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MODELING OF AIRPLANE PERFORMANCE FROM FLIGHT-TEST RESULTS AND VALIDATION WITH AN F-104G AIRPLANE

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16. Abstract

A technique of defining an accurate performance model of an airplane from limited flight-test data and predicted aerodynamic and propulsion system characteristics is developed. With the modeling technique, flight-test data from level accelerations are used to define a 1g performance model for the entire flight envelope of an F-104G airplane. The performance model is defined in terms of the thrust and drag of the airplane and can be varied with changes in ambient temperature or airplane weight. The model predicts the performance of the airplane within 5 percent of the measured flight-test data. The modeling technique could substantially reduce the time required for performance flight testing and produce a clear definition of the thrust and drag characteristics of an airplane.

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INTRODUCTION

The number of flight tests required to define the performance of modern aircraft and the associated costs of the tests are increasing at an alarming rate. Larger flight envelopes, the multitude of geometric variables (for example, wing sweep or inlet geometry, or both), and the variability of external store configurations of modern high performance aircraft create a matrix of conditions that is nearly impossible to encompass with conventional testing techniques. For this reason, studies are being conducted by NASA to develop a mathematical performance model from flight-test data so that the performance for the entire flight envelope of an aircraft can be determined from a limited number of flight tests.

An aircraft performance model determined from flight-test data can be defined in terms of either excess thrust (thrust minus drag) or the specific values of thrust and drag over the Mach number—altitude operating region. The use of excess thrust data to define an accurate model is limited in that the individual values of thrust and drag are not independently known; therefore, excess thrust must be determined for each geometric configuration and power setting under consideration. Thus many flight tests are necessary to obtain data over the operating envelope of an aircraft. Furthermore, once a model is defined in terms of excess thrust, it is difficult to adjust it to variations from standard-day atmospheric conditions, again because the thrust and drag are combined in one term, making it difficult to separate individual variations of the two parameters. This problem could be eliminated if a performance model were defined in terms of absolute values of thrust and drag. The determination of thrust and drag in flight is a complex, difficult, and tedious process, however, that requires considerably more flight-test time and instrumentation than the definition of excess thrust.

One way to solve this problem would be to develop a technique of defining a performance model for the flight envelope of a particular aircraft configuration from limited flight-test data and the aerodynamic and propulsion system characteristics of the aircraft. Once defined, such a model could be used to predict the flight performance of the aircraft at every point in the flight envelope without additional flight testing. If this could be done, it would reduce the time required for performance flight testing and produce a clear definition of the thrust and drag characteristics of an aircraft. This report presents the results of a study made at the NASA Flight Research Center to develop such a technique. The technique is applied to an F-104G airplane. The measured performance of the airplane is compared with the computed performance of the model.

SYMBOLS

Physical quantities in this report are given in the International System of Units (SI) and parenthetically in U.S. Customary Units. Measurements were taken in Customary Units. Factors relating the two systems are presented in reference 1.

C_D	drag coefficient, D/qS
C_L	lift coefficient, L/qS
D	total airplane drag, N (lb)
F	net thrust, N (lb)
FU	fuel used, kg (lb)
g	acceleration due to gravity, 9.8 m/sec^2 (32.2 ft/sec^2)
g_c	mass-to-force conversion factor, 9.8 N/kg (1 lbf/lbm)
h	pressure altitude, m (ft)
he	specific energy, $h + \frac{V^2}{2g}$, m (ft)
$\frac{dh}{dt}$	rate of change of altitude, m/sec (ft/sec)
$\frac{dhe}{dt}$	rate of change of specific energy, m/sec (ft/sec)
K_D	model coefficient for drag, D_t/D_p
K_F	model coefficient for thrust and fuel flow, F_t/F_{p_t} and W_{f_t}/W_{f_p}
L	airplane lift, N (lb)
M	Mach number
N	normal load factor, L/g_cW
p_{t_2}	compressor inlet total pressure, N/m^2 (lb/in^2)
q	dynamic pressure, N/m^2 (lb/ft^2)
S	wing reference area, m^2 (ft^2)
SFC	specific fuel consumption, $\frac{kg/sec}{N}$ $\left(\frac{lb/sec}{lb}\right)$
T	total temperature, °K (°R)
T_a	ambient temperature at altitude, °K (°R)

