 and VATOL peculiar characteristics (See Tables 8-5 and 8-6). Yaw and roll requirements have been transposed to accommodate the VATOL landing attitude. A flat riser uses pitch and roll for translation whereas a VATOL aircraft uses pitch and yaw for translation. This philosophy is also carried through for control in hover where the stick is used for translation control (See Section 8.3.4 for complete discussion).

Table 8-5 - VATOL Hover Minimum
Attitude Change in One Second or Less (Degrees)

LEVEL	PITCH	YAW (ROLL)**	ROLL (YAW)**
1	$\pm$ 4.0	$\pm$ 6.0	$\pm$ 3.0
2	$\pm$ 2.5	$\pm$ 3.0	$\pm$ 2.0
3	$\pm$ 2.0	$\pm$ 2.0	$\pm$ 2.0
AGARD 577*	$\pm$ 3.0	$\pm$ 3.0	$\pm$ 6.0

* AGARD 577 - Attitude Command

** Roll and yaw sense transposed here for VATOL; roll is about SF-121 longitudinal axis.

Table 8-6 - VATOL Hover
Response to Control Input in One
Second or Less (Degrees per Inch)

LEVEL	PITCH		YAW		ROLL	
	MIN	MAX	MIN	MAX	MIN	MAX
1	4.0	20	6.0	23	3.0	23
2	2.5	30	3.0	45	2.0	30
3	1.0	40	1.0	50	1.0	40
AGARD 577*	3.0	-	3.0	-	6.0	-

* AGARD 577 - Attitude Command

Control power required to meet the minimum levels specified in Table 8-5 has been calculated. A step input with first order lag and control time constant of 0.1 second was assumed. The minimum levels are presented in Table 8-7 below for both MIL-F-83300 and AGARD 577.

Table 8-7 - Minimum Control Power Required
In Hover For VATOL Attitude Change in One Second*
(Radians/Seconds^2)

LEVEL \ RATE	PITCH ($\ddot{\Theta}$)	YAW ($\ddot{\psi}$)	ROLL ($\ddot{\phi}$)
1	0.176	0.261	0.130
2	0.099	0.130	0.087
3	0.088	0.087	0.087
AGARD 577**	0.132	0.130	0.260

* Assumed: Step input with first order lag and control time constant T_C = 0.1 seconds.

** AGARD 577 with attitude command for maneuver, trim and upset.

The following discussions pertain only to control in hover. Control in transition is reviewed in Section 8.3.4. Maximum pitch, yaw, and roll control available is compared to required control levels for the basic and alternate systems respectively. The maximum VTOGW and DLI VLGW loadings were selected to illustrate the extremes of hover control available with two engines operating. Single engine vertical landing was chosen to evaluate the validity of the SF-121 design thrust to weight. Assumptions used for calculation of maximum available control were:

- o All controls are single axis (no cross coupling)
- o Control inputs are limited to values which maintain T/W $\geq$ 1.0
- o Reaction jet controls use demand bleed.
- o Pitch and yaw control are limited by thrust vector deflection of 15 degrees only.
- o Roll control is limited by available excess thrust to weight. (For single engine vertical landing additional roll control can be obtained by allowing transient overtemperature.)
- o Inertia and weights are for the SF-121 point design. A summary of weights and inertias for configurations evaluated in this section is presented as Table 8-8.
- o Propulsion characteristics are for the 1.131 scale engine of the SF-121 point design.
- o Reaction jet thrust for alternate control system pitch and yaw thrusters is twice that for each roll reaction jet
- o Thrust available for all transitions is with ejector bleed off.

Maximum hover control powers available about pitch, yaw and roll axes, for both control systems are presented in Figure 8-17 and 8-18 and Table 8-9. For comparison purposes, maximum control power required per MIL-F-83300 has been postulated. This was done by extrapolating control power required per inch of control motion to typical maximum stick and pedal throws. The throws selected for maximum control input are:

Control	Throw
Longitudinal (Stick)	$\pm$ 6.0 inches
Lateral (Stick)	$\pm$ 2.5 inches
Directional (Pedal)	$\pm$ 2.5 inches

Table 8-8 - Summary of Weights and Inertias For Attitude Control System Study

CONFIGURATION	W (LBS.)	C.G. (F.S.)	I_{XX} (SLUG-FT^2)	I_{YY} (SLUG-FT^2)	I_{ZZ} (SLUG-FT^2)
MAX. VTOGW (1)	27,500	401.4	17,438	62,153	75,838
S/E DLI VLGW (2)	14,662	401.6	5,923	51,257	55,450
DLI VLGW (3)	16,299	400.6	7,726	52,609	58,338

NOTES: (1) Sized to 1.1 T/W with ejector bleed off. Full fuel included.

(2) Sized to 1.03 T/W with ejector bleed off. Stores off. 1,000 pounds fuel.

(3) Includes DLI mission stores plus 1,000 pounds fuel.

Control power per inch of control throw is that tabulated in Table 8-7. AGARD 577 control power is as required for maneuver trim and upset. Maximum control power required for all levels is plotted on Figures 8-17 and 8-18. Levels shown in Table 8-9 are those required for the respective design conditions. It is recognized that a final design control system will probably not be linear. Usually, initial control gains are higher than used here and the maximum levels are approached at a much lesser gain. Level 1 control is highly desirable for normal two engine operation. Level 2 control was the design goal for single engine flight.

At maximum vertical takeoff gross weight the basic control system provides control exceeding Level 1 and AGARD 577 requirements about all axes. This is primarily due to the high thrust level required for takeoff. The pure reaction control system is inadequate about all axes. Core engine bleed corresponding to the 10 percent excess thrust (3.9 percent) provides only 470 pounds of thrust for pitch and yaw control, or 10,000 foot-pounds of moment. This compares to over 92,000 foot-pounds available from the basic thrust vectoring system. Roll control for the alternate system is, of course, indentical to that for the basic system. Both systems provide roll control much greater than the minimum required.

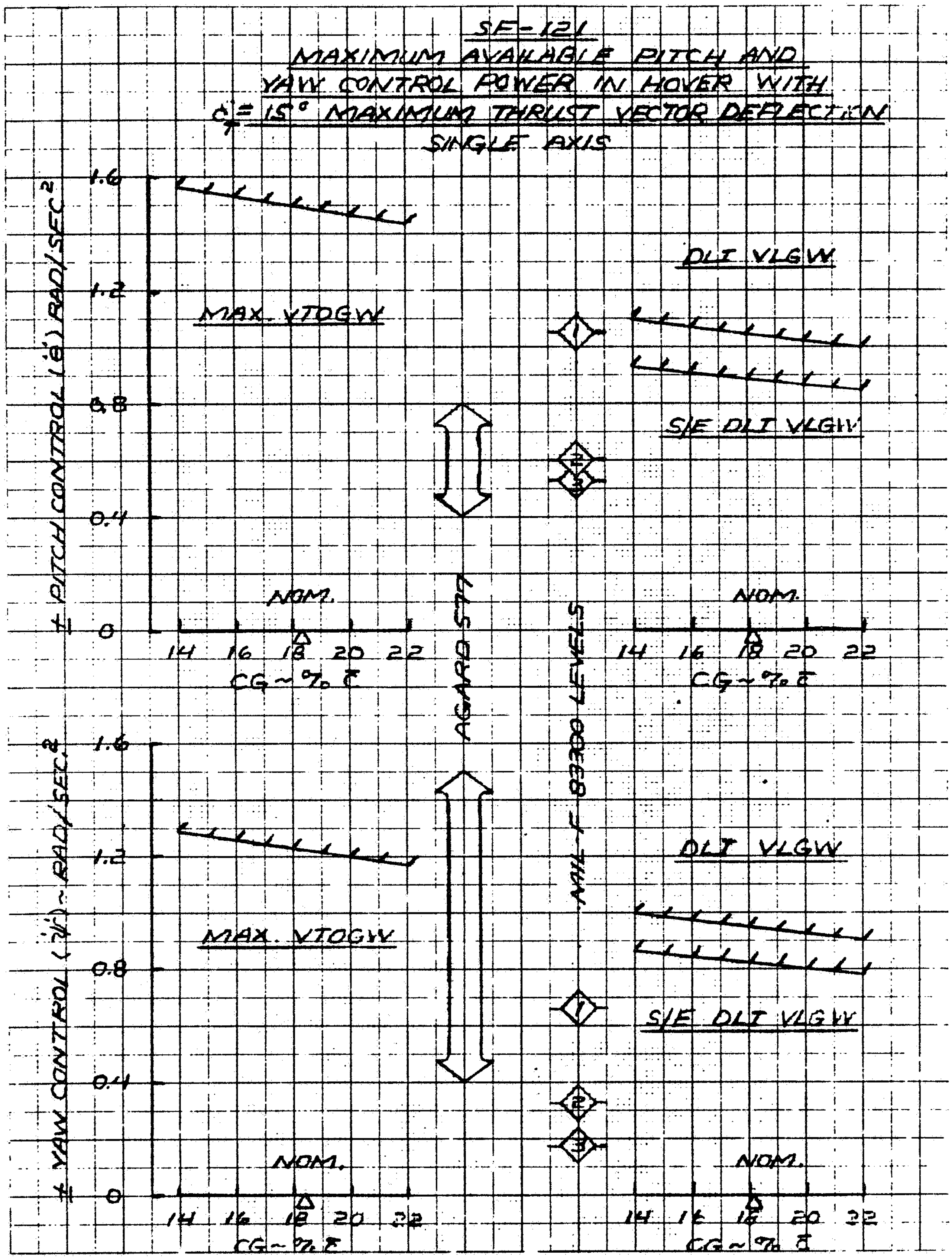


Figure 8-17 - Hover Control Power with TVC + Roll Jets

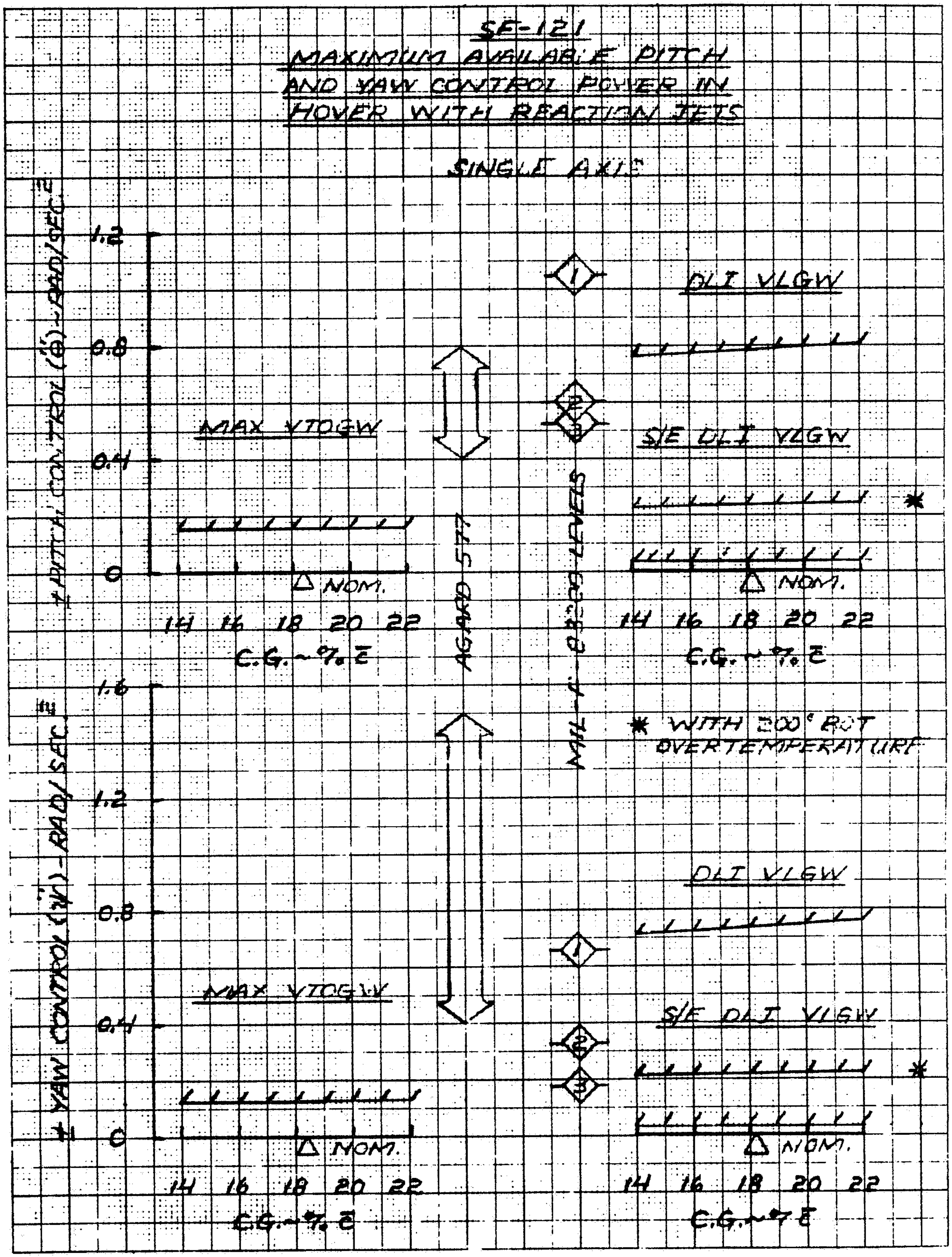


Figure 8-18 - Hover Control Power with 3 Axis Reaction Jets

Table 8-9

SF-121 Maximum Available Roll

Control Power in Hover With Continuous Bleed

Single Axis

CONFIGURATION	PERCENT BLEED	ROLL CONTROL $(\ddot{\phi})$ - RAD/SEC^2
MAX VTOGW W = 27,500 LB I_{xx} = 17,438 $SLUG\text{-}FT^2$	3.9 (1)	0.433 (0.325) (5) (0.35-0.8) (7)
S/E DLI VLGW W = 14,662 LB I_{xx} = 5,923 $SLUG\text{-}FT^2$ (SINGLE ENGINE)	7.0 (2) (4)	1.325 (0.325) (5) (0.35-0.8) (7)
	1.0 (3) (4)	0.163 (0.218) (6) (-) (7)
DLI VLGW W = 16,299 LB I_{xx} = 7,726 $SLUG\text{-}FT^2$	3.9	0.852 (0.325) (5) (0.35-0.8) (7)

NOTES:
(1) BASED ON 1.1 T/W EXCESS THRUST.

(2) BASED ON 1.03 T/W EXCESS THRUST PLUS 200°F BOT OVERTEMPERATURE.

(3) BASED ON 1.03 T/W EXCESS THRUST.

(4) AT 8.0 DEGREES YAW TO ALLOW VERTICAL THRUST VECTOR TO PASS THROUGH C.G.

(5) LEVEL 1 MINIMUM CONTROL POWER REQUIRED AT MAXIMUM THROW.

(6) LEVEL 2 MINIMUM CONTROL POWER REQUIRED AT MAXIMUM THROW.

(7) AGARD 577 MINIMUM CONTROL POWER REQUIRED AT MAXIMUM THROW.

At the normal two engine landing condition the basic system provides adequate control about all axes. The alternate system is satisfactory for roll and yaw control. It does not meet the postulated Level 1 requirement for pitch control, but is adequate for AGARD 577. It should be noted that 15 percent bleed (2,000 pounds reaction jet thrust) was assumed for the all-reaction jet pitch and yaw control. Bleed for roll control was set at 3.9 percent for comparison with the takeoff case.

For single engine landing, the control requirements were relaxed to Level 2. The basic control system provides satisfactory control power in pitch and yaw, but is deficient in roll with only 3 percent thrust margin available for bleed. If a transient 200°F BOT overtemperature is allowed more than ample roll control is available. In the absence of an overtemperature allowance, an increased engine size would be needed. Subsequent discussion reviews the impact of no overtemperature allowance on aircraft sizing. The three axis reaction jet control system performance with the 3 percent thrust excess is completely unsatisfactory. With the overtemperature allowance, roll performance again exceeds Level 1 assumed requirements. The basic system again is clearly superior but an attractive sizing factor is apparent if transient overtemperature is allowed for emergency operations.

Capability of the SF-121 basic control system to meet MIL-F-83300 and AGARD 577 attitude control response requirements is illustrated in Table 8-10 below. Minimum requirements are met in all cases except for single engine DLI vertical landing roll control response for AGARD 577 requirements. As shown, a 200°F BOT transient overtemperature allows a substantial improvement. Higher design T/W would also yield acceptable performance. The responses in this table represent a likely minimum design capability. In practice, output to input gains are normally high at low control input and decrease as more control is demanded. Responses for the alternate all reaction jet system are not shown because of the deficiencies discussed previously.

Aircraft sizing studies discussed in Section 8.1 revealed that single engine vertical landing thrust to weight was the principal engine sizing factor. Because of the deficiencies noted for single engine landing roll control the T/W = 1.03 sizing criterion appears inadequate. MIL-F-83300 paragraph 3.2.3.1 requires: "Simultaneous application of pitch, roll and yaw controls

Table 8-10 - SF-121 Response in Hover to Control Input In One Second or Less (Degrees per Inch)

CONFIGURATION	PITCH	YAW	ROLL
MAX. VTOGW (1)	5.25 (4.0) (3.0)**	11.14 (6.0) (3.0)**	3.9 (3.0) (6.0)**
S/E DLI VLGW (2)	3.45 (2.5)	7.45 (3.0)	1.50/12.1* (2.0)
DLI VLGW (1)	4.0 (4.0) (3.0)**	8.6 (6.0) (3.0)**	4.9 (3.0) (6.0)**

* With 200°F BOT Overtemperature

** AGARD 577 - with Attitude Command control system

NOTES: (1) Level 1 Minimum Requirements IN ( ).

(2) Level 2 Minimum Requirements IN ( ).

in the most critical combination produces at least the attitude changes specified in Table IV (Table 8-5 of this report) within one second from the initiation of control force application." T/W = 1.036 is needed to meet these attitude control requirements alone. Roll control absorbs 0.035 of the excess 0.036 T/W. Height control requirements in paragraph 3.2.5 of MIL-F-83300 call for an incremental vertical acceleration of 0.05 g. This is essentially a direct T/W increment because high disk loading aircraft have virtually no vertical damping. The steady state T/W = 1.02 would be satisfactory only for low disk loading vehicles such as helicopters. It is recommended that these requirements be additive, resulting in the three levels of control power in Figure 8-19. To achieve a safe single engine vertical landing, it is recommended that the powerplant be sized to yield a thrust/weight = 1.086. An alternative to simply scaling up the engines is to use a short term rating to cover roll control transients. For the SF-121 point design (T/W = 1.03), Level 2 would require only a 60°F overtemp. The corresponding scale-up would increase takeoff weight 400 pounds.

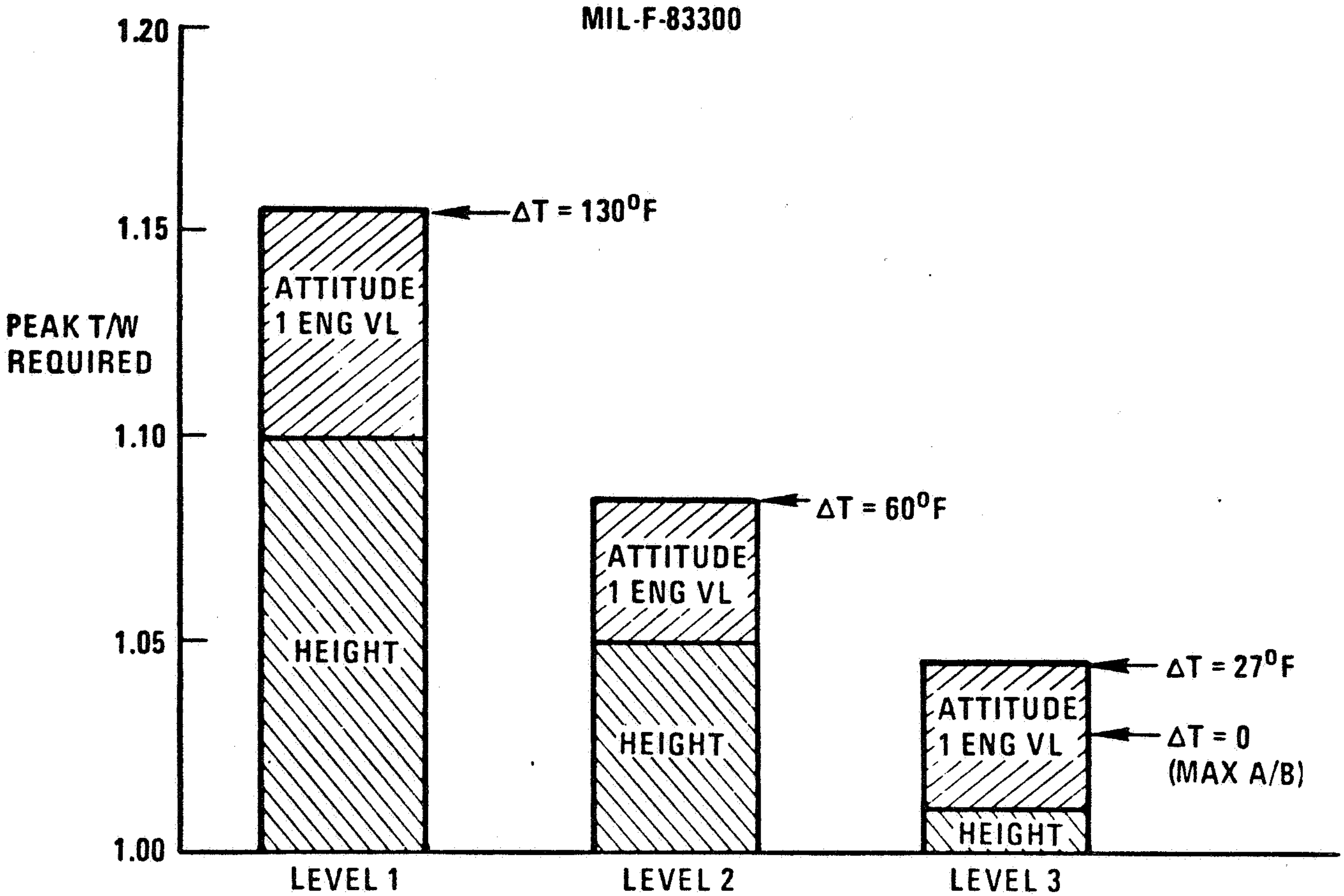


Figure 8-19 - Hover Control Power Requirements

Another concern regarding the control system is the effect of control application on horizontal translation. The basic system will act like a conventional aircraft control. That is, a rotational pitch or yaw control input, which is needed before a translation can be effected, imparts an initial force in the opposite direction. The all-reaction jet system force input is always in the direction of the desired motion. Translation in the wrong direction could be a problem during close in maneuvers near the landing platform. A simplified (no damping; step input with $T_C = 0$) translation maneuver was calculated for a two engine minimum weight landing condition. For this comparison equal reaction jet and TVC control power was applied to position the aircraft for translation. Results of the simulation are presented in Figure 8-20. The input control power is 0.4 rad/sec^2 which corresponded

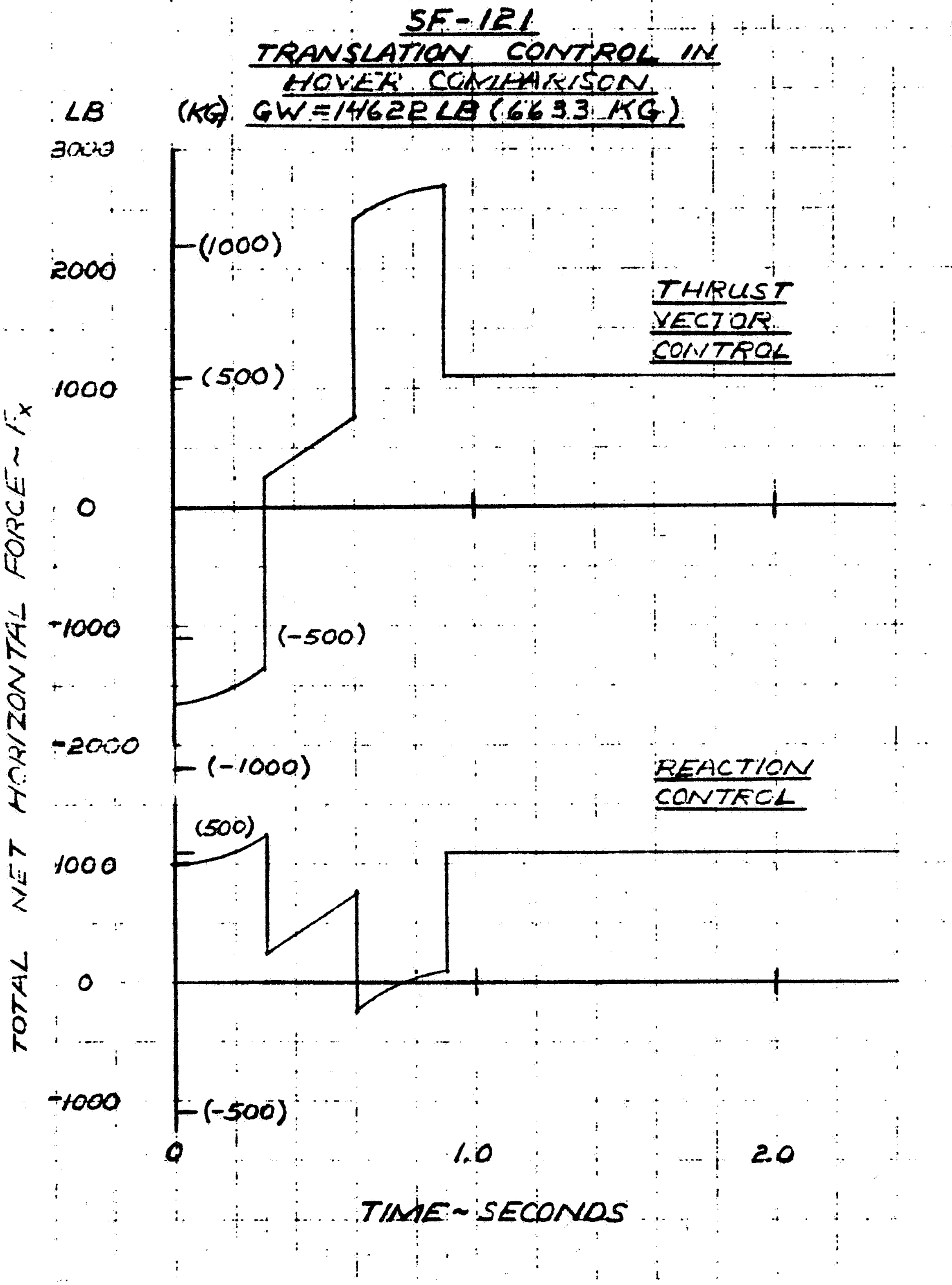


Figure 8-20a - Transient Response Comparison

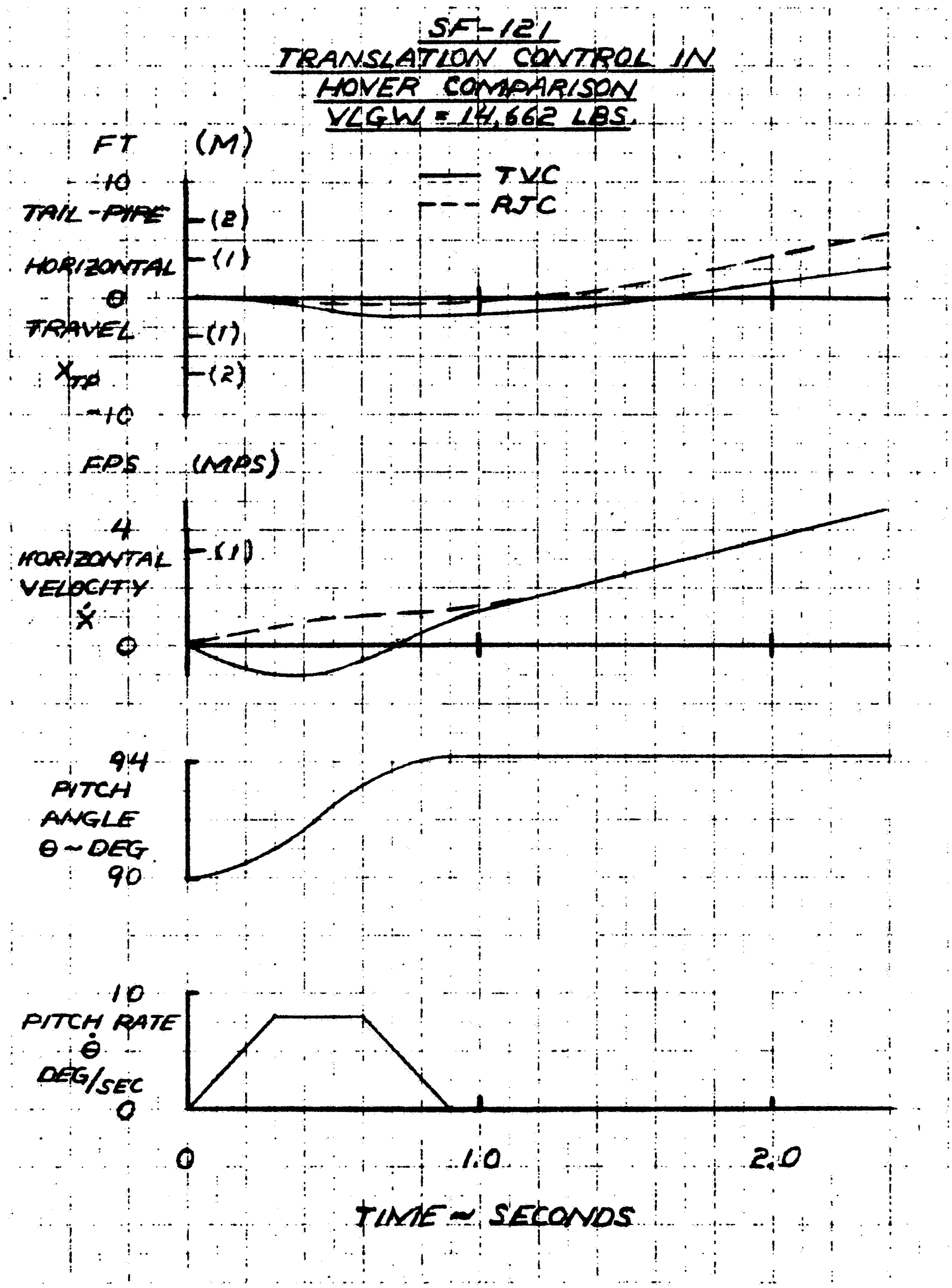


Figure 8-20b - Transient Response Comparison to 1,000 pounds of reaction jet force and 6.5 degree thrust deflection. Reaction jets are superior as a translation control. For a given input, more than twice the distance was traveled within 3.0 seconds with no adverse motion versus control from the basic system. Adverse transition at the center of gravity is approximately 0.5 feet. The bottom edge of the tailpipe, however, moves as much as 1.5 feet adversely. This is approximately one-third of the tail clearance available with the gear touching the landing platform.

Capability to trim the aircraft in vertical attitude in a 35 knot crosswind is ample. If we assume a side force drag coefficient of 1.0 the calculated side force is 870 pounds. The centroid of area or assumed center of pressure is at the center of gravity. Tilt required to trim out the crosswind is:

$$\beta = \sin^{-1}\left(\frac{870}{16,299}\right) = 3.06 \text{ degrees.}$$

For the single engine vertical landing the required tilt would be increased to 3.4 degrees. Confirmation of the aerodynamic estimates made for this calculation will have to come from wind tunnel tests.

8.3.4 Reconversion Control Phasing

Reconversion roll/yaw control phasing which minimizes opposite axis coupling with aircraft pitch attitude has been evaluated. Schedules determined maintain the relationship of pilot's conventional flight controls with the horizon through reconversion to hover which should minimize pilot workload and training time. The basic roll reaction jet plus yaw thrust vectoring and an all thrust vectoring system have been studied. Phasings of roll reaction jet thrust and differential thrust deflection with yaw thrust deflection have been determined for full lateral stick and rudder pedal inputs. Control power required at maximum control throw was set to meet MIL-F-83300 requirements. Phasing of the required control power from conventional to vertical altitude was made proportional to inertias about the respective control axes. Weights and inertias used are for the SF-121 design VL condition.
Results show that required thrust vector deflections and roll reaction jet thrust levels are easily attainable over the entire pitch range.

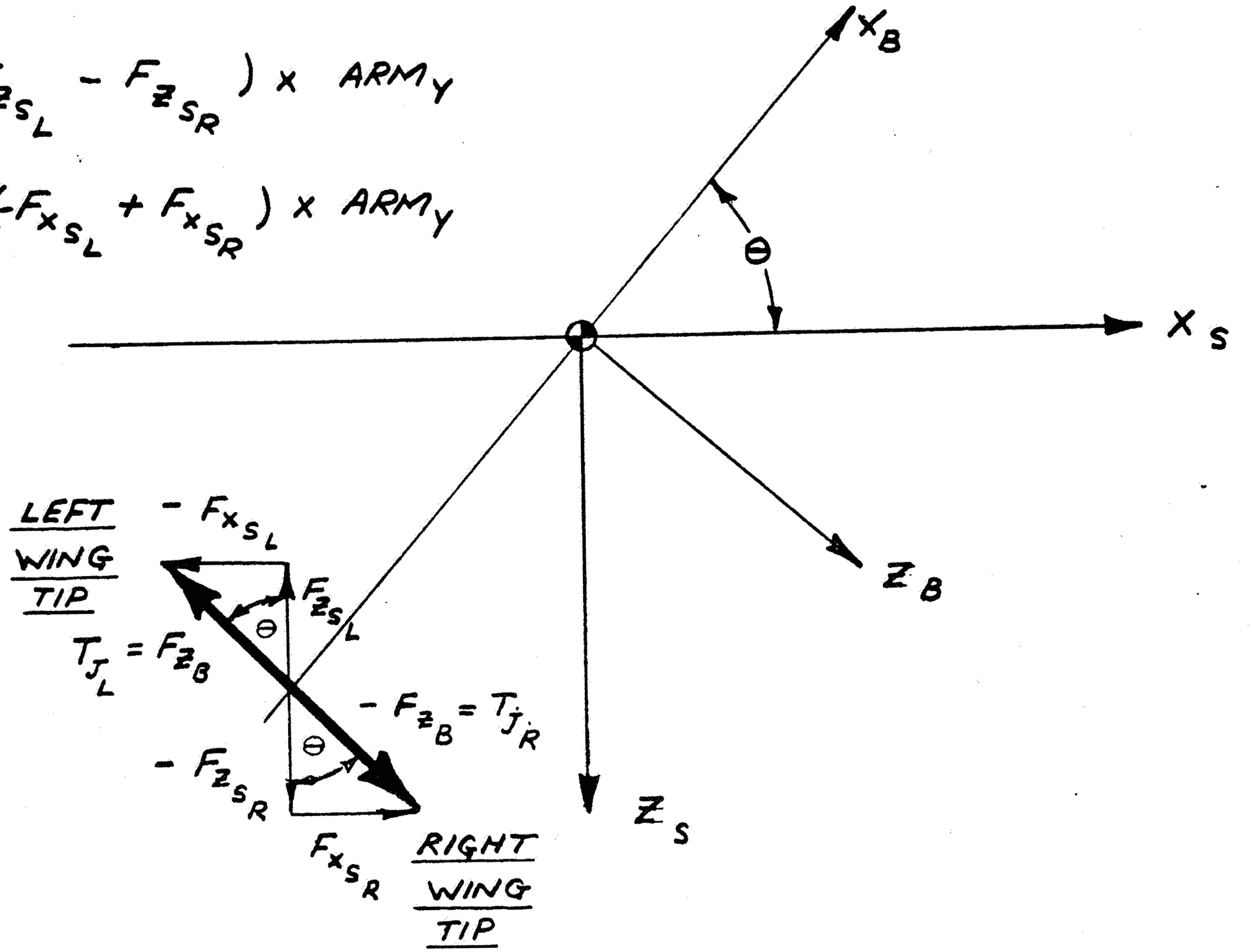
The nature of lateral-directional cross axis coupling for a VATOL aircraft in transition is illustrated in Figures 8-21 through 8-23. Aircraft body axes and body axis forces are noted with a 'B' subscript. Flight path or stability axes and stability axis forces are noted with an 'S' subscript. Direct forces due to engine or reaction jet thrust are indicated for positive control action (right wing down roll or nose left yaw). Both types of roll control cause adverse yaw at zero sideslip. If nose right sideslip is combined with RWD roll the results will differ. Adverse yaw decreases and becomes favorable with increasing right sideslip for differential thrust roll control. Increased adverse yaw will occur with increasing right sideslip and RWD reaction jet roll control. However, thrust vectoring yaw control induces favorable roll. Thus, it is likely that either roll/yaw control system will work satisfactorily. Clearly, extensive analyses will be needed to tailor the control phasing for all anticipated flight conditions. Phasing schedules presented in this section are for zero sideslip only.

Stability axis, three degrees of freedom lateral-directional force and moment equations were used to calculate control phasing schedules. Aerodynamic control forces and moments were calculated at full deflection. Low angle of attack control effectiveness was extrapolated using flap effectiveness vs. deflection characteristics from DATCOM. All aerodynamic control was phased out at 47 degrees angle of attack where aileron moment reversal occured and directional control became nil. The equations of motion were solved for thrust needed to provide the required moments.

Required yaw and roll control power was determined using the linear extrapolation method described in Section 8.3.3. Control power was specified in level flight at 1.1 V_{PAMIN}. Directional control meets Level 1 requirements of paragraph 3.3.10.1 of MIL-F-83300 which call for 6.0 degrees yaw attitude change within the first second following an abrupt step displacement of the yaw control with all other cockpit controls fixed. Roll control meets Level 1 requirements of paragraph 3.3.9 which calls for bank angle to change 30 degrees within 1.3 seconds from a trimmed zero roll rate condition. These are summarized in Figure 8-24 in terms of control power. Control power required vs. pitch angle is proportional to the inertias about the respective axes (See Figure 8-25). A breakdown showing the available aerodynamic control power is also presented

$$\ell_s = (F_{z_{S_L}} - F_{z_{S_R}}) \times ARM_Y$$

$$N_s = (-F_{x_{S_L}} + F_{x_{S_R}}) \times ARM_Y$$



- POSITIVE (RWD) ROLL CONTROL INDUCES ADVERSE YAW

Figure 8-21 - Roll-Yaw Coupling for Reaction Jets

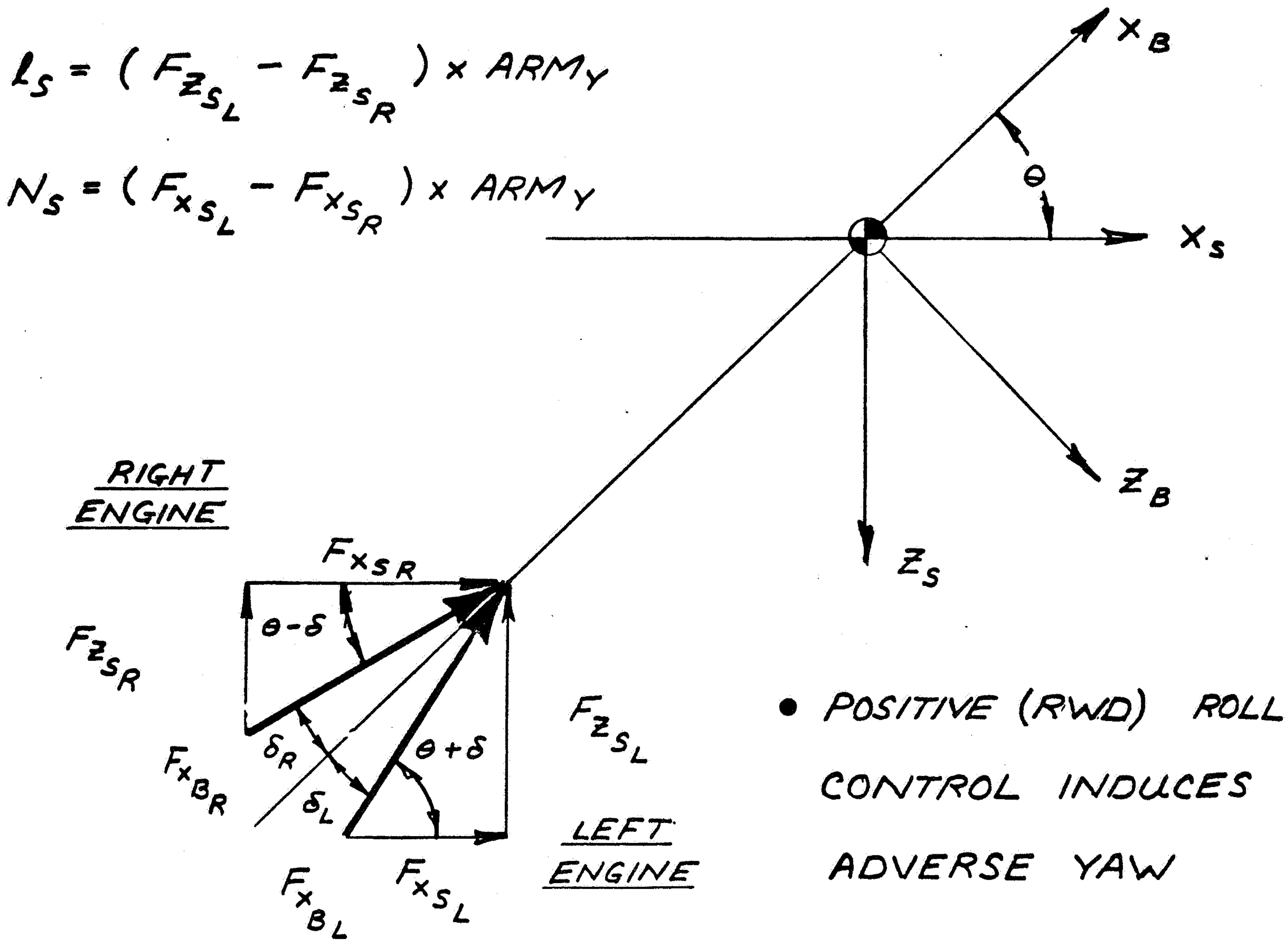
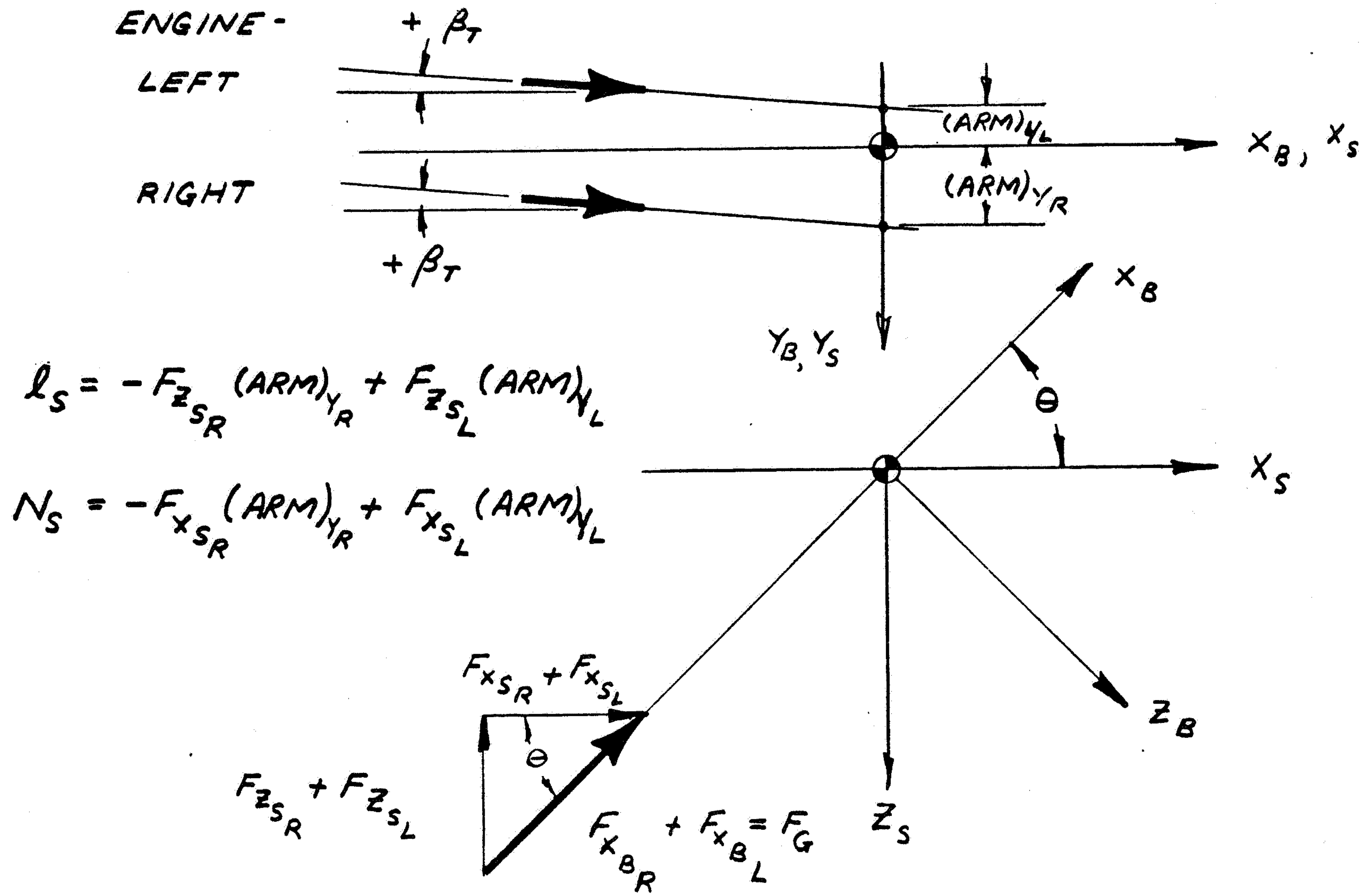