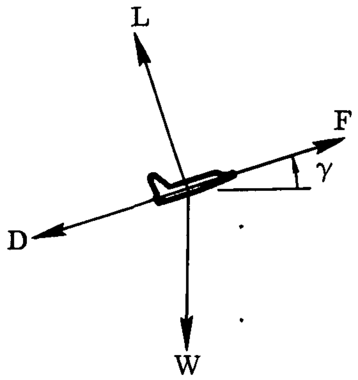
t	elapsed time, sec
V	velocity along flight path, m/sec (ft/sec)
$\frac{dV}{dt}$	rate of change of velocity, m/sec^2 (ft/sec^2)
W	airplane gross weight, kg (lb)
W_f	total fuel flow, kg/sec (lb/sec)
α	angle of attack, deg
γ	flight path angle, deg
$\Delta(\ )$	change in specific parameter
δ_{t_2}	compressor inlet total pressure ratio, $\frac{p_{t_2}}{101.33 \times 10^3}$, $\left(\frac{p_{t_2}}{14.7}\right)$
λ	thrust deflection angle, deg

Subscripts:

p	predicted
s	standard day
t	test day

PRINCIPLES OF PERFORMANCE MODELING TECHNIQUES

The term "performance model" is used in its simplest form in this report to refer to a mathematical description of the motion of an aircraft in the vertical plane as given by equations (1) and (2). For this discussion the forces were resolved parallel and perpendicular to the flight path, as indicated in the adjacent sketch, and the angle of attack was assumed to be small.



$$F - D - g_c W \sin\gamma = \frac{g_c W dV}{g dt} \qquad (1)$$

$$L - g_c W \cos\gamma = g_c W (N - \cos\gamma) \qquad (2)$$

In the development of the performance model, only nonmaneuvering flight (that is, a normal load factor of 1, constant direction of flight, and constant power setting) was considered. For a normal load factor near 1, equation (2) becomes a secondary correction to the model. As shown in references 2 to 4, equation (1) can be rewritten in terms of the rate of climb, acceleration, and velocity of the airplane as follows:

$$\frac{\left(F_t - D_t\right)V_t}{g_c W_t} = \left(\frac{dh}{dt} + \frac{VdV}{gdt}\right)_t \tag{3}$$

The right side of this equation can be determined easily by measuring the velocity and the rates of change of altitude and velocity of an airplane. On the left side of this equation, the weight can also be obtained easily; however, thrust and drag cannot be explicitly defined in flight and are unknowns in the equation. Therefore, a second equation must be developed which allows either the simultaneous solution of the two equations or the determination of an explicit value of thrust or drag.

Experience with in-flight thrust measurements on the XB-70 airplane showed that even though the measured thrust did not usually agree with the predicted thrust, the predicted specific fuel consumption, SFC_p, which is the ratio of fuel flow to thrust, was generally accurate. The ratio of the measured to the predicted specific fuel consumption is presented in figure 1 for the XB-70 airplane at maximum power. As indicated by the dispersion in the data at any given Mach number, the predicted specific fuel consumption is generally within 5 percent of the measured value. On the basis of this observation, it was assumed that the ratio of the in-flight measured fuel flow to thrust should be approximately equal to the ratio of the predicted fuel flow to thrust corrected for the test-day temperature as shown by equations (4) and (5):

$$SFC_{p_t} = W_{f_{p_t}}/F_{p_t} = W_{f_t}/F_t \tag{4}$$

where

$$\left.\begin{aligned} W_{f_{p_t}} &= W_{f_p} + \Delta W_f \\ F_{p_t} &= F_p + \Delta F \end{aligned}\right\} \tag{5}$$

The values of ΔW_f and ΔF are obtained from equations (A5) and (A6) of appendix A. Furthermore, reference 5 indicates that the climb and acceleration performance of an aircraft is fairly insensitive to nominal errors in specific fuel consumption. In reference 5 a ±10 percent deviation in specific fuel consumption resulted in a -0.4 to 0.7 percent change in the climb and acceleration performance of an F-111B airplane at maximum power.

A second assumption that must be made for equation (4) to be valid is that the predicted propulsion system characteristics available for the airplane are consistent within themselves. It is also assumed that the relationship of the thrust and fuel flow is accurately described and that the temperature corrections for these quantities are accurate.

If these assumptions are valid, the in-flight thrust can be calculated with equation (4) by measuring the fuel flow and making use of the predicted propulsion system characteristics of the airplane as follows:

$$F_t = \left(W_{f_t}/W_{f_{p_t}}\right)F_{p_t} \tag{6}$$

The drag can then be determined from equation (3). With this technique the thrust and drag of the airplane can be determined for each test point for use in the performance model.