Figure 8-22 - Roll-Yaw Coupling Due to Differential Thrust Deflection



$$\ell_S = -F_{Z_{S_R}}(ARM)_{Y_R} + F_{Z_{S_L}}(ARM)_{Y_L}$$

$$N_S = -F_{X_{S_R}}(ARM)_{Y_R} + F_{X_{S_L}}(ARM)_{Y_L}$$

- POSITIVE (NOSE LEFT) YAW CONTROL INDUCES FAVORABLE ROLL

Figure 8-23 - Yaw-Roll Coupling Due to Thrust Deflection

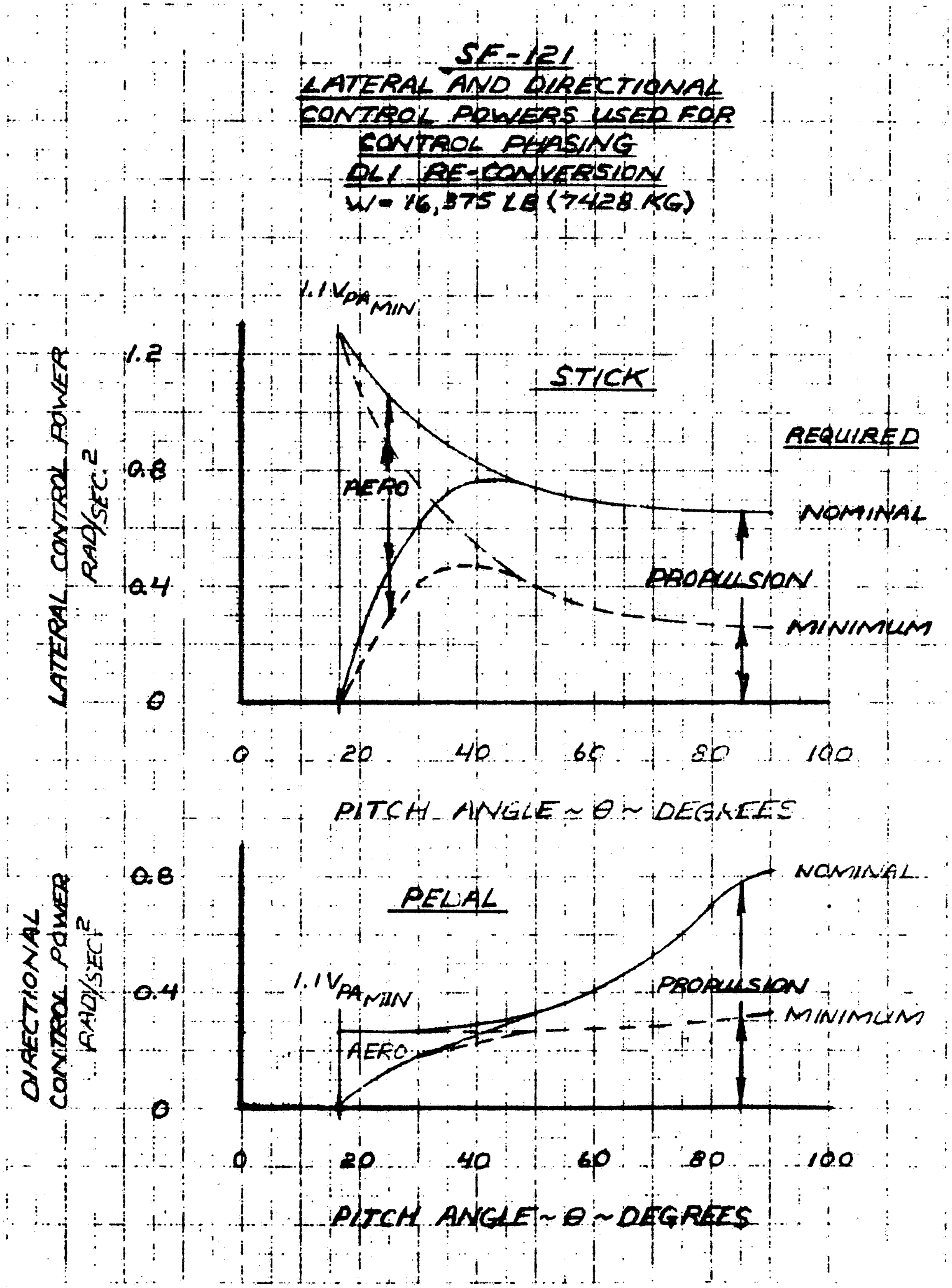


Figure 8-24 - Lateral and Directional Control Powers

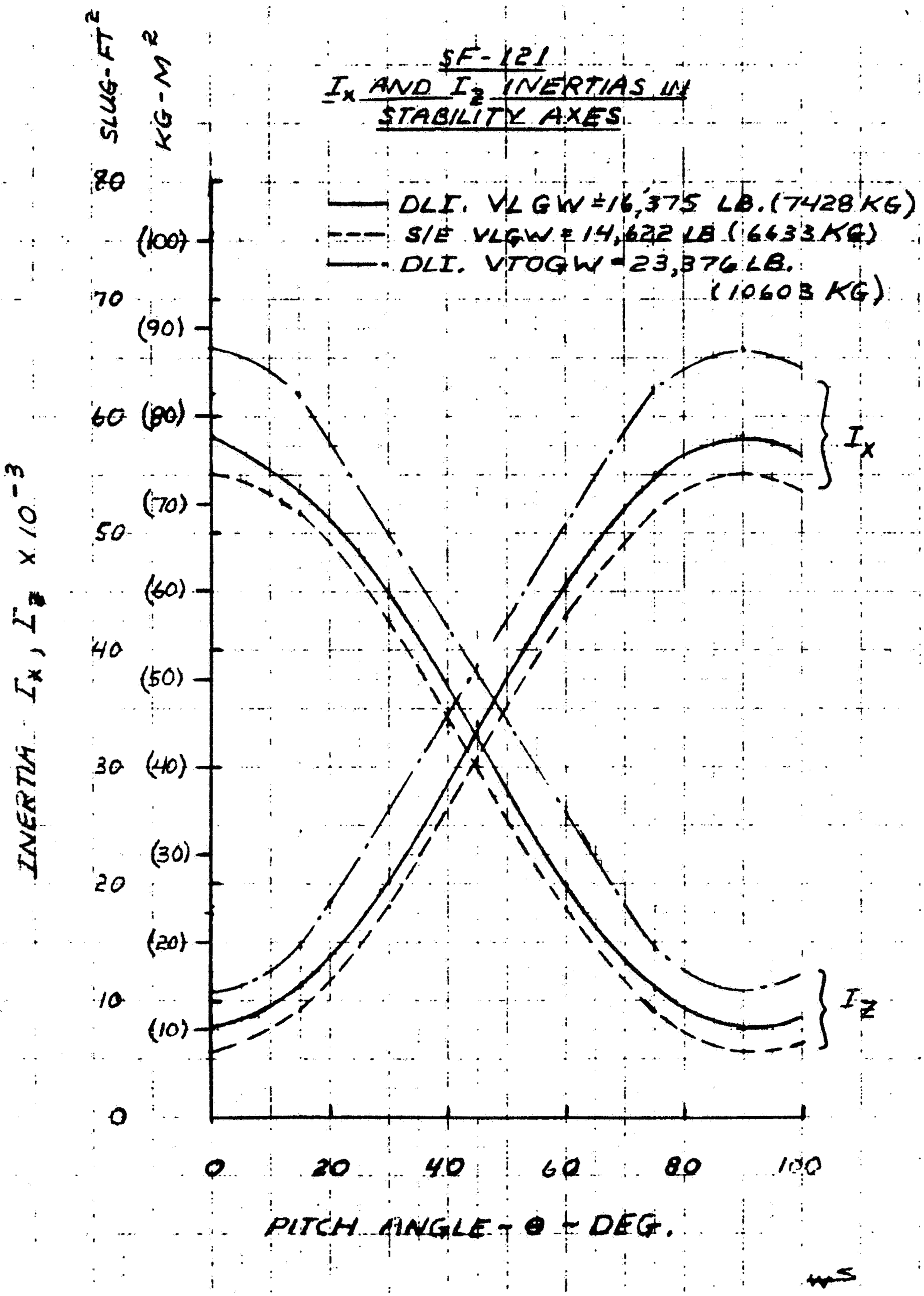


Figure 8-25 - SF-121 I_x and I_z Inertias to show its rapid decay with pitch attitude. The minimum requirement, which reduces control power at hover by 60 percent, was evaluated to determine the effect of reduced control power on reaction jet thrust. The resulting control inputs needed to meet the control power requirements are displayed in Figure 8-26 through 8-32.

Reaction jet thrust and thrust deflection angle vs. pitch angle for the basic SF-121 controls are presented in Figure 8-26. These are phased control outputs needed to meet the full throw stick and rudder control power requirements shown in Figure 8-24. Maximum required roll reaction jet thrust occurs near 45 degrees pitch angle where there is ample excess thrust for bleed. This means that adequate single engine roll reaction jet thrust should be available throughout the pitch range (see discussion in Section 8.3.1 on increased T/W margins required for roll attitude control). The rapid buildup of thrust deflection required for stick and pedal controls is due primarily to the low thrust levels near the stall plus the decay in aerodynamic control. Maximum thrust deflection is less than the 15 degrees throw available. The transition of the stick from a body X-axis control to a body Z-axis control is clearly illustrated. Of course, the opposite is shown for the pedal control. Quasi control input-output gearing is shown in Figures 8-27 and 8-28. For the case evaluated 100 percent authorities are the maximum values of Figures 8-26. Development of design gearings will require a thorough aerodynamics data base and extensive analysis.

Figures 8-29 through 8-31 show control phasing for the all thrust vector control system. Maximum asymmetric thrust deflection for roll to meet requirements was 9 degrees at 40 degrees pitch angle. Thus, only 6 degrees of authority remains for pitch control. Half of that, or 3 degrees, was required for trim. Full control or 15 degrees of deflection would be needed to meet the linearized maximum nose down control power indicated in Section 8.3.1. An alternate approach may be to deflect the nozzles toward each other when calling for roll control. This would increase the arm for roll control considerably with a concurrent reduction in asymmetric deflection for roll control. Thorough study of this area is needed before selection of thrust vectoring roll control.

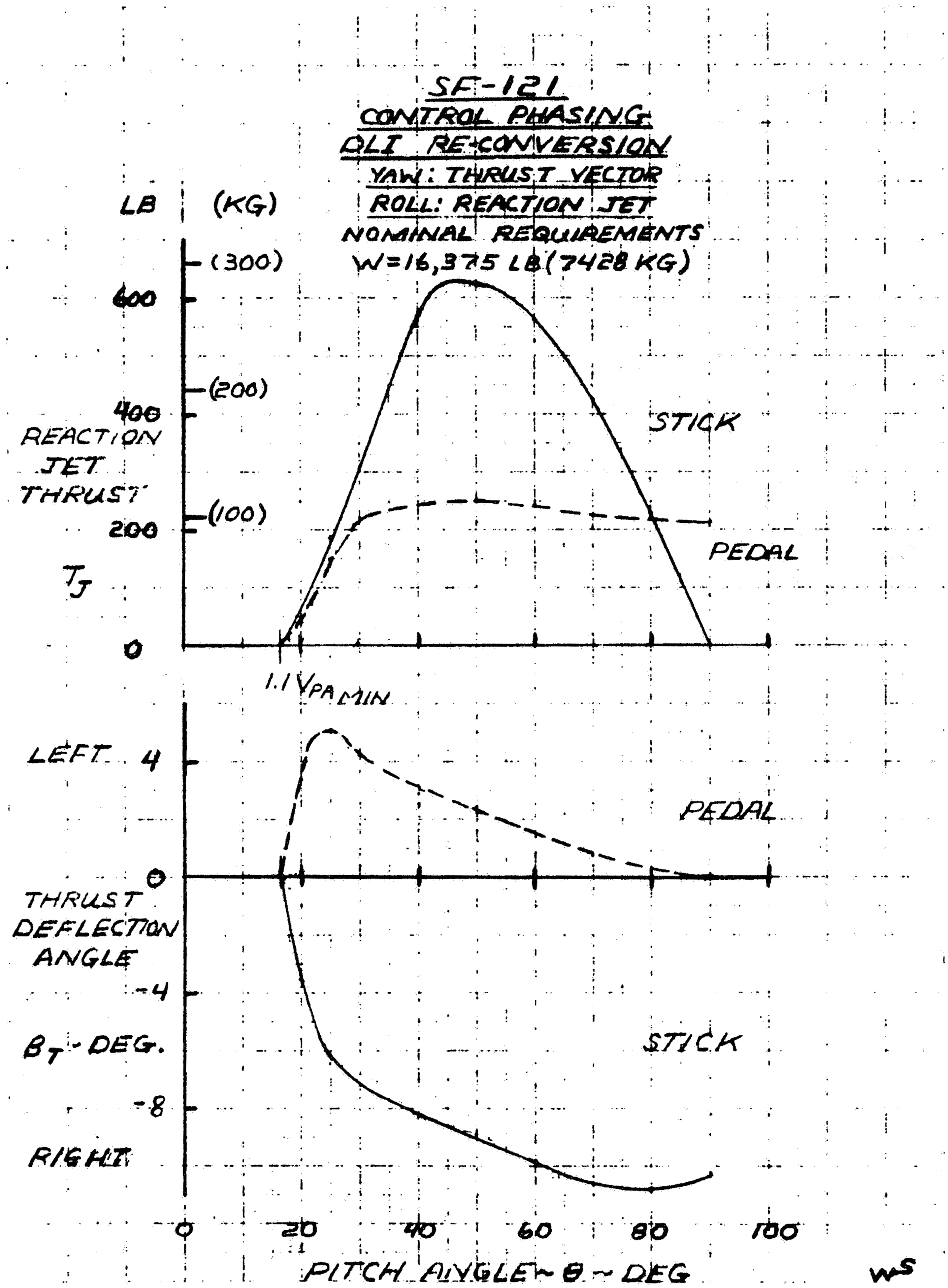


Figure 8-26 - SF-121 Control Phasing

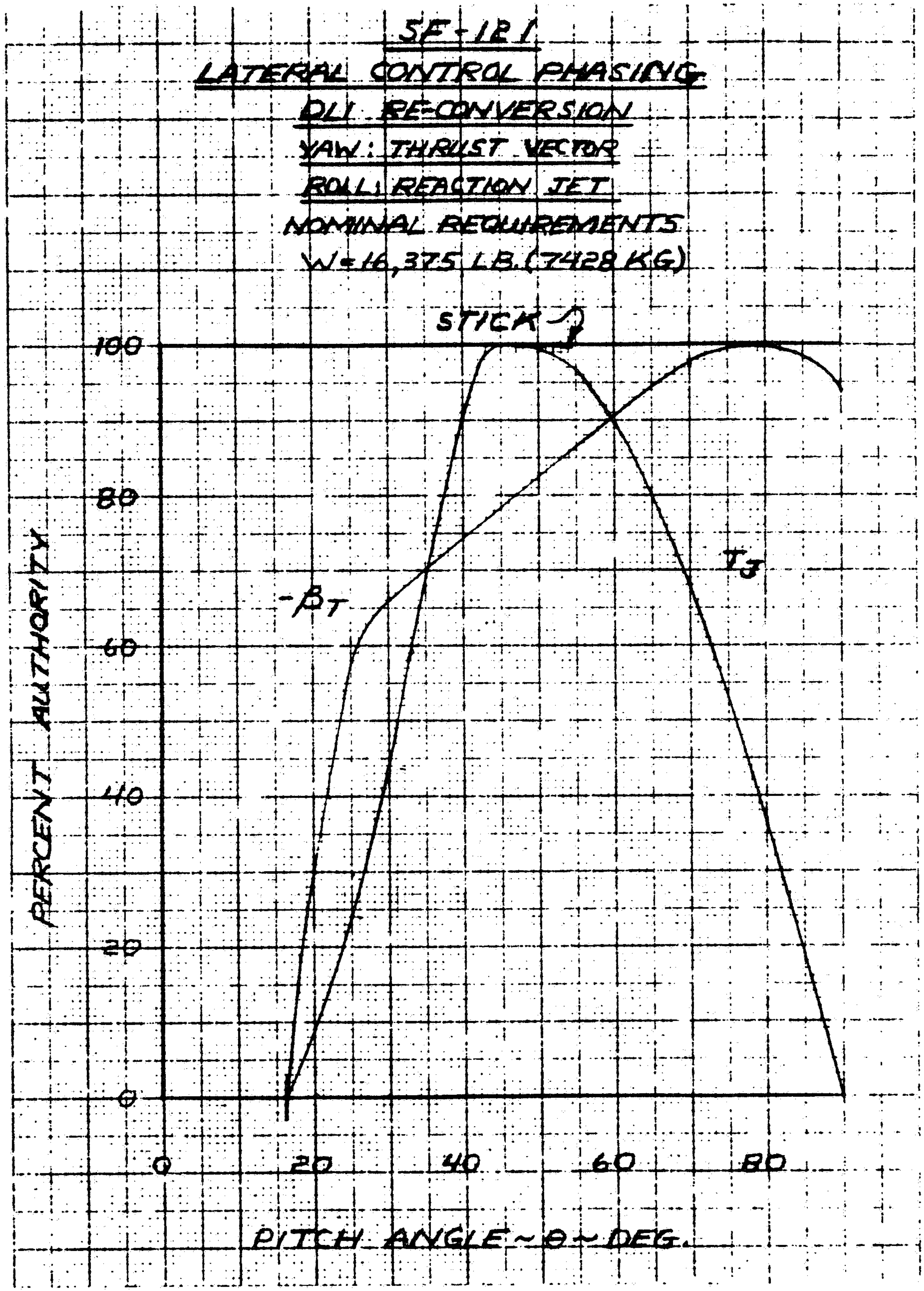


Figure 8-27 - Lateral Control Phasing

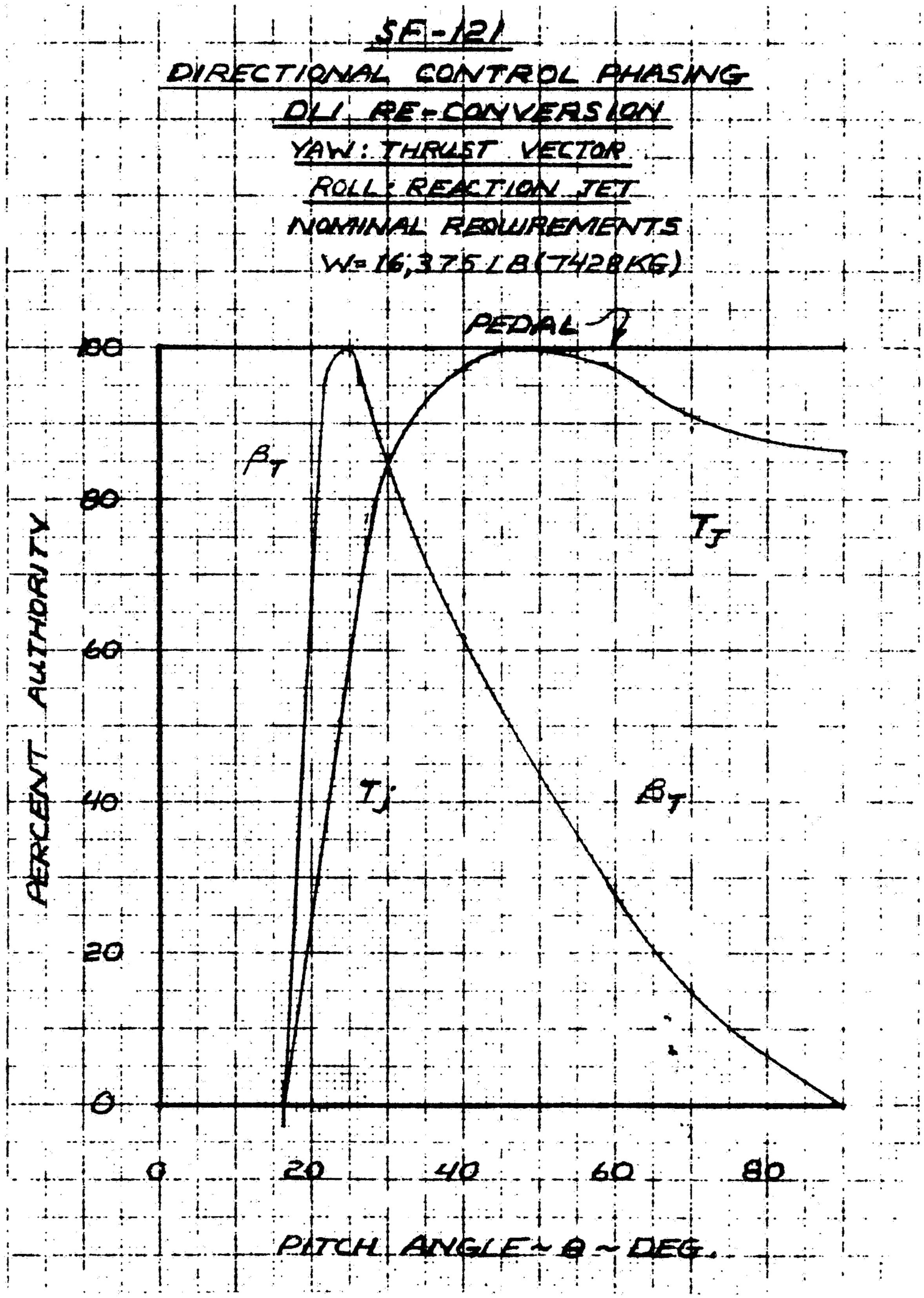


Figure 8-28 - Directional Control Phasing

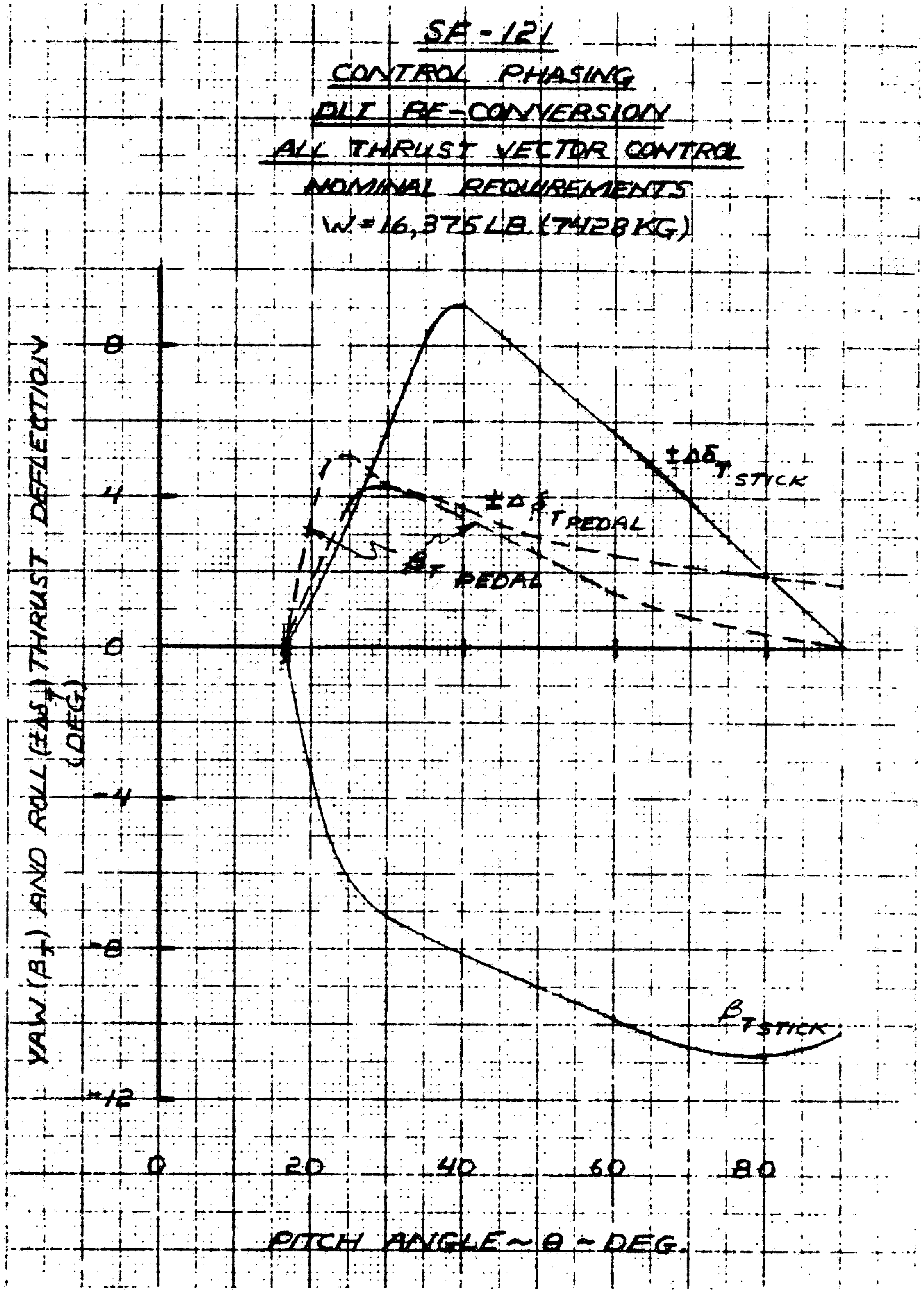


Figure 8-29 - Control Phasing - All TVC

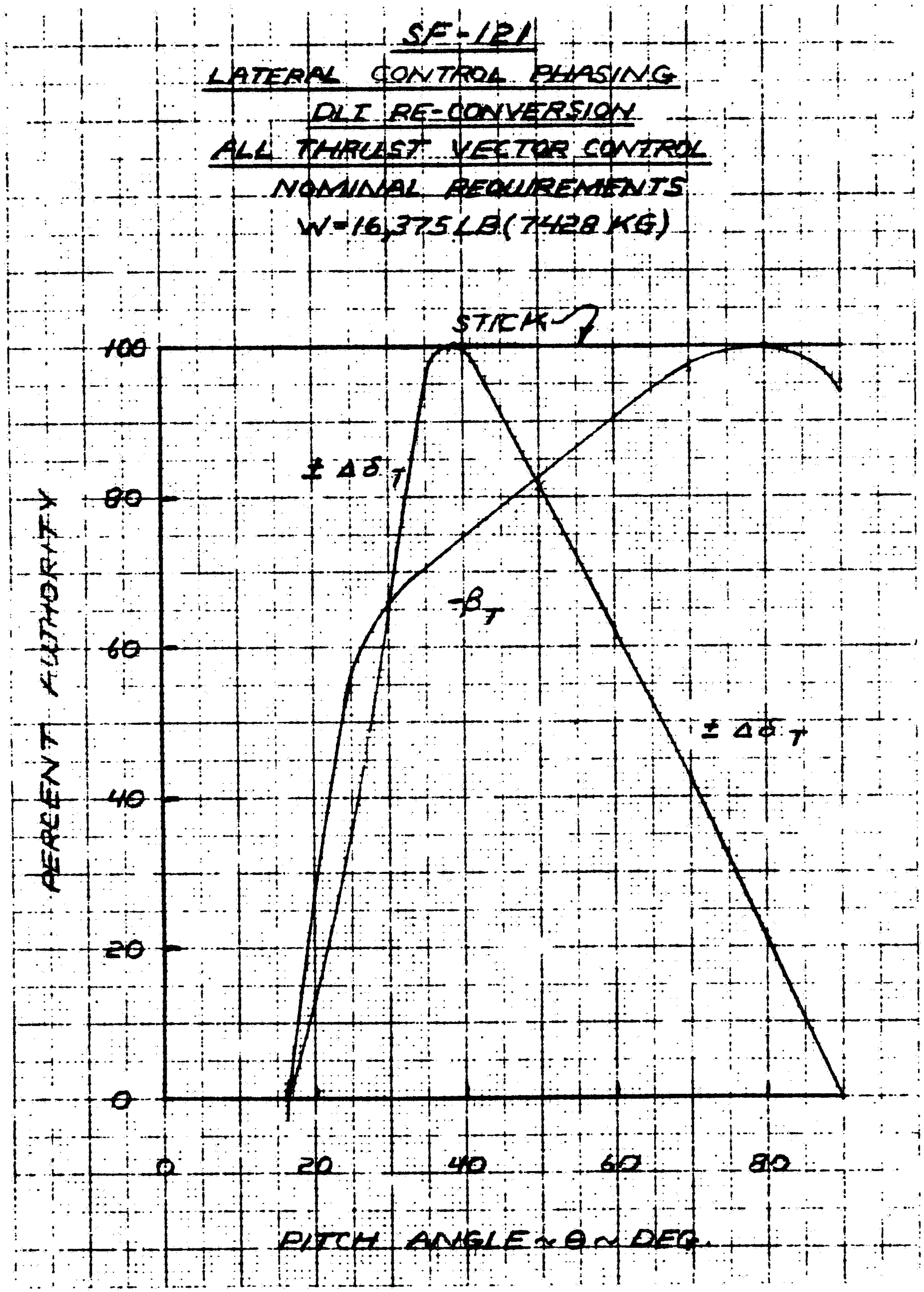


Figure 8-30 - Lateral Control Phasing - All TVC

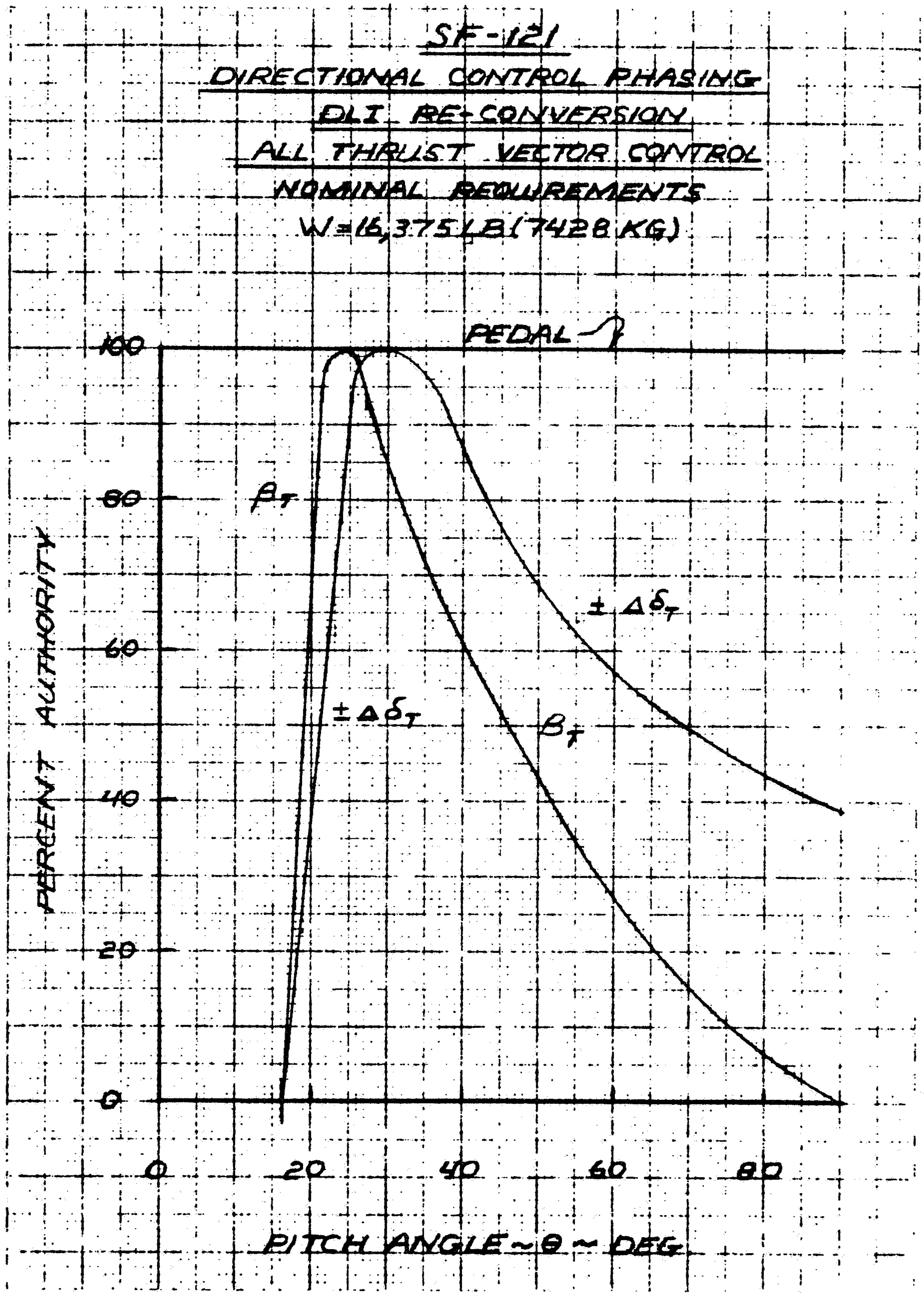


Figure 8-31 - Lateral Control Phasing - All TVC

Reducing the required control power to the minimum levels of Figure 8-24 resulted in decreased maximum roll reaction jet thrust and thrust deflection (see Figure 8-32). The net payoff of this approach would be to reduce the bleed required for roll control in vertical attitude. This would allow approximately a 1.5 percent reduction in single engine design T/W (see Figure 8-9). Mid transition roll reaction jet thrust is ample, even for a single engine landing (Figure 6-9). It is evident that roll control requirements would have to be relaxed considerably for SF-121 single engine hover with T/W = 1.03, but there is no apparent problem in meeting MIL-F-83300 requirements at mid transition conditions.

8.4 SHORT TAKEOFF

The Superfly concept has three distinct takeoff modes. The VATOL mode is used with small ship and Marine forward site basing. A free deck short takeoff can be made from ships with flight decks 300 feet long or greater. The STO mode permits naval operations at maximum gross weight, which is 10,000 pounds above design VTO weight. All shipboard landings are made in the vertical attitude; there are no catapulting and arresting provisions. The SF-121 can also operate in the CTOL made from runways.

The Superfly short takeoff is a dynamic maneuver in which thrust vector control is employed to rotate to a nose high attitude. The canard flaps and elevons are drooped to augment aerodynamic lift. As the aircraft nears the deck edge the pilot pulls the stick full aft, as with a catapult launch. Once the nose comes up the stick is moved forward to arrest rotation and maintain a 25 to 30 degree attitude angle for climbout.

The critical parameter is sink over the bow. In this respect the Superfly is like a conventional catapulted Navy aircraft. There is a brief transient upon departing the deck when the aircraft settles while pitch attitude is building up. A parametric analysis was performed to establish bounds on free deck takeoff feasibility. The results for a 400 foot flat deck, are presented in Figure 8-33. Sink over the bow is appreciable only for unrealistically low thrust to weights and high wing loadings, which do not apply to the SF-121.

Figure 8-34 shows SF-121 STO performance from a 400 foot deck. Even at the maximum weight with a 10,000 pound overload only ten knots wind over deck is needed to limit sink over the bow to five feet.

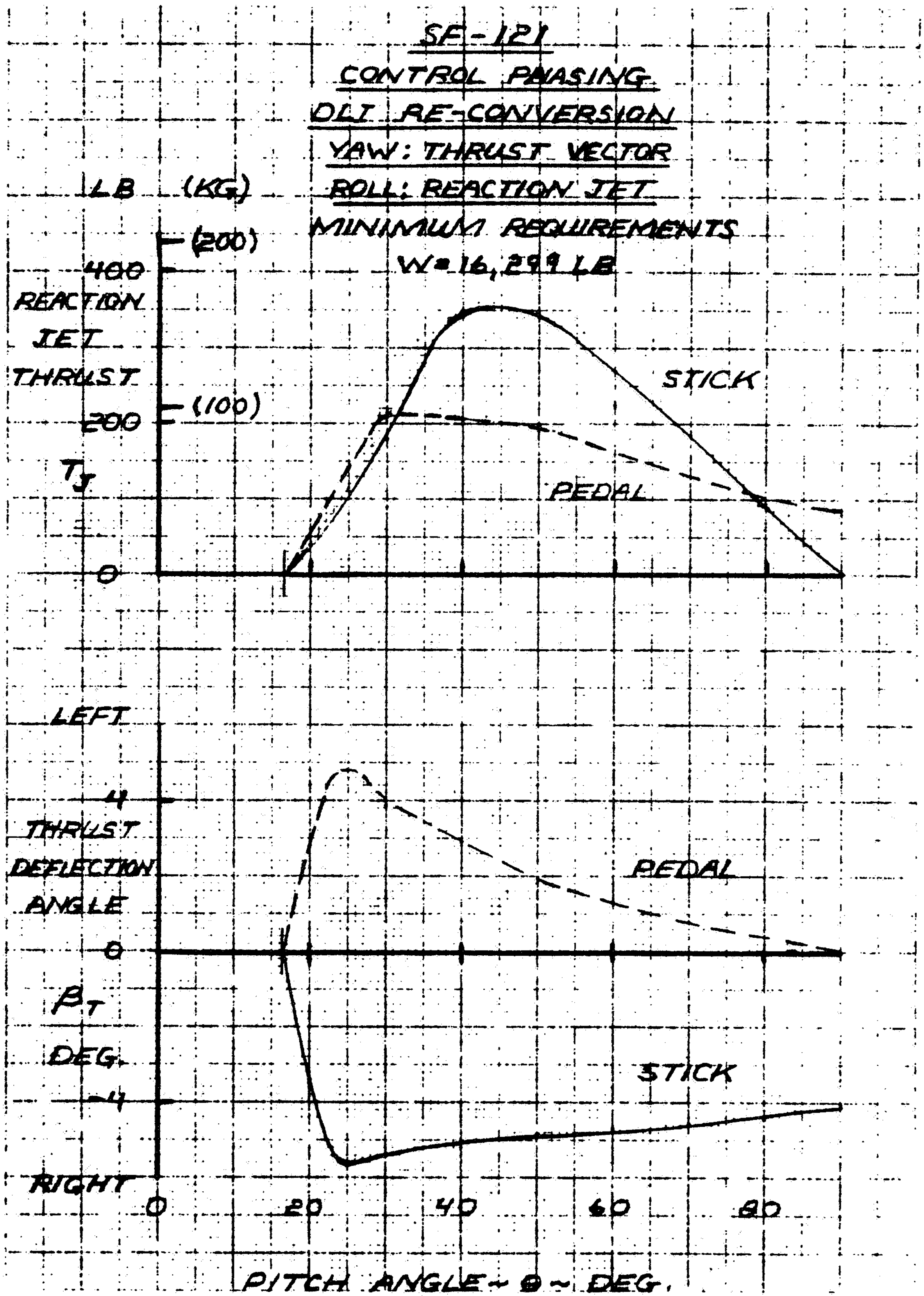


Figure 8-32 - Minimum Control Power Phasing

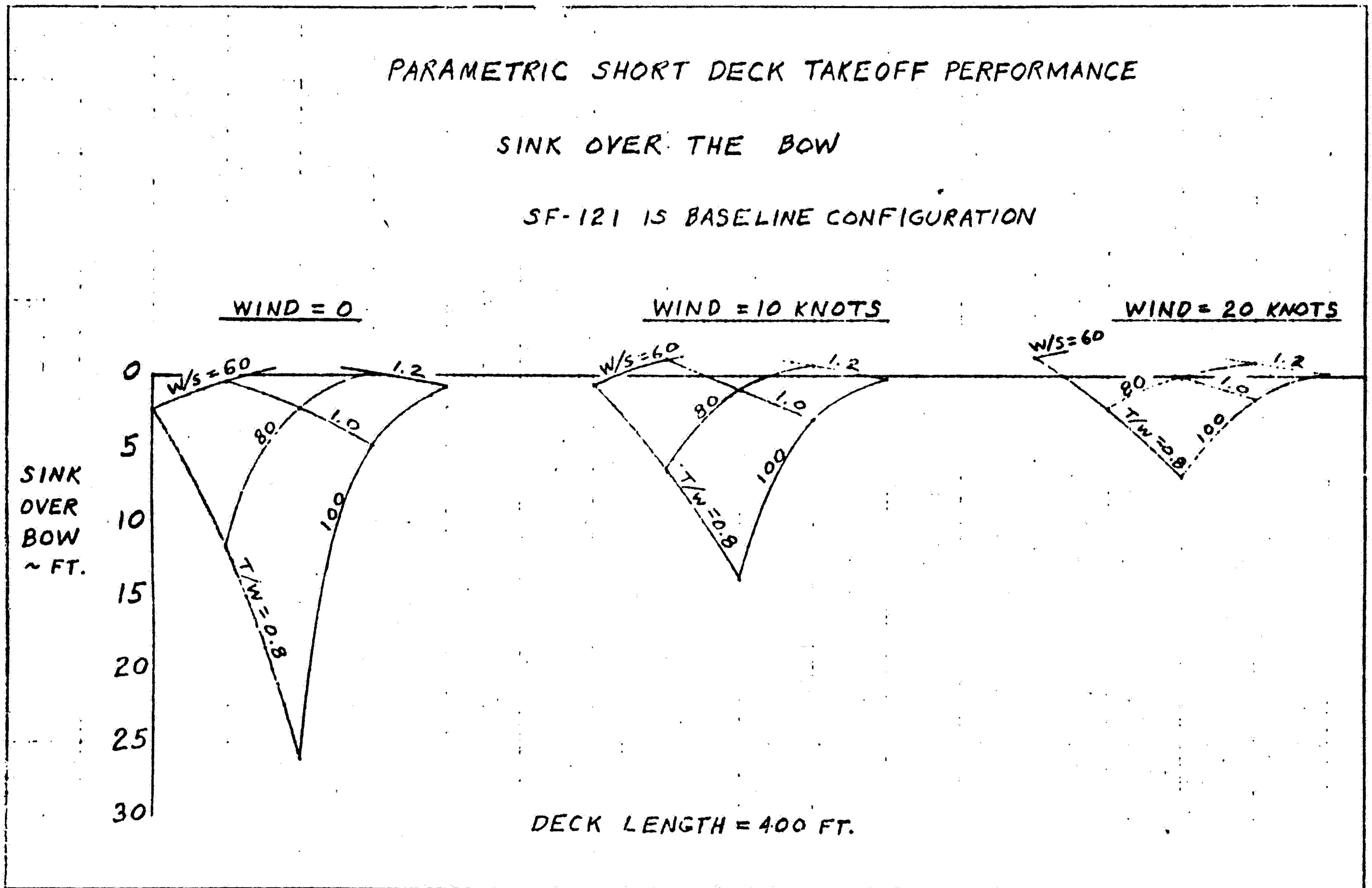


Figure 8-33 - Sink Over Bow Is Limitation on Short Takeoff Weight

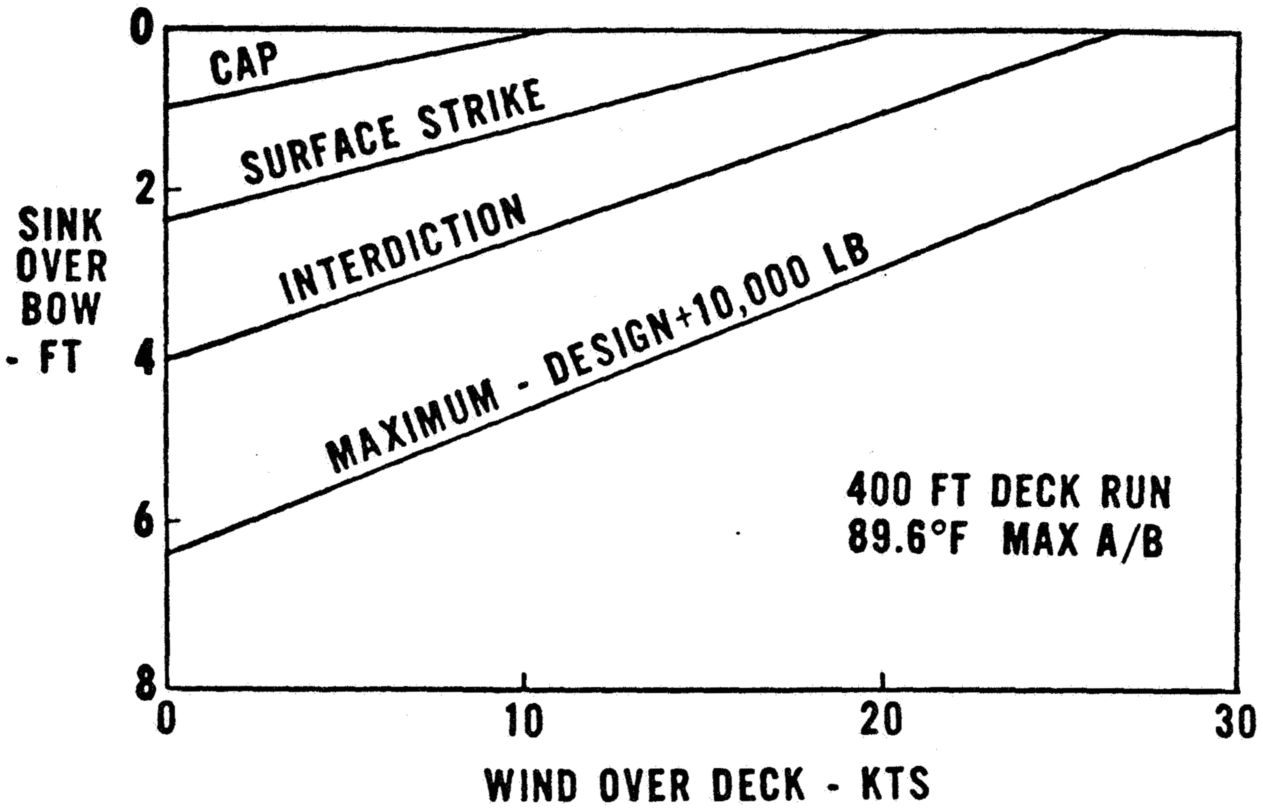


Figure 8-34 - SF-121 Short Takeoff Performance

A least squares regression analysis was performed on the sink over bow results, resulting in the following relation:

$$SINK_{FT} = -9.4 + .037*EW + \frac{17.01 + .063*EW}{3.805 + .034*EW - \frac{(W/S - 46.2)^{0.2458}}{TW}}$$

where the effective wind is

$$EW = (\text{WIND OVER DECK})_{KTS} + 0.1*[(\text{DECK LENGTH})_{FT} - 400]$$

The equivalence of deck length and wind over the deck was established by calculations at three weights with deck lengths of 300, 350, 400, 450, and 500 feet. The regression equation matches all calculated points with 1.2 feet of sink or less except at two points (10,000 lb. overload, 300 and 350 foot deck length, zero wind) where the errors are 5.4 feet out of 32.0 and 2.2 feet out of 20.5, respectively. The equation should not be used for thrust to weight ratios less than 0.8, because between 0.8 and 0.6 the thrust moment becomes insufficient to rotate the aircraft during the deck run.

A brief investigation was made of the curved ramp, or ski jump technique. The effect of a curved ramp was quite dramatic in that it essentially eliminated sink over the bow for the entire range of parameters. For the SF-121 with 10,000 pound overload, 350 foot deck length, and zero wind, the sink was less than one foot. The ramp used was only 5.25 feet high, 100 feet long, and had a deck edge slope of six degrees. Nearly all operational and safety factors are improved. The optimum rotation point is further down the deck such that tail clearance is increased by nearly two feet. The only unfavorable effect relative to a flat deck is a slight (1/3 g) increase in main landing gear load.

8.5 HIGH SPEED THRUST VECTORING

8.5.1 Thrust Vectoring for Maneuvering

The SF-121 VATOL concept offers thrust vectoring in pitch and yaw throughout the flight envelope as a bonus, without additional penalty. Also, it has a canard flap which can be deflected to better exploit TVC. Thrust vectoring effects on specific excess power and sustained and maximum instantaneous load factors have been investigated at the M = 0.6, 10,000 ft. (3,048 M) design condition. The weight used is 20,570 lb. (9,931 kg.), which corresponds to 88 percent of DLI mission takeoff weight. Maximum instantaneous load factor and fuselage aiming control benefited from thrust vectoring. There was no improvement noted for specific excess power or sustained load factor.

These results included the effects of supercirculation and thrust recovery as reported in Reference M1. Data was used at M = 0.7 for a C_T = 0.25 with a nozzle exit at 0.275 exposed root chord aft of the wing trailing edge-fuselage intersection. Lift increments for rectangular exits were increased by 35 percent for axisymmetric nozzles using data in Reference M1. Thrust recovery data were not adjusted. It was assumed that performance for the circular exits could be improved to that for the rectangular exits. The supercirculation and thrust recovery (thrust recovery expressed ad drag) used are:

ΔC_L = 0.0018 per degree δ_T

$\Delta C_{D_{MIN}}$ = -0.00013 per degree $|\delta_T|$

ΔC_{L_B} = 0.00113 per degree $|\delta_T|$

ΔC_M = 0

A canard configuration is well suited to exploit thrust vectoring by use of a canard upload to trim positive (nozzle down) thrust deflection. This benefit is displayed in the power on lift curve of Figure 8-35. For the basic SF-121, thrust deflection used was that which could be trimmed with a maximum 25 degree canard flap deflection. Lift shown for the improved canard was based upon a doubled canard flap effectiveness (this could be obtained with lower canard sweep, increased canard area, or powered systems). If wing trailing edge flap trim is used a small increase in power-on lift is obtained with negative thrust deflections (Figure 8-36). However, these lift benefits did not result in improved maneuver performance in a classical sense. Normal acceleration (n_Z) at a given angle of attack is increased but flight path acceleration (n_X) is decreased (Figure 8-37). This result is also reflected in reduced specific excess power vs. normal acceleration (Figures 8-30 and 8-39). It should be noted that the penalties decrease with increasing n_Z as the thrust deflection approaches its theoretical optimum. Sustained load factor at zero n_X is essentially unaffected with canard trim but is degraded with wing flap trim.

Thrust vector deflection increases direct thrust lift, decreases flight path thrust and for the SF-121 creates a moment which must be trimmed out. A benefit is derived at constant load factor only if the incremental drag from trimming the thrust vector moment plus reduced wing-body induced drag is less than the penalty due to decreased flight path thrust. That is, excess thrust must be increased. This effect was not achieved on the SF-121 because the airplane drag polar had already been optimized to meet the sustained maneuver requirement. A benefit may be shown for a less optimum canard and wing flap combination or via optimization with thrust vectoring included.

Substantial payoff can be shown for thrust vectoring in combat (Figure 8-40). Direct lift control is generated through simultaneous thrust vector and canard trim control deflection. Fuselage aiming control offers

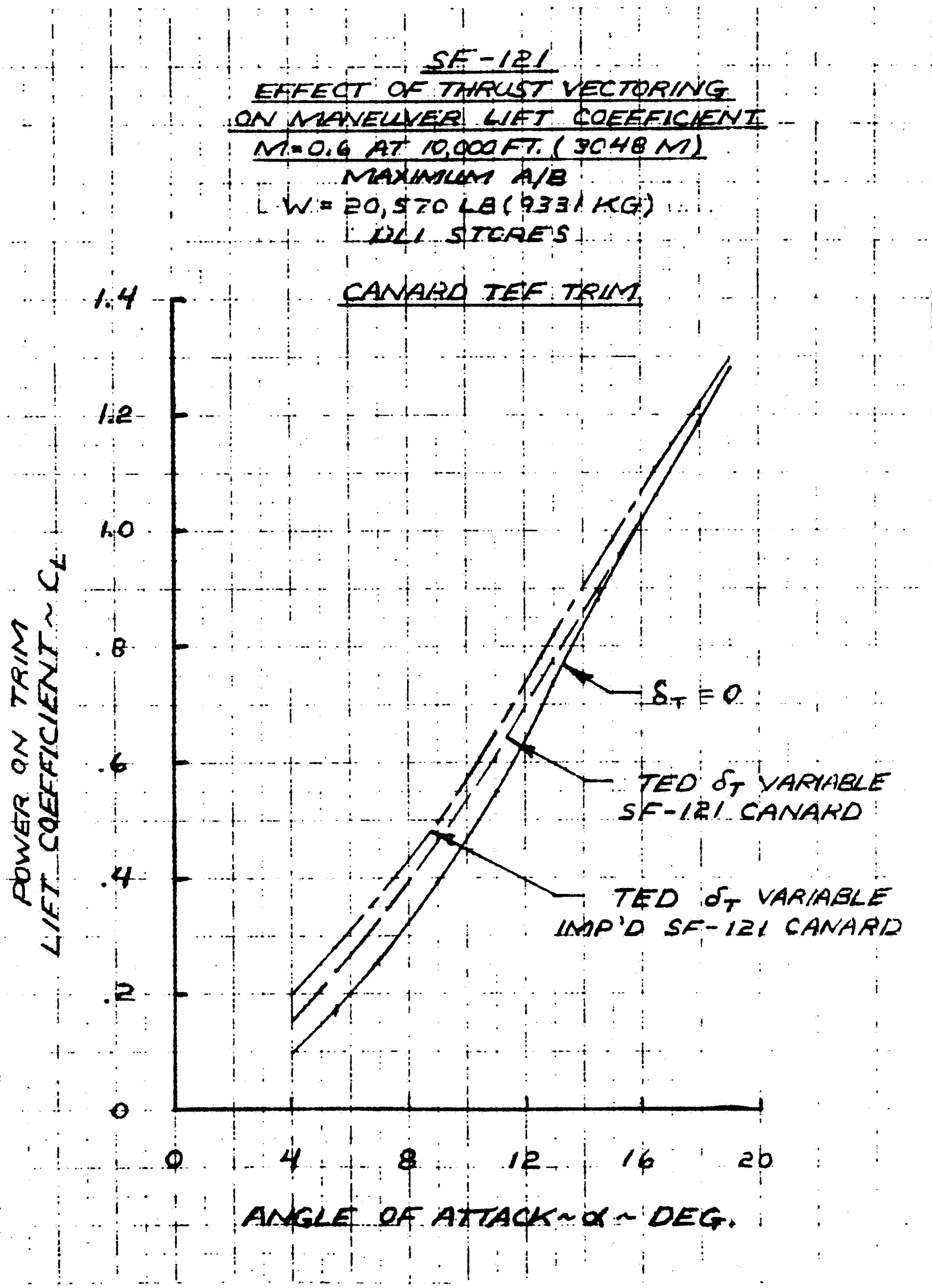


Figure 8-35 - Effect of Thrust Vectoring on Lift-Canard Trim

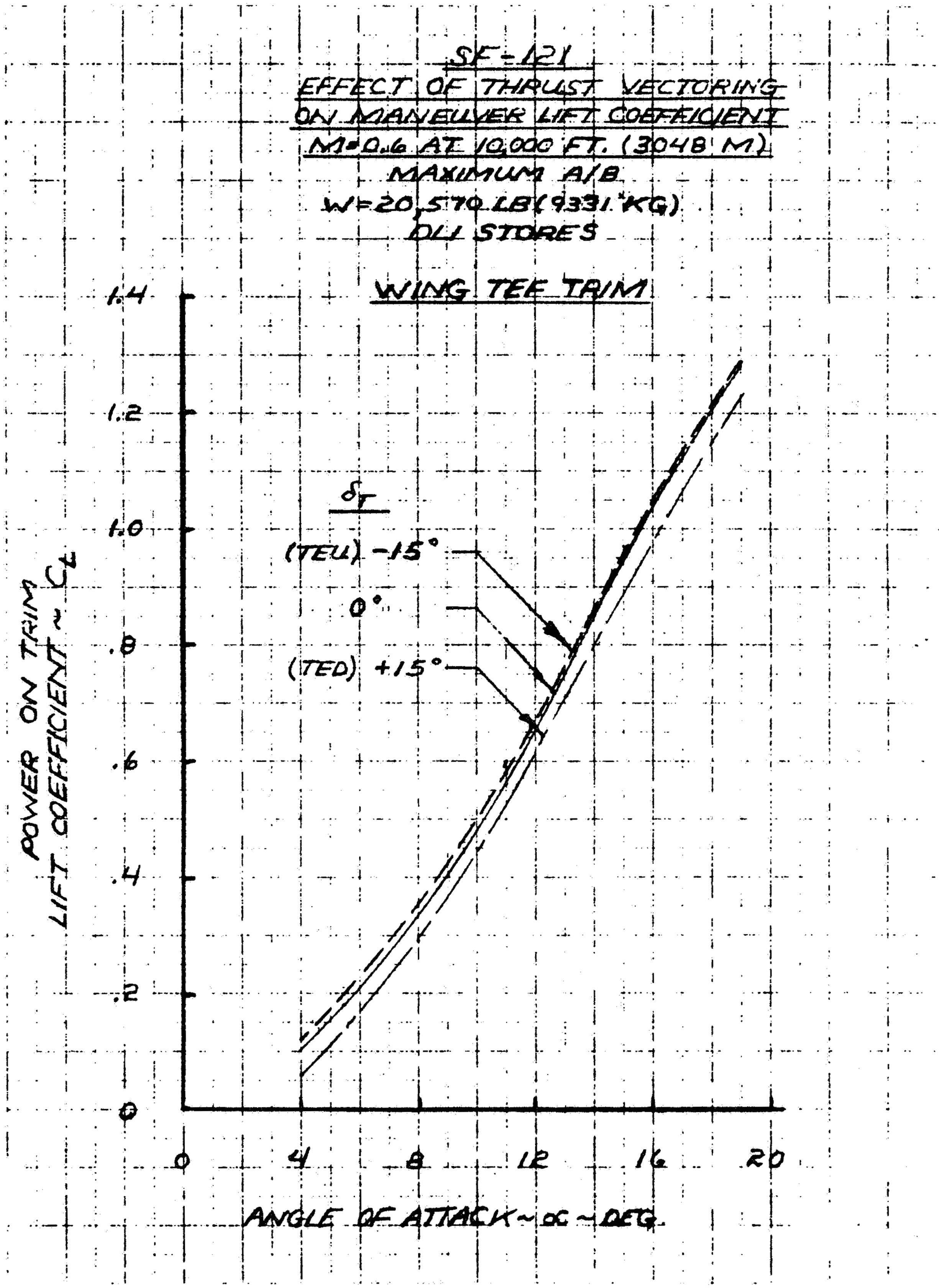


Figure 8-36 - Effect of Thrust Vectoring on Lift - Elevon Trim

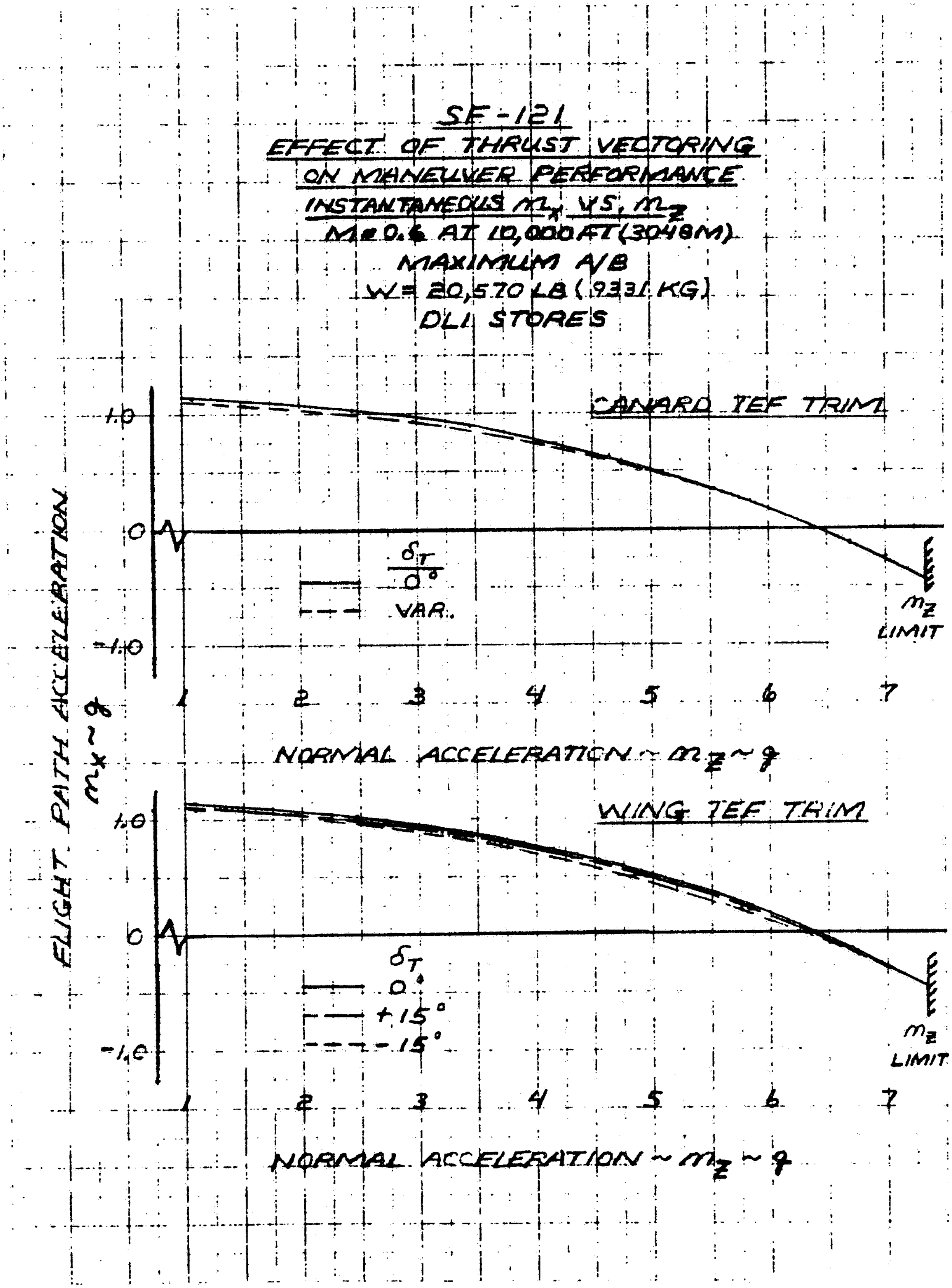


Figure 8-37 - Effect on Thrust Vectoring on Maneuverability

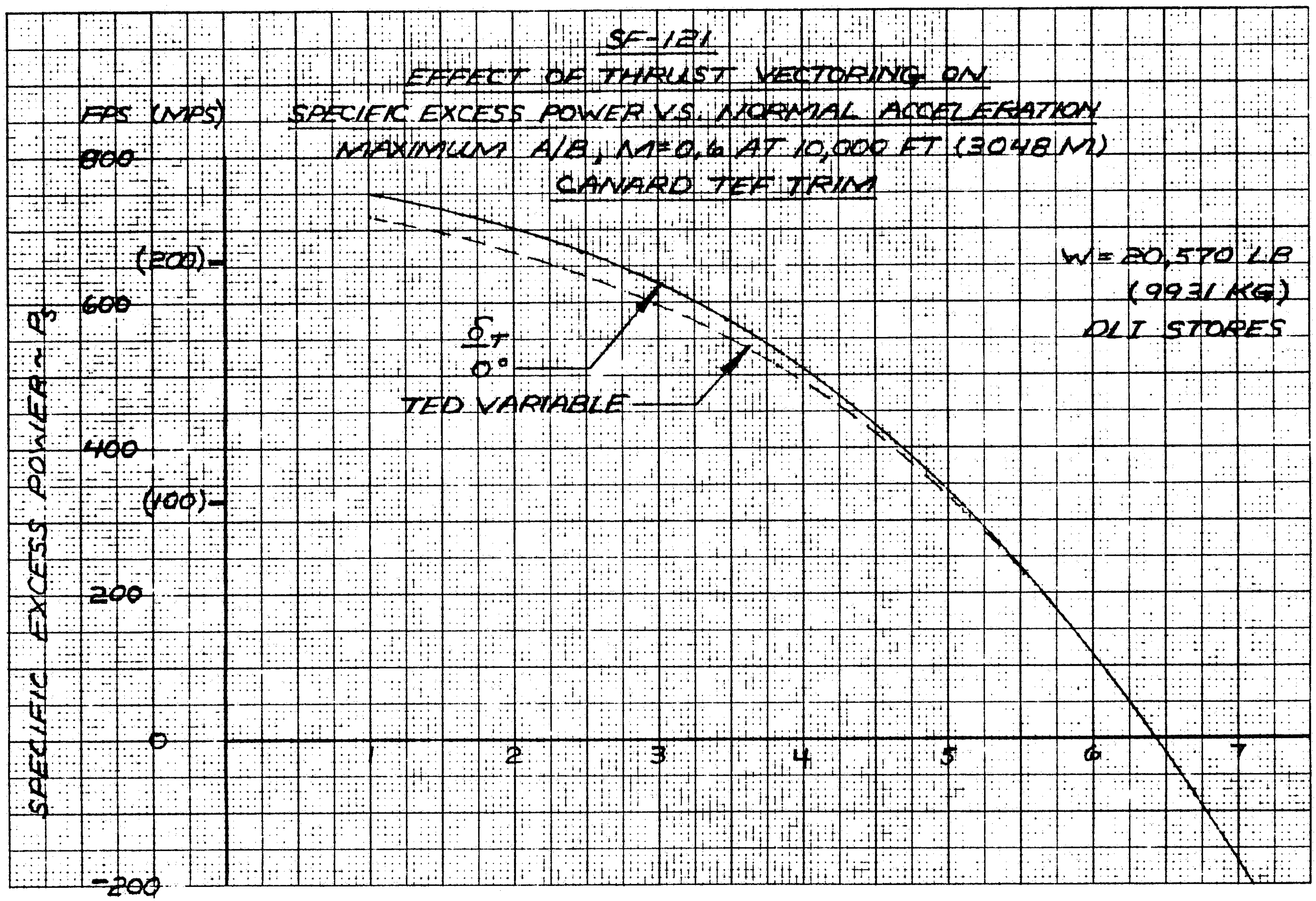


Figure 8-38 - Effect of Thrust Vectoring on Specific Excess Power - Canard Trim

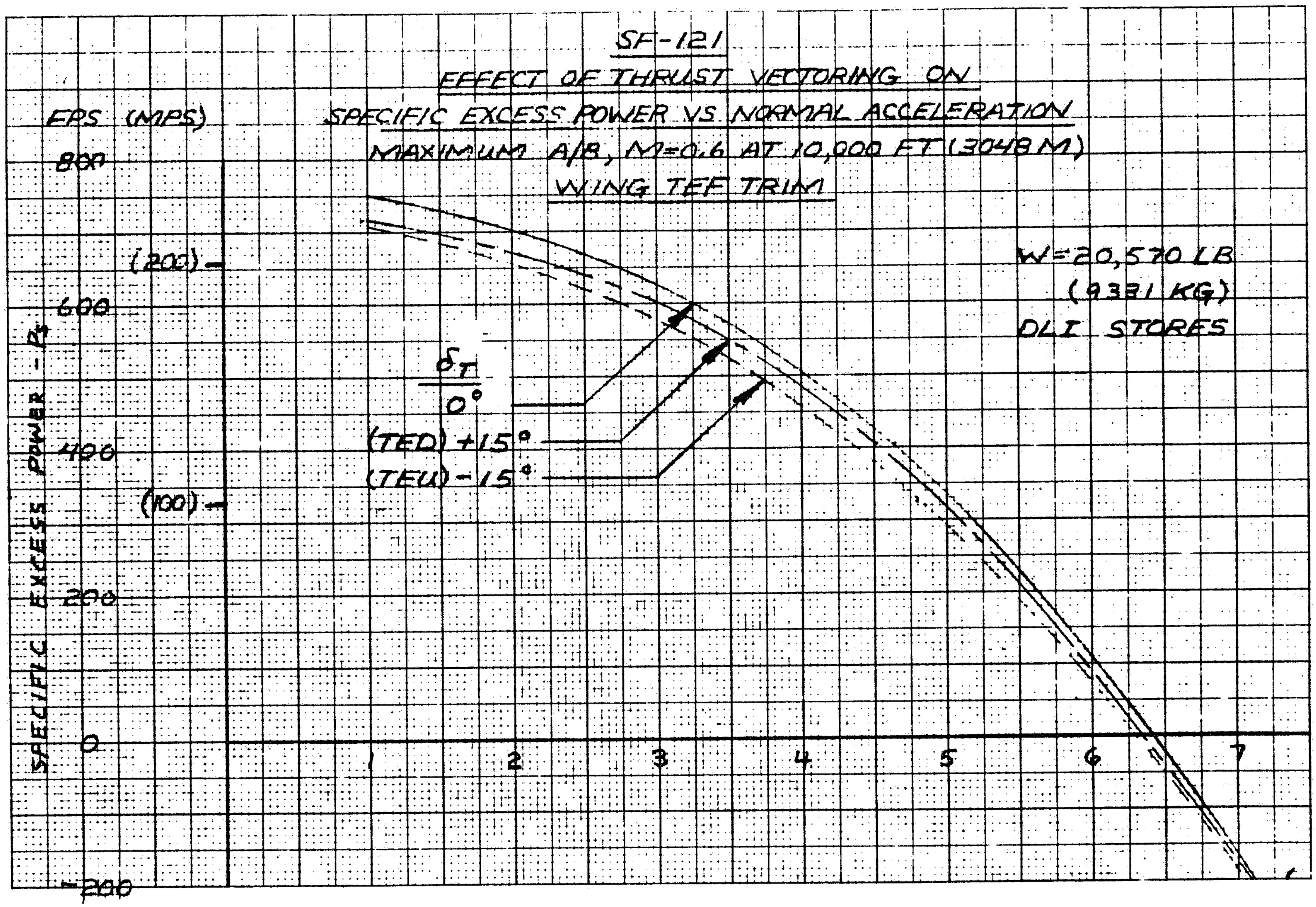


Figure 8-39 - Effect of Thrust Vectoring on Specific Excess Power - Elevon Trim

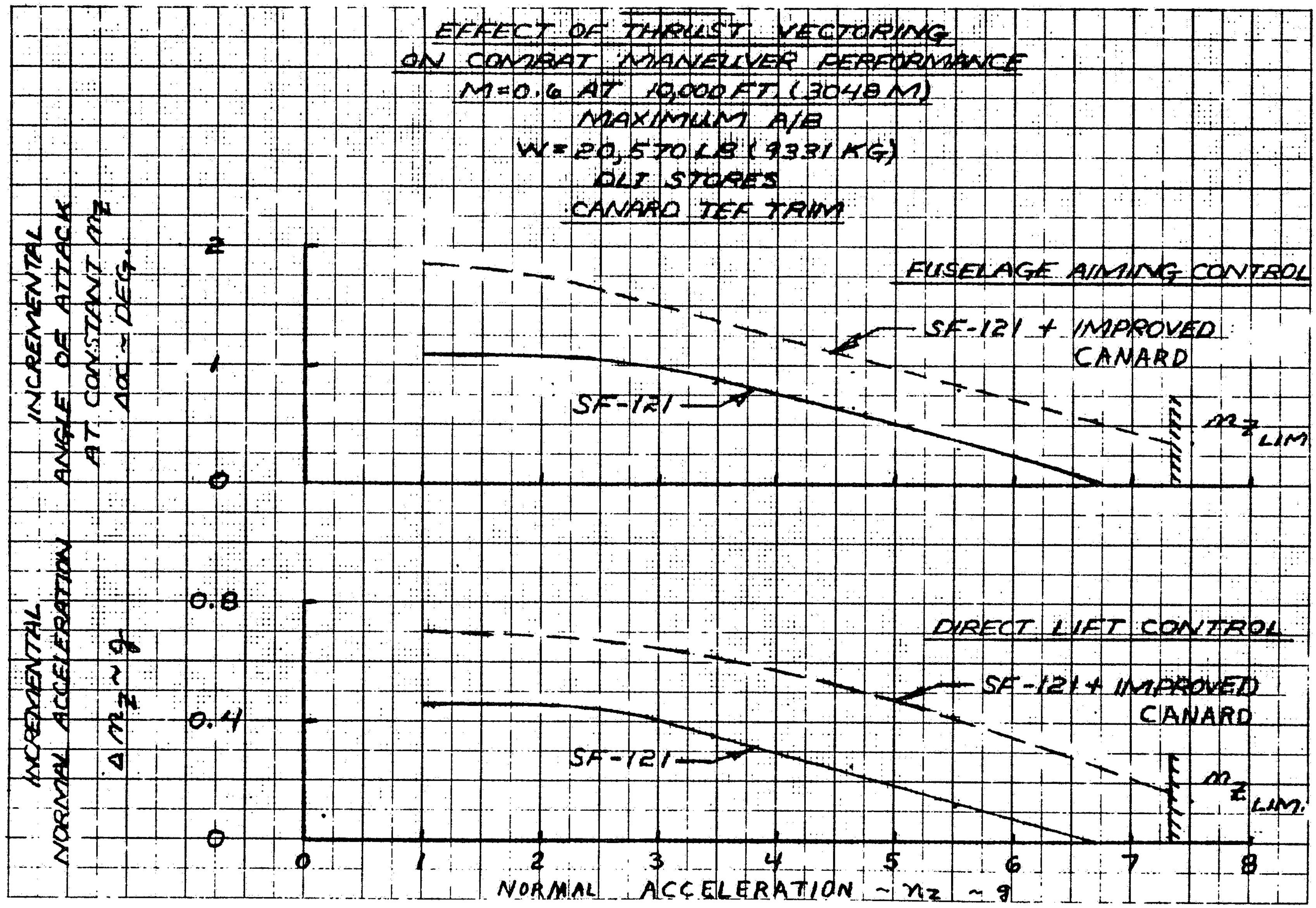


Figure 8-40 - Summary of Effects of Thrust Vectoring on Maneuverability capability to independently change fuselage elevation angle (up or down) for target tracking without change in flight path. This is achieved by trading off wing lift with thrust vector plus canard lift. During air-to-ground gunnery, fuselage aiming provides more time on target and level flight strafing maneuvers can be done with the nose depressed. Air-to-air application provides higher aspect conversion capability with more and longer firing opportunities.

8.5.2 Thrust Vectoring For Supersonic Cruise

A small improvement in cruise drag was shown for thrust vectoring at M = 1.6 at 40,000 ft. (12,192 M). Supercirculation and thrust recovery data used from Reference 6 (at M = 1.2) are:

ΔC_L	0
$\Delta C_{D_{MIN}}$	-0.00028 per degree $\lvert\delta_T\rvert$
ΔC_{L_B}	0.0008 per degree $\lvert\delta_T\rvert$
ΔC_M	0

The optimum thrust vector angle of 2.0 degrees was determined from the express-expression:

$$\frac{\delta_T}{\alpha} = \frac{2\ C_{L\alpha}}{\pi\ A\ e} - 1 \qquad \text{(Reference 14)}$$

Drag coefficient was reduced by 0.0006 with a resultant 1.3 percent increase in specific range. This small effect was due to the 40,000 foot dash altitude being much below optimum.

9.0 AERODYNAMIC UNCERTAINTIES

Most of the technology requirements for the SF-121 are common to advanced fighters in general. The degree to which structural weight, for instance, is reduced will influence the size of the airplane, but is unlikely to determine feasibility. The payoffs for high speed thrust vectoring appear very uncertain, particularly thrust induced effects.

The principal issues center around the VATOL mode of operation. The X-13 demonstrated the basic feasibility of VATOL from a land site over 20 years ago. Valid doubts remain about the operational suitability of VATOL. The major uncertainties are:

- Flying qualities in transition
- Aircraft/ship aerodynamic flow interactions
- Attitude control system power and response
- Inlet recovery and distortion during transition
- Propulsion induced spray
- Pilot visibility and landing aid requirements

The spray question can only be resolved by full scale jet engine tests above water. There are several fundamentally different flow phenomena at work, and reliable scaling of small scale tests is questionable.

The other issues can be effectively addressed by developing a powered model wind tunnel data base to cover VATOL transition boundaries and by manned moving base simulation of transition and docking on a ship.

The land based VATOL landing should not be a major risk, since the rather primitive X-13 accomplished it many times. But the effects of jet blast on ground erosion, foreign object damage and reingestion require considerable attention. Relative impact of these concerns on design development and the need for testing and analysis to resolve them is discussed in the following paragraphs.

9.1 TRANSONIC AND SUPERSONIC AERODYNAMICS

Sizing criteria for the SF-121 described in Section 8.1 are Supersonic Intercept radius, single engine vertical landing thrust to weight and 6.2 g sustained load factor at M = 0.6 10,000 feet (3,048 M) altitude. A breakdown of the DLI mission fuel usage is presented below in Table 9-1 to aid discussion of the effects of design requirements on SF-121 sizing.

Table 9-1 - SF-121 Point Design DLI Mission Breakdown

	Fuel Used lb (kg)	Percent of Total
Take-off	934 (424)	11.5
Subsonic climb to 40,000 ft. (12,192 M)	945 (429)	11.7
Accelerate to M = 1.6	510 (231)	6.3
Cruise to 150 NM (278 km), M = 1.6 @ 40,000 ft. (12,192 M)	2,105 (955)	26.1
Combat - 2.0 min. max. A/B, M = 1.6 @ 40,000 ft. (12,192 M)	1,428 (648)	17.7
BCA - 150 NM (278 km)	614 (278)	7.6
Loiter - 10 min. @ S.L.	386 (175)	4.8
Landing fuel	751 (341)	9.3
Reserve (5 percent total)	404 (183)	5.0
TOTAL	8,077 (3,664)	100.0

Approximately 32 percent of fuel use is directly affected by supersonic drag. Lift coefficient in the acceleration and supersonic dash varies from 0.2 at M = 1.0 to 0.08 at M = 1.6. Thus, the need to reduce supersonic drag due to lift is less than for minimum drag. However, reduced maneuver drag at Mach 0.6 would permit a smaller wing, which would enhance supersonic performance. Reduced minimum drag obtained from wave and nozzle/afterbody drag optimization could permit supersonic dash at M = 1.6 with Intermediate thrust. Reduced bypass ratio combined with optimum wave and nozzle/afterbody dray may yield the desired result. The SF-121 supersonic drag was optimized using the Area Rule method for body and interference wave drag and Stancil's modified linear theory (Reference 5) for airfoil surfaces wave drag. Advanced development of modified linear theory is being done by Vought under Navy contract (Reference 15). Area Rule is notably weak at M < 1.4 for body wave drag optimization. The modified linear theory being developed will not be available for application, however, until late 1979. Its development would be enhanced considerable by having an up-to-date data base and model available to confirm predictions.

Thrust vectoring for supersonic drag and subsonic maneuver improvements were limited by available canard control power and by uncertainty in induced effects. Because the induced effects are highly configuration dependent, powered tests are recommended to evaluate application of thrust vectoring to the SF-121. Small thrust deflections may offer potential nozzle/afterbody drag reductions. The real key to exploiting high speed thrust vectoring, however, is to augment the moment capability of the canard.

Subsonic fuel use is 24 percent of the total. Half of this is used in the subsonic acceleration to climb speed followed by climb at the drag rise Mach number. Improved drag due to lift and drag rise Mach number arising from continued wing optimization would reduce fuel for this segment. These benefits would also spill over into return cruise fuel savings and improved transonic maneuver drag. The Mach 0.6 sustained load factor requirement was satisfied with wing and canard variable camber applied to a thin uncambered wing. It is possible that built-in wing twist and camber with decamber flaps for supersonic cruise would yield better overall performance.

Subsonic and transonic drag due to lift estimates were based on tests of a non-representative coplanar canard-wing geometry (Reference 3). Wing leading edge flap incremental effects obtained from a model without a canard applied to the baseline wing without leading edge devices. Uncertainties arising from these projections result in a need for more representative test data to compare with analytical results. Analytical predictions for the test configuration should be made using the Bailey-Ballhaus or Jameson techniques (References 16 and 17 respectively). Vought is currently working to combine these optimization techniques under NASA contract NAS2-9653.
Questions to be resolved include:

- o Should twist and camber be built-in or introduced through maneuvering flaps?
- o What is the influence of the canard flow field of transonic wing characteristics?
- o Can a canard and wing be optimized simultaneously?
- o Can incremental effects be linearly superposed with reasonable accuracy?

Time and resources are not sufficient to permit a full wing optimization on the Phase II baseline model. However, comparison of the test results with predictions will expedite future wing-canard design optimization.

Vought did extensive wind tunnel development of a CTOL canard fighter which differed from the SF-121 in having a much higher aspect ratio (3·8). Test experience is summarized in Figures 9-1 and 9-2, providing qualitative guidelines for improving directional stability and pitchup. The SF-121

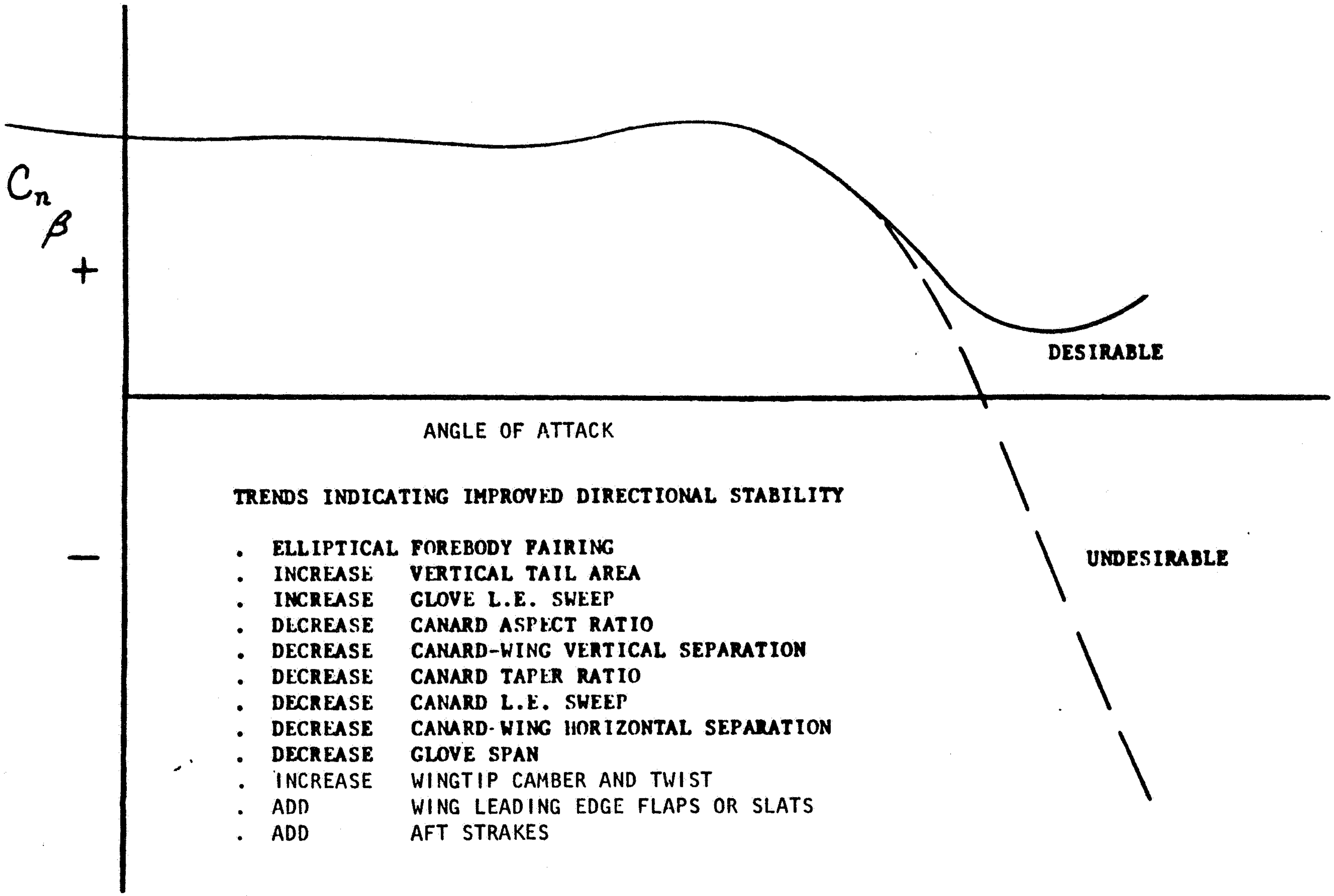


Figure 9-1 - Measures to Improve Directional Stability at High Angle of Attack

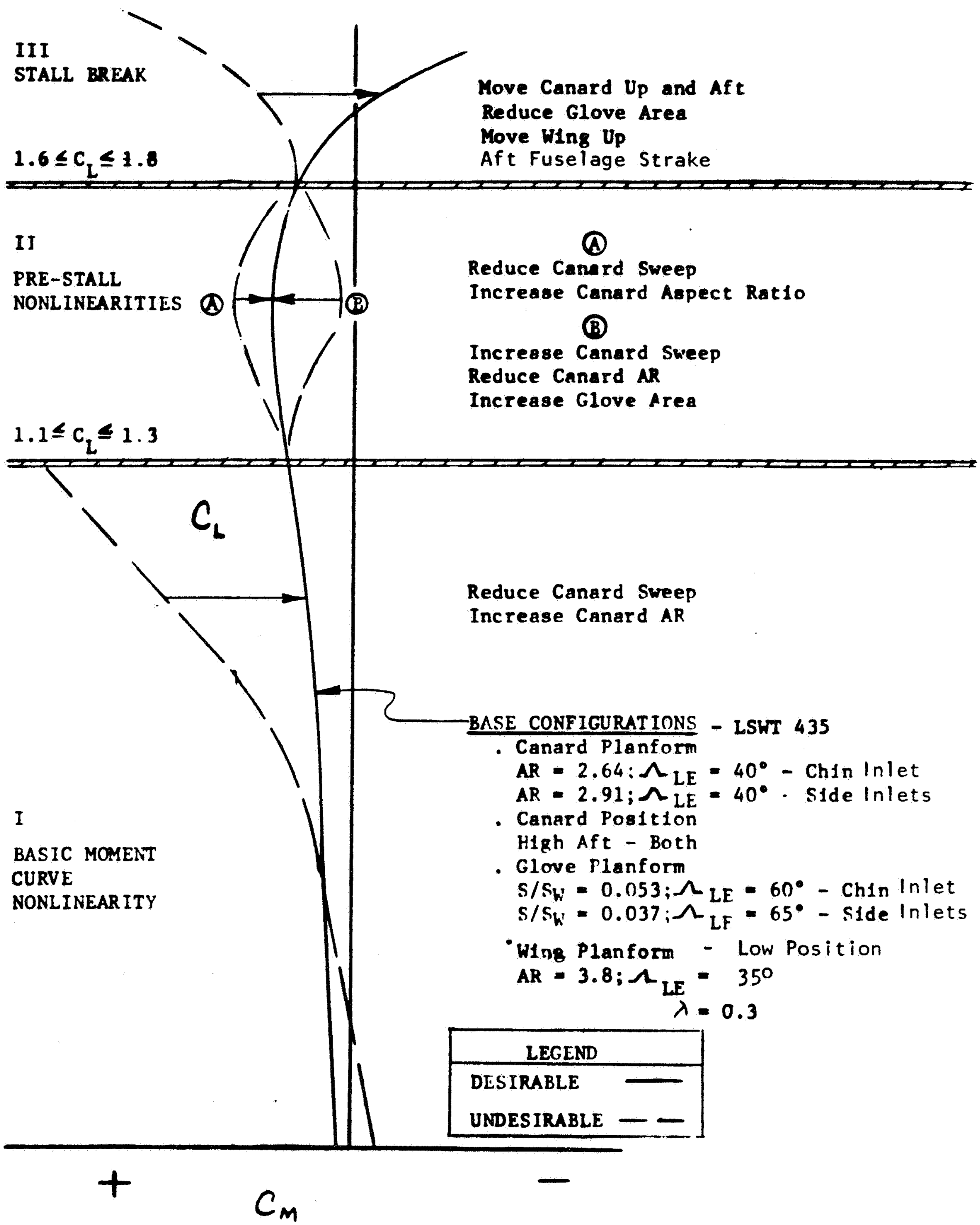


Figure 9-2 - Measures to Improve Pitching Moment Characteristics already incorporates some of these features. Others are addressed in the recommended research program described in Section 10.0.

9.2 BUFFET CHARACTERISTICS

Buffet onset lift coefficient is estimated to be higher than that required to sustain 6.2 g at M = 0.6 at 10,000 ft. (3,048 M). Primary design variables include variable camber on the wing and canard, canard and wing planform, and canard-wing horizontal and vertical spacing. The influence of the canard is complex and difficult to predict. Early canard separation will lower buffet onset C_L, but a canard which is too resistant to stall promotes pitchup. To fully evaluate these effects wing and canard root bending moment strain gauges should be installed.

Buffet is also a concern in transition, particularly during reconversion where extremely high angles of attack will be encountered at low airspeeds. Intensity of this buffet could influence reconversion profile selection and ultimately control phasing requirements.

9.3 TRANSITION AERODYNAMICS

The data base for the transition flight regime is derived from Mach 0.6 data on a configuration which is non-representative in the very features which are paramount to achieving good flying qualities. Data for extremely high angle of attack was developed using trends from a different tailed delta wing configuration. (See Figure 5-2) The aerodynamic analysis described in Section 5.0 and Appendix A was a strenuous effort to account for every significant configuration difference. The resulting aerodynamic coefficients are a composite of many test runs. It is very evident, particularly in the lateral/directional predictions, that powerful and nonlinear flow phenomena are at work. This realization immediately undermines confidence in the (necessary) approach of linear superposition of incremental effects. We were unable to find any quantitative basis to correct for the strong lateral vortex (discussed in Reference 1) which was a major contributor to directional instability in the high wing model (so much that twin vertical tails were destabilizing above 20 degrees angle of attack). The SF-121 was specifically configured to counter such effects (e.g., the wing trailing edge is moved down and aft to shield the fin; the canard is above the wing to energize topside flow).

Tests on an accurately defined model will confirm the effectiveness of the configuration refinements and also determine the validity of estimates by linear superposition.

9.4 INLET AERODYNAMICS

Horizontal ramp external compression inlets were selected for the SF-121 because of their adaptability to a wide range of Mach numbers and angle of attack conditions. The ramps provide a flow turning effect at high angle of attack which reduces distortion. During VATOL transition it is distortion index rather than total pressure recovery which is important since thrust requirements are relatively low. As the hover condition is approached maximum recovery is important.

Our analysis indicated that distortion of the SF-121 inlet flow due to angle of attack is comparable to that on the F-14 inlets. The effects of combined angle of attack and sideslip (both can approach 90 degrees at low velocities) have not been determined. Performance of the downstream inlet could be a problem. The XM2R propulsion simulator could be a valuable tool to implement powered model tests to very high angles.

9.5 PROPULSION INDUCED EFFECTS

The VATOL aircraft is largely free from the propulsion induced effects which plague flat riser VTOL aircraft. Its aft exhaust nozzles and vertical attitude minimize propulsion induced ground effects. There may be a slight effect when the nozzles are used to rotate the aircraft for STO liftoff. Nozzle deflection for control will induce higher flow velocities on the nozzle side opposite the deflection. This would be expected to be favorable, but knowledge of its magnitude is essential for control system development. Powered model tests in crosswinds and in the presence of a simulated ship and platform will be required to assess this problem.

10.0 RESEARCH PROGRAM

This section presents a research program formulated to resolve the aerodynamic uncertainties described in Section 9.0. Recommended aerodynamic analysis methods to be developed have been integrated into a wind tunnel test plan. The result is a total research program which closely related analytical development to a known concise test data base. The analyses are proposed as distinct, parallel programs beyond the scope of the model development contract. Methods will be developed and applied for guidance of subsequent wind tunnel tests. Each test or analytical activity is stated with a list of objectives and test or analysis variables. The objectives relate directly to the uncertainties of Section 9.0, plus relevant data needed to define basic aerodynamic characteristics. Model variables presented show the full range of model parts required. Test variables are shown to illustrate a minimum level needed for evaluation of uncertainties and to define basic aerodynamic design data. Analysis variables are oriented specifically to cover the uncertainties and detailed performance requirements. Before getting into details of the research program we will first describe the proposed Phase II wind tunnel model itself.

10.1 WIND TUNNEL MODEL

10.1.1 Baseline Model Concept

A highly versatile, modular wind tunnel model is proposed to implement the test program described in Section 10.2. The model scale and construction concept assure compatibility with two XM2R compact propulsion simulators. The XM2R is a small axial flow compressor driven by high pressure air, and is capable of simulating a wide range of engine operating conditions. It may prove even more valuable for low speed VSTOL transition testing than in its intended high speed flight mode. The model will initially be tested in a flow-through mode.

Figure 10-1 shows the modular construction proposed for Phase II. The heart of the model is a steel box structure which can house an internal strain gage balance when sting mounted. Alternatively, a top or bottom mounted blade can support the model and supply compressed air to the XM2R simulators when installed. All the other model parts attach to the central core without interfering with the balance or inlet ducts. The wings and empannage are attached by steel tangs to permit various mounting locations.

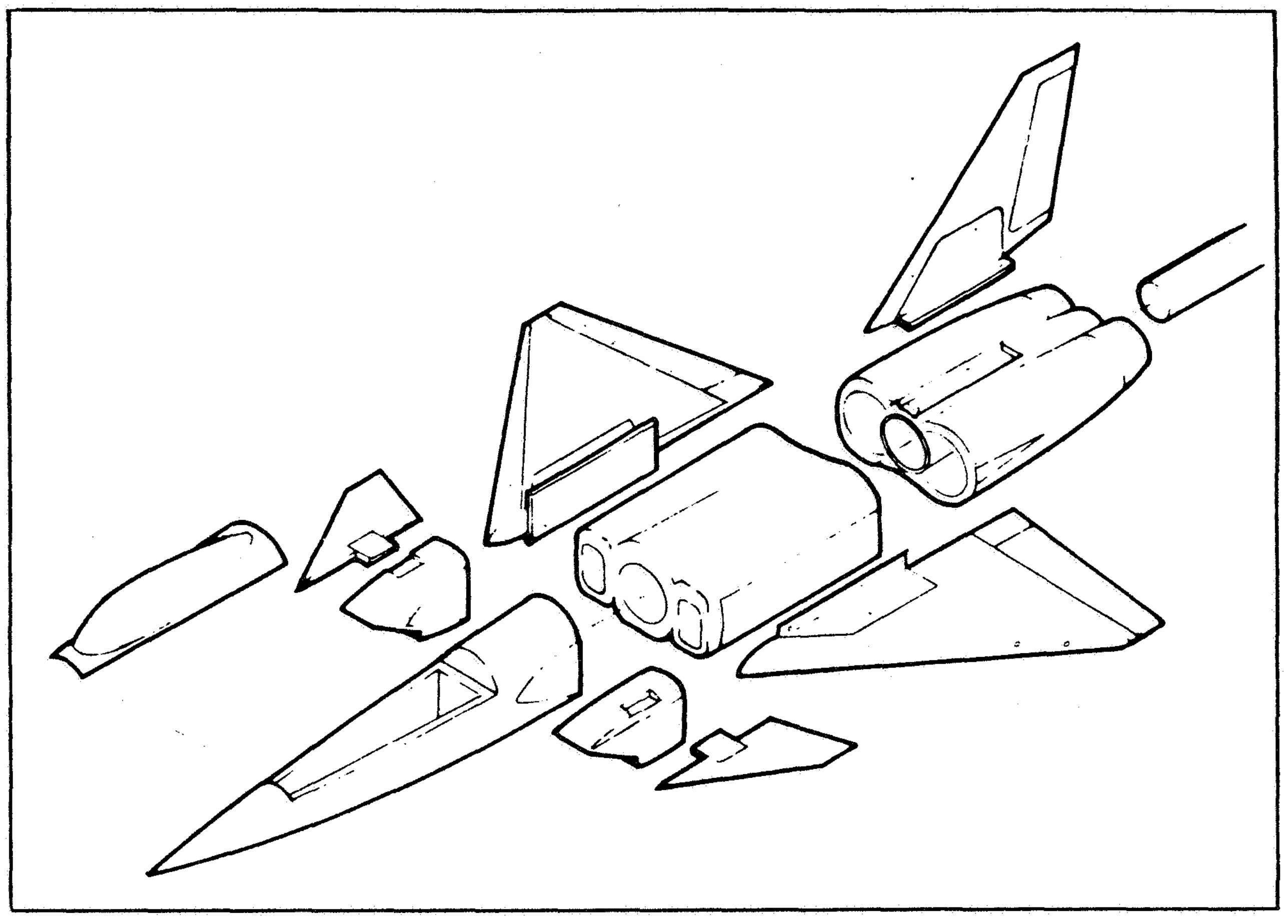


Figure 10-1 - VATOL Wind Tunnel Model Concept

Figure 10-2 reveals additional details of how the model is constructed. The drawing is of the SF-120 proposal configuration (wing area = 330 ft^2 full scale). The SF-121 model will be very similar except for having a larger wing. At 0.10 scale, model length and span will be 4.53 feet and 2.85 feet (1.78 m and 1.12 m), respectively. It will be suitable for testing up to Mach 2.4 in the Ames 9 by 7 foot supersonic tunnel, but is also large enough to provide valuable low speed data at high angles of attack and sideslip. It will be compatible with the NASA Ames 11 foot transonic and 12 foot pressure tunnels. The Proposal, Reference 14, describes the model construction and design in more detail.

Vought has tested a number of aerodynamic devices to improve high angle of attack characteristics. It is expected that the developed SF-121 configuration will incorporate some of them. Since the completed study

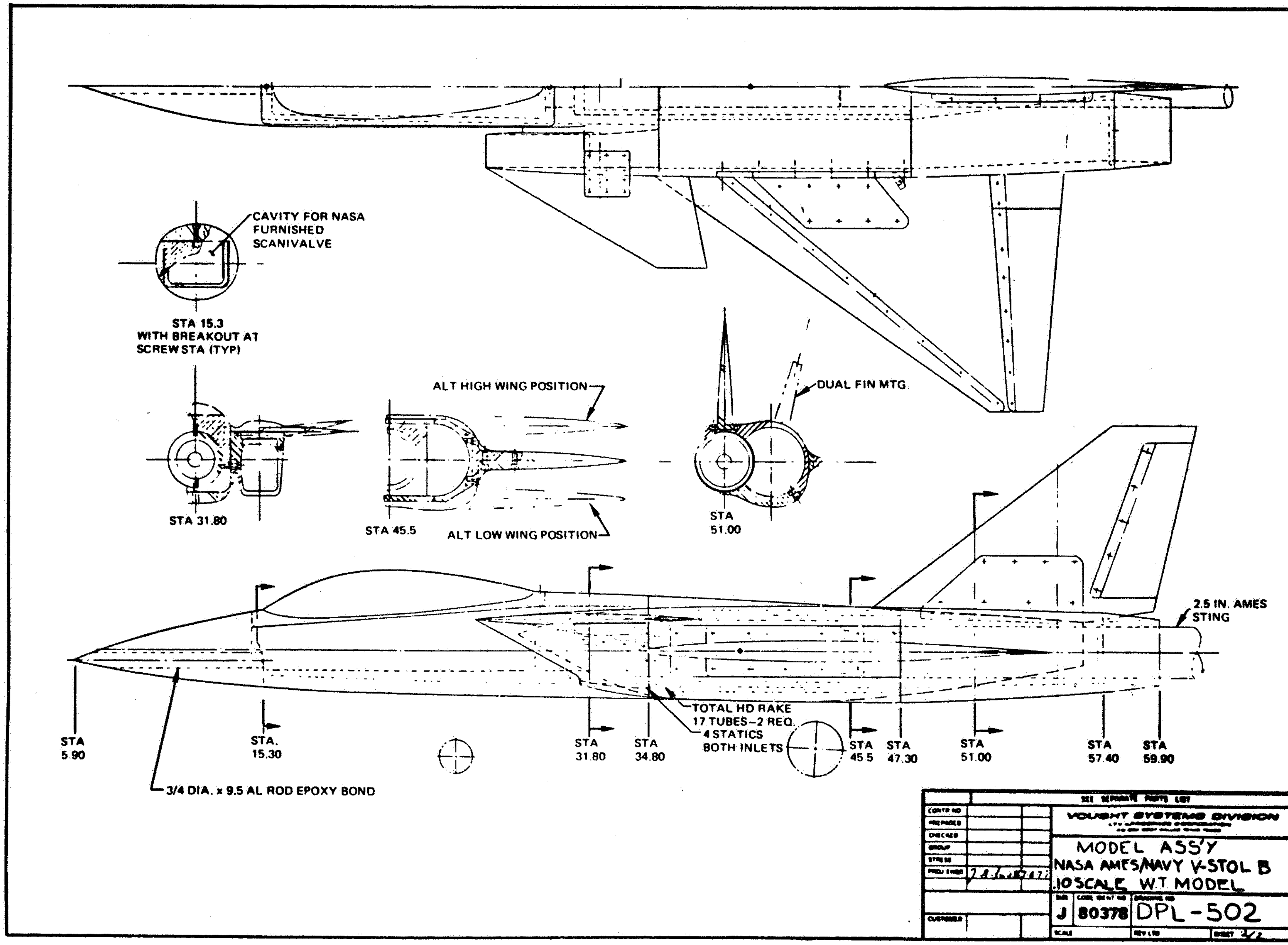


Figure 10-2 - VATOL Model Assembly

indicates the need for aerodynamic refinements, it may be preferable to apply them to the first test configuration. Changes which should prove beneficial include:

- a more elliptical nose
- nose strakes
- reduced canard sweep
- increased canard aspect ratio
- canard camber
- reduced vertical tail sweep
- ventral fins
- wing twist (washout)
- increased leading edge flap tip chord
- a small wing glove

10.1.2 Model Growth Options

Both the SF-121 design and the wind tunnel model just described are intended to be baselines from which more highly optimized variants will evolve. The research program defined in Section 10.2 will require numerous hardware variations to complete. The Phase II effort should begin with a study to ensure that the model is compatible with anticipated variations.

Examples of model options are:

- other wing airfoils and/or planforms
- powered lift wing and/or canard concepts
- other wing leading edge contours or deflections
- other elevon and speedbrake areas and/or deflections
- spacers to vary fuselage or inlet length
- added variations in canard dihedral and incidence or replacement with other canard geometry
- provision for other single or twin vertical tail locations
- nozzle or afterbody changes, such as 2-D nozzles
- removable canopy to enable change in canopy shape and blending to body
- tilting nose section/cockpit

The baseline model is a rear sting supported, flow-through model designed to permit future conversion to a blade supported powered model using two XM2R compact simulators. A conceptual arrangement of the blade mounted system is shown in Figure 10-3. Note that both the flow-through and simulator powered configurations may be tested with the blade mount. Test with the blade mounted flow-through model will help correlate the data between initial flow-through and future powered tests. A dummy rear sting can be used with the blade mount to determine sting effects. The blade is located approximately

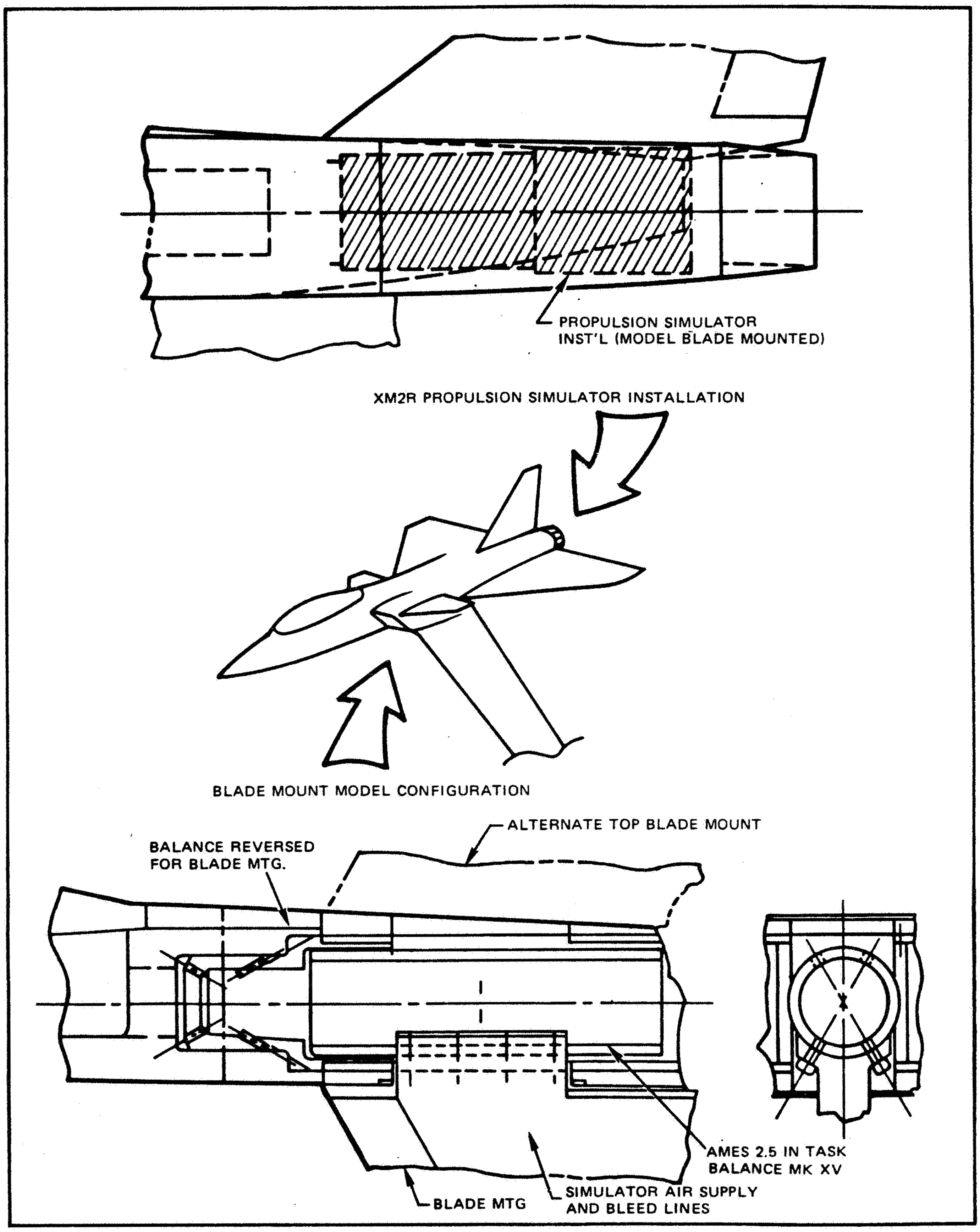


Figure 10-3 - Propulsion Simulator Installation and Support at mid-fuselage to avoid interference with the inlets and yet have acceptably low interference with the afterbody during the powered tests. The blade will be located on the bottom of the model for low to moderate angles of attack, and on the top of the model for very high angles of attack to minimize interference effects. The blade will attach to a sting that in turn is supported by the tunnel pitch mechanism. Such a mounting arrangement will permit angles of attack around 90 degrees with the normal tunnel pitch system and will thus avoid any of the flow angularity associated with the existing special high angle pitch mechanism used with the rear sting mount.

The foregoing discussion has been in the context of a VATOL fighter. It should be noted that the SF-121 is really a highly maneuverable CTOL configuration with thrust vectoring. The aerodynamic configuration is not compromised to achieve VSTOL capability. It is also likely that continued development to improve extreme angle of attack characteristics required for VATOL transition will carry over to enhanced combat agility.

10.1.3 Flat Riser Variants

Vought has paralleled its VATOL research with VSTOL fighter design studies using other propulsion systems. One of the most attractive is a lift plus lift/cruise variant of the canard superfly configuration, as sketched in Figure 10-4.

The flat riser differs from the SF-121 in only two essentials:

- o The axisymmetric gimballed nozzles are replaced by two-dimensional 90 degree vectoring nozzles.
- o The forward fuselage houses one or more lift engines.

With this propulsion arrangement the lift engine(s) must support approximately half of the aircraft weight. If the lift/cruise engines are shifted forward lift engine size may be reduced, but at the expense of a compromised low drag configuration. The extra lift engine weight (about 150 pounds) is a smaller penalty than the sum of:

- o higher wetted area
- o higher wave drag
- o scrubbing drag of lift/cruise exhaust on airframe
- o airframe heating by exhaust

which characterize the forward biased lift/cruise engine layout. The same

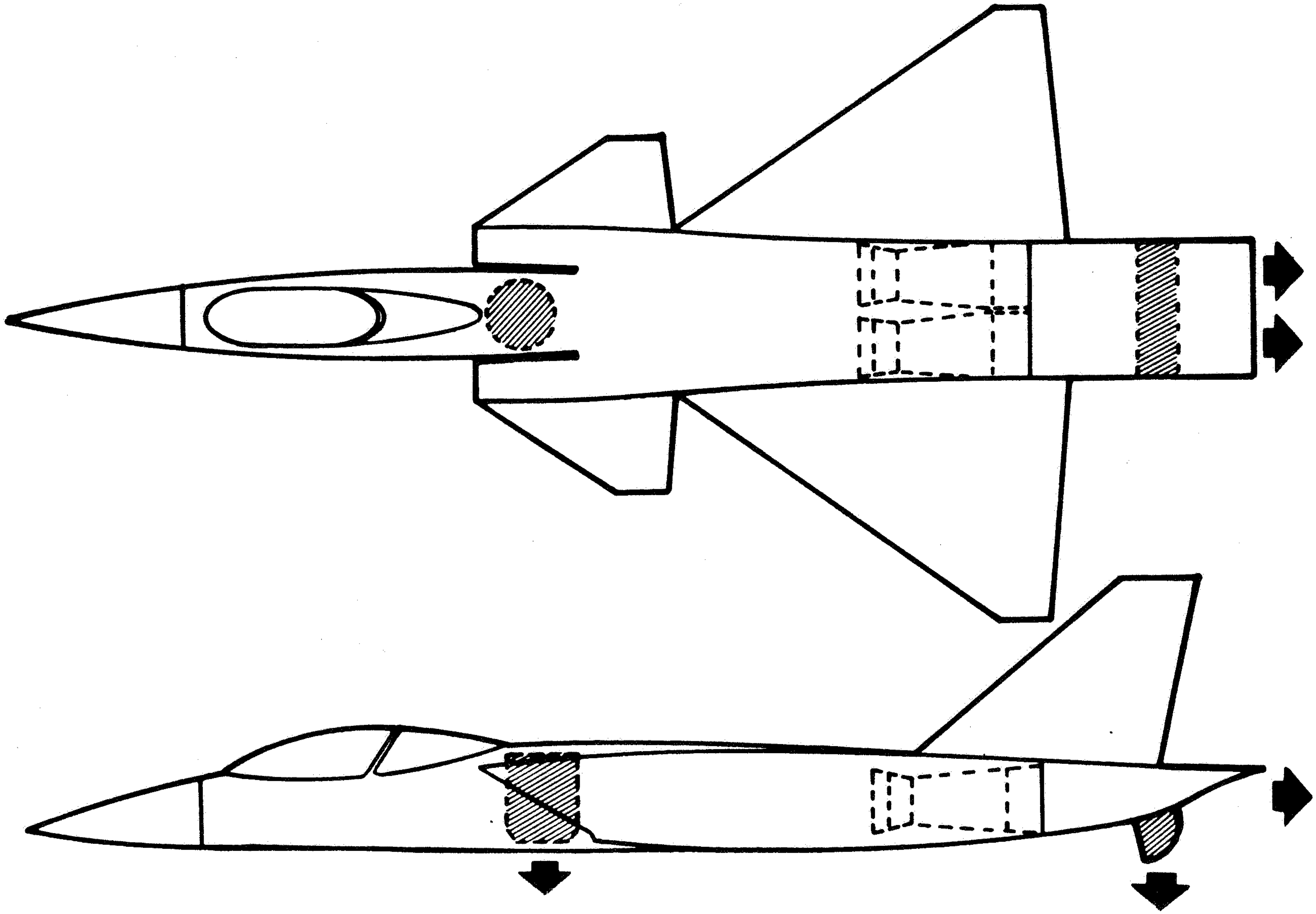


Figure 10-4 - Flat Riser Superfly Variant considerations apply to the Remote Augmentor Lift System (RALS), in which remote burners replace the lift engines.

The model concept in Figure 10-1 provides for lift engine simulation by mounting air ejector lift engine simulators in the forward fuselage section. (The XM2R is too long to fit in this application.) This could also be achieved by inserting a fuselage plug containing the ejectors between the nose section and the central core. The lift/cruise vectoring nozzles, of cruise, are easily implemented in a new aft fuselage fairing. This same aft fuselage would then be available for 2-D nozzles for VATOL or CTOL applications.

Flat riser configurations tests are not included in the VATOL plan in the next section, but are an obvious and straightforward extension of the powered model phase.

10.2 WIND TUNNEL TEST PROGRAM

Initial test plans and model requirements for the unpowered model are presented as items (1), (4) and (7) in Table 10-1. Each test is quite comprehensive and will require more than one entry to complete all objectives. An overlap of test Mach numbers is suggested for the transonic and subsonic tests to assure compatibility of data from the 11-foot and 12-foot tunnels. However, their ranges are close enough to permit interpolation of results between them. These tests are aimed at definition of the basic configuration aerodynamics and problem solving.

As discussed in the Proposal (Reference 14), the recommended blockage limit for the 12-foot tunnel will be exceeded as angle of attack approaches 90 degrees. If the blockage is unacceptable, use of the 14-foot tunnel is suggested for high angle of attack tests. The 40 x 80 foot tunnel may warrant consideration for later XM2R powered transition tests. The large test section will make possible investigation of far field recirculation and ship turbulence phenomena.

Later test plans and model requirements for the XM2R simulator powered model are presented as items (8), (9) and (10) in Table 10-1. The same overlap and blockage considerations apply. New model components will be needed to evaluate design refinements emanating from analytical studies (See Section 10.3) and analysis of results from the first three tests. These will

Table 10-1 - SF-121 Research Program

ACTIVITY	OBJECTIVES	VARIABLES
(1) Transonic Wind Tunnel Test (ARC 11-foot)	o Eliminate pitchup at high angle of attack. o Eliminate or minimize unstable $C_{n\beta}$ at high angle of attack. o Evaluate variable camber for C_{D_L}, $C_{L_{BO}}$, and $C_{L_{max}}$. o Determine drag rise o Evaluate longitudinal, directional, and lateral stability and control. o Determine transonic wave drag. o Determine store drag.	Model o Wing - position, LEF δ, TEF δ o Canard-planform, LEF δ, TEF δ. incidence, position o Inlet-shape, length, MFR o Nozzle-convergence, MFR o Fuselage-nose length and shape o Vertical tail-single ℄, rudder δ, twin o Stores-DLI design mission o Strakes-nose, aft Test o M = 0.6, 0.8, 0.9, 0.95, 1.2, 1.4 o α = 0 to 12 degrees o $\beta = \pm 4$ degrees o M variable 0.6 to 1.4 @ α = 0 degrees.
(2)* Analytical evaluation of transonic variable camber with the Jameson or Bailey-balhaus techniques	o Validate method for prediction of canard-wing variable camber aerodynamics	o Wing - LEF δ, TEF δ o Canard - LEF δ, TEF δ, incidence o M = 0.6, 0.8, 0.9, 0.95 o α = 0 to 12 degrees
(3)* Optimize variable camber with methods from (2) plus Vanderplaats technique	o Minimize cruise C_{D_L} o Optimize maneuver $C_{L_{BO}}$ and C_{D_L} o Define variable camber chord and deflection	o Wing - LE and TEF δ and planform o Canard - LE and TEF δ and planform, incidence o M = 0.6 and 0.8 o α as required

* Analytical efforts proposed to be done as contract effort in support of test activity.

Table 10-1 - SF-121 Research Program

(Continued)

ACTIVITY	OBJECTIVES	VARIABLES
(4) Supersonic Wind Tunnel Test (ARC 9 foot by 7 foot)	o Determine supersonic wave drag. o Evaluate variable camber for minimum C_D. o Evaluate longitudinal, lateral and directional stability and control. o Increase C_{n_β} at $M > 2.0$ o Determine store drag increments.	Model o Wing - LEF δ, TEF δ, position o Canard - LEF δ, TEF δ, incidence, position. o Inlet - shape, length, MFR o Nozzle - convergence, MFR o Fuselage - nose length o Vertical tail-single ℄, rudder δ, twin o Stores - DLI design mission Test o $M = 1.6, 2.0, 2.4$ o $\alpha = -4$ to 20 degrees o $\beta = \pm 4$ degrees
(5)* Analytical evaluation of model wave drag with Stancil modified linear theory	o Validate prediction of total configuration wave drag.	o Selected configurations o $M = 1.2, 1.4, 1.6$ o $\alpha = 0$ and 4 degrees
(6)* Optimize SF-121 fuselage-wing-canard with modified linear theory.	o Minimize acceleration and supersonic dash wave drag	o DLI design configuration o $M = 1.2, 1.4, 1.6$

* Analytical efforts proposed to be done as contract effort in support of test activity.

Table 10-1 - SF-121 Research Program

(Continued)

ACTIVITY	OBJECTIVES	VARIABLES
(7) Subsonic Wind Tunnel Test (ARC 12-foot) NOTE: 14-foot or 40 x 80 or LRC 15 x 20 may also be considered for extreme angles of attack not attainable in 12-foot due to blockage.	o Evaluate variable camber for low speed (pre-transition) $C_{L_{MAX}}$ o Compare maneuver $C_{L_{BO}}$ and C_{D_L} with 11-foot results. o Compare cruise C_{D_L} with 11-foot results o Evaluate longitudinal, lateral and directional stability and control in transition. o Appraise Reynolds number e Effects on C_{D_L} and $C_{L_{BO}}$.	<u>Model</u> o Wing - LEF δ, TEF δ o Canard - LEF δ, TEF δ, incidence o Nozzle - convergence, MFR o Vertical tail (s) - rudder δ o Strakes - nose, aft o Stores - DLI mission, wing tanks <u>Test</u> o M = 0.2, 0.6, 0.8 o α = -4 to 100 degrees @ M = 0.2 -4 to 36 degrees @ M = 0.6, 0.8 o $\beta = \pm 15$ degrees o $\phi = \overline{0}$ to 180 degrees @ α = 90 degrees, M = 0.2 o Reynolds number
(8) Transonic Powered Model Wind Tunnel Test with XM2R Propulsion Simulators (ARC 11-foot)	o Evaluate thrust effects on C_{D_L}, $C_{D_{MIN}}$, $C_{L_{BO}}$, $C_{L_{MAX}}$. o Evaluate optimized canard and wing variable camber trim C_{D_L} and $C_{L_{BO}}$. o Evaluate stability and control with optimized variable camber and thrust effects	<u>Model</u> o Wing - Optimum and basic LEF and TEF δ and planforms. o Canard - Optimum and basic LEF and TEF δ and planforms, incidence. o Inlet - lips, throat, bleed, MFR

Table 10-1 - SF-121 Research Program

(Continued)

ACTIVITY	OBJECTIVES	VARIABLES
(8) Continued	o Evaluate afterbody drag improvement	o Nozzle - Convergence, NPR, deflection o Fuselage - afterbody contours o Vertical tail(s) - rudder δ o Stores - DLI mission <u>Test</u> o M = 0.6, 0.8, 0.9, 0.95, 1.2, 1.4 o α = -4 degrees to maximum o β = ± 4 degrees
(9) Supersonic Powered Model Wind Tunnel Test (ARC 9 foot by 7 foot)	o Evaluate thrust effects on $C_{D_{MIN}}$, C_{D_L}, and C_L. o Evaluate optimized variable camber for minimum trimmed C_D. o Appraise wave drag and afterbody drag improvements.	<u>Model</u> o Wing - optimized LEF and TEF δ o Canard - Optimized LEF and TEF δ o Inlet - throat, MFR o Nozzle - convergence, NPR, deflection o Fuselage - contour o Vertical tail(s) - rudder δ o Stores - DLI mission <u>Test</u> o M = 1.6, 2.0, 2.4 o α = -4 to 20 degrees o β = ± 4 degrees

Table 10-1 - SF-121 Research Program

(Continued)

ACTIVITY	OBJECTIVES	VARIABLES
(10) Low Speed Powered Model Wind Tunnel Test (ARC 12 foot) See Activity (7) note.	o Evaluate optimum variable camber for low speed C_{LMAX} o Compare maneuver C_{LBO} and C_{LMAX} with 11-foot results. o Evaluate inlet performance in transition. o Appraise thrust effects on C_{D_L}, $C_{D_{MIN}}$, C_{LBO}, C_{LMAX} and control in cruise, maneuvers and transition. o Determine effects of TVC and reaction jets in transition on induced forces, moments and water spray. o Evaluate thrust deflection effects in presence of ground for STO configuration.	Model o Wing - optimized LEF and TEF δ o Canard - optimized LEF and TEF δ, incidence o Nozzle - exit area, NPR, δ o Vertical tail(s) - Rudder δ o Reaction jet - MFR o Inlet - MFR, lip shape, cowl flaps, bleed Test o α = -4 to 36 degrees @ M = 0.2, 0.6, 0.8 o α = -4 to 130 degrees @ V_0/V_J = 0.1, 0.2 o In and out of ground effect
(11)*Analytical evaluation of propulsion induced effects in transition and STO using Hess plus jet math model technique.	o Validate prediciton methodology for extreme angles of attack in free air and small angles of attack in ground effect.	o Transition configurations o STO configuration o V_0/V_J = 0, 0.1, 0.2 o Free air and in ground effect.

* Analytical efforts proposed to be done as contract effort in support of test activity.

include leading and trailing edge flaps, strakes and external fairings. Roll reaction jets will also be simulated. Their air supply will be routed through the leading or trailing edge flap attach stations. This model design approach will also facilitate tests of leading and trailing edge boundary layer control. The main wing beam will be designed to allow maximum flexibility for leading and trailing edge flap variations. These tests will also assess direct thrust and jet induced effects.

10.3 METHODS DEVELOPMENT

While the Phase II contracted effort is concerned exclusively with design and fabrication of the baseline wind tunnel model, a discussion of appropriate paralleling research is in order. Vought has been active in analytical aerodynamics and aeropropulsion methods development. Three current programs relevant to SF-121 configuration research will be described in the following paragraphs.

10.3.1 Supersonic Modified Linear Theory

Recent service design studies (USAF ATF and USN NFA) have stressed the need for design of efficient supersonic cruise and dash aircraft. Selection of the "best" configuration during preliminary design of a new military aircraft or missile depends more on the accuracy of predicted trend, or incremental, data than on the absolute accuracy of the data. Current wave drag prediction techniques are based primarily on the supersonic area rule. The slender body assumption inherent in the area rule is often violated in one or more local areas on a fighter or missile configuration; while overall drag prediction may still be fairly accurate, incremental predictions are often unreliable. Improved analytical methods are needed for moderate to low fineness ration configurations typical of fighters and missiles.

Similarly, all present methods of analyzing or designing supersonic camber and of analyzing supersonic drag due to lift effects utilize linearized theory. The small perturbation assumption of linearized theory is severely violated near the leading edge of conventional airfoils (rounded leading edges). The leading edge is also where the primary effects of camber originate. Predicted and measured supersonic camber effects often disagree when the wing has a rounded leading edge, or when wing-body interference effects are significant. Thus, in order to allow rational design of missile or fighter which cruises or maneuvers supersonically, a higher order analytical method is needed for supersonic camber and twist design and for supersonic drag due to lift evaluation.

Vought has completed feasibility studies that show that near field solutions to obtain accurate (nonlinear) pressure distributions need not require exorbitant computer time or core. These studies used modified linear theory and significantly improved the accuracy of prediction of wave drag due to thickness for wings, cones and axisymmetric bodies. The method is not inherently restricted to planar or axisymmetric surfaces, but as yet it has not been programmed for more general shapes. The fact that a unified method has provided accuracy comparable to Van Dyke's second order theory for the variety of shapes for which it has been programmed leads to the conclusion that it should work equally well for general 3-dimensional shapes. Therefore, the modified linear theory method will be programmed to allow calculations of local flow conditions, pressures and integrated lift, pressure drag, and pitching moment on complete configurations and on wings with camber, twist and thickness.

The method has been developed to date under Vought's Independent Research and Development Program, and was initially reported in Reference 5. The objective of this project is to improve the capability to design fighter aircraft and missiles having low supersonic drag. This will be accomplished by developing higher order analysis and design routines and substituting them for the linear theory modules in the NASA/Middleton integrated supersonic design and analysis system. This objective can be divided into two parts. Part I involves the accurate prediction of zero-lift wave drag, and Part II includes prediction of supersonic drag due to lift and camber drag, thickness and camber interactions, and methods for designing optimum cambered surfaces.

Work on Part I was begun on March 13, 1978 under joint Navy/NASA sponsorship as proposed in Reference 15. This nine month effort will provide a computer program capable of calculating supersonic flow conditions over a single, non-axisymmetric body such as a fuselage with canopy. Successful completion of this program would provide a basis for a follow-on program to extend the computational capability to complete configurations.

Future work on Part II will involve the same basic techniques as in Part I with modifications as required for lifting analysis. These modifications would involve dividing the flows above and below the lifting surface (wing) into separate regions using a diaphragm technique, or adding doublet or vortex panels to the source panels used in the Part I procedure. Completion of these tasks is expected in 18-24 months if sufficient funding becomes available.

10.3.2 Transonic Wing Optimization

Maneuverability at M = 0.6 and subsonic cruise are SF-121 design factors which can be improved by transonic wing optimization. Vought Corporation and NASA Ames Research Center began a cooperative effort in 1973 to apply predictions with experiment. It was hoped that this work would establish guidelines for computational wing design and also identify areas where improvement was needed in the analysis codes. In 1975 Vought and NASA Ames began a joint effort to develop wing optimization procedures and to verify them experimentally. Camber distributions for Vought's variable camber semi-span wing model were defined using a transonic analysis code combined with an optimization procedure. The designs were tested in the NASA Ames 14 foot transonic tunnel and compared against results from previous design studies on the wing. With the feasibility of the approach now established, the procedures are being extended to encompass arbitrary wing planforms.

A three dimensional analytical design procedure was formulated by utilizing potential flow wing analysis techniques and numerical optimization within the geometric constraints of a variable camber wing. The Bailey-Ballhaus transonic potential flow (Reference 16) and Woodward-Carmichael linear potential flow analysis (Reference 18) codes were linked to Vanderplaat's constrained minimization routine (Reference 19) through a geometry module. The flap hinge lines and angle of attack were used as decision variable in the optimization routine to define the camber and twist distributions to minimize drag for the wing. The actual optimization procedure consists of pertubating each of the decision variables independently to determine gradients. The direction and relative deflection magnitudes to change the decision variables are then computed from the gradients. The controlling module of CONMIN then changes the decision variables simultaneously until either the drag increases or a constraint is encountered. A new set of gradients, along with a new move direction, is then computed. If a constraint has been reached a new direction is selected in an attempt to further reduce drag without violating the constraint. Physical limits of the flap deflections plus a maximum pitching moment limit were the constraints imposed on the design configurations. The pitching moment constraint was imposed on the design space to restrict the trim drag penalty incurred with anticipated aft wing loading. When the configuration drag could not be reduced further without violating a constraint, the optimum camber distribution has been found. Strip theory incorporating viscous effects would have to be included for analysis of the M = 0.6 maneuver condition. This is currently being developed under contract to AFFDL. A more comprehensive discussion of the methods employed and comparisons with test data is in preparation.

10.3.3 Propulsion Induced Effects

Propulsion induced effects at angles of attack exceeding stall are presently not calculable. Separated flows from the stalled aircraft invalidate estimates of near field velocities at or near the jet exit. However, if separated flows are limited to upper surfaces, freestream velocities may be used to estimate jet induced effects. Review of test results would be needed to determine the relevant flow properties. Comparison of jet-off vs. jet-on test results would permit validation or development of methodology for predicting high angle of attack jet induced effects.

Vought has been working to develop prediction methods for propulsion induced effects since 1975. The approach has been to superimpose jet effects via jet math models onto an aircraft flow field. Hess' potential flow aerodynamic analysis computer routine is the cornerstone of Vought's activity (Reference 20). Jet math models used or to be used include those by Wooler (Reference 21), Weston and Dietz (Reference 22 and 23), and Thames (Reference 24). The latter model is being developed for an NADC contract at NASA/LRC. It is being done to determine math models for rectangular jets. Vought is also currently working under contract to NADC to develop a computerized prediction method for propulsive induced forces and moments in transition and short take-off flight (Reference 25). The method is based on the Vought V/STOL Aircraft Propulsive Effects computer program (VAPE). VAPE currently calculates propulsive induced effects in the transition region at low to moderate angles of attack and in a limited portion of the STO region. This effort is concentrated upon improving the existing calculation techniques and adding new methods.

New methods are primarily aimed at incorporating improved jet modeling techniques into VAPE. First, multiple jet modifications to Weston's model (Reference 23)will be finished. This will include techniques for merged jets and methods to account for partially blocked jets. Then, the jet model will be modified to account for wake effects behind the jets. Finally, rectangular jet math models being developed under contract (Reference 24) will be integrated into VAPE. The rectangular jet math models developed also include co-flowing and small deflection cases which may be applied for analysis of high subsonic maneuver aerodynamics.

It should be noted that the analytical techniques described above are applicable to a wide range of configurations. Correlation or validation with any test configuration would offer insight into this realm of computational aerodynamics. Special attention would be needed to cover the high angles of attack experienced by a VATOL aircraft. VAPE will provide a broad based analytical capability.

11.0 CONCLUSIONS

- Comparative propulsion concept studies by Vought (Reference 1) show the Vertical Attitude Takeoff and Landing (VATOL) to be superior in performance to the alternatives.
- The SF-121 conceptual design meets or exceeds all objective performance guidelines.
- The VATOL concept exhibits excellent short takeoff performance.
- Vertical attitude transition to hover is feasible with one engine disabled.
- Aerodynamic estimates indicate the baseline configuration is directionally unstable in the post-stall regime.
- With proper reaction control phasing the indicated directional stability can be tolerated during transition.
- Sufficient thrust from one engine should be available to achieve MIL-F-83300 Level 2 combined control response in hover.
- Principal aerodynamic uncertainties are:
  - Low speed post-stall aerodynamics
  - Control power around 50 degrees angle of attack
  - Buffet characteristics in VATOL transition and in transonic maneuvering flight
  - Effectiveness of high speed thrust vectoring
  - Close coupled canard aerodynamics in the transonic/supersonic regimes
  - Inlet distortion at large angles of attack and sideslip.
- Other uncertainties about VATOL mode operations are:
  - Effects of ship wake turbulence
  - Propulsion induced spray
  - Pilot visibility requirements.
- The aerodynamic uncertainties can be resolved by a comprehensive wind tunnel test program complemented by analytical methods development program.
- The XM2R compact propulsion simulator should be a valuable adjunct to both high speed and VATOL transition regime wind tunnel tests.
- The proposed Phase II wind tunnel model can provide a quality data base and is compatible with many growth options.

- The model can easily be configured to represent VATOL or flat riser propulsion concepts.
- The study configuration is essentially uncompromised for VSTOL and is representative of advanced CTOL fighters.

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