Rather than calculate the thrust and drag for each test point, it was easier to use a model which incorporated the predicted drag, thrust, and fuel flow and then to refine these values by determining a set of coefficients, K_F and K_D, so that the performance of the model matched the test-day performance of the airplane. This was done by first computing the coefficient K_F as defined by equation (6) as follows:

$$K_F = W_{f_t}/W_{f_{p_t}} = F_t/F_{p_t} \tag{7}$$

The coefficient K_D was then determined (for test conditions) from the following version of equation (3):

$$\left[\frac{\left(K_F F_{p_t} - K_D D_p\right)V}{g_c W}\right]_{\text{model}} = \left(\frac{dh}{dt} + \frac{VdV}{gdt}\right)_{\text{airplane}} \tag{8}$$

where

$$K_D = D_t/D_p \tag{9}$$

Values of K_F and K_D were determined in this manner for each point in a test trajectory, and since values of predicted thrust and drag were available in the model, the true thrust and drag of the airplane were readily calculated.

The data analysis procedures used in this investigation are discussed in detail in the appendix.

VALIDATION OF THE PERFORMANCE MODELING TECHNIQUE

The performance modeling technique was validated on the basis of the following criterion: That having determined a set of coefficients (K_F and K_D) from a series of level accelerations performed at different altitudes, weights, and temperatures over the flight envelope of an airplane, it should be possible to calculate the performance of the airplane for any arbitrary flight trajectory encompassing climbs, dives, and accelerations as long as the maneuvering load factor increments are low (±0.2g). For the validation, the modeling technique was applied to level acceleration flight-test data obtained on an instrumented F-104G prototype airplane (fig. 2) and reported in reference 6.

The F-104G is a fixed-wing, fixed-inlet-geometry airplane powered by a J79-GE-11A, high-pressure-ratio, afterburning turbojet engine. The airplane is 16.7 meters (54.8 feet) in length and 4.1 meters (13.5 feet) in height, with a wingspan of 6.7 meters (21.9 feet) and a sea-level ratio of maximum power thrust to weight of 0.78. The test instrumentation, the airplane, and the engine are described in detail in references 6, 7, and 8, respectively.

Model Coefficients Computed From Level Acceleration Maneuvers

The model coefficients K_F and K_D were determined from the level acceleration flight-test data using a computer program. (See appendix.) The coefficients computed for two level acceleration maneuvers at an altitude of 9144 meters (30,000 feet) are presented in figure 3 together with curves faired through each set of coefficients. The analysis procedure was then repeated for level accelerations made at 3048 meters (10,000 feet), 6096 meters (20,000 feet), 9144 meters (30,000 feet), 12,192 meters (40,000 feet), and 15,240 meters (50,000 feet) to define the model coefficients for each of these test altitudes. The fairings of the coefficients K_F and K_D obtained from this analysis are presented in figures 4 and 5, respectively.

A complete performance model of the F-104G test airplane was then obtained by combining the set of model coefficients presented in figures 4 and 5 with the predicted aerodynamic and propulsion system characteristics of the airplane (tables 1 to 4, adapted from refs. 7 and 8) in a digital trajectory analysis computer program. To validate the coefficients, the trajectory program, which constitutes the performance model of the test airplane, was used to compute the model test-day performance for each of the level accelerations, with the test-day Mach number-altitude profile and the test-day temperatures at altitude used as inputs to the program. Computed and measured test-day performance of the airplane for the level accelerations made at 9144 meters (30,000 feet) is compared in figures 6(a) and 6(b). As shown, an excellent match was obtained, thus validating the coefficients for this altitude. The coefficients for the other altitudes were validated in the same manner. The performance model for the test airplane was obtained from approximately 31 minutes of flight-test data.

The computer model could also have been obtained by adjusting the predicted drag polars and the thrust and fuel flow curves in proportion to the model coefficients K_F and K_D and then entering these new curves into the program. Although this approach was not used because of the time required to adjust and reprogram the curves each time a coefficient was changed, typical adjusted and predicted drag polars for Mach numbers of 0.90 and 1.60 are presented in figures 7(a) and 7(b).

Validation of the Model Using Arbitrary Profiles

To evaluate the validity of the performance model as a whole, model and measured performance for other, arbitrarily chosen flight profiles were compared. The performance of the test airplane was measured along three different flight profiles flown from a Mach number of approximately 0.90 at an altitude of 3048 meters (10,000 feet) to a Mach number of 2.0 at an altitude of 12,192 meters (40,000 feet). The criterion established to validate the model was that the model and the measured performance of the airplane agree within 5 percent.

The measured performance of the airplane and the computed performance of the model are compared in figures 8(a) to 8(c). The width of the ±5 percent band about the flight data curve indicates the region within which the model data should fall for a satisfactory match. As shown in figure 8(a), a good match of the model and measured performance was obtained for the first flight profile. But for the other two flight profiles (figs. 8(b) and 8(c)), the computed performance does not agree with the measured performance of the airplane within the established 5 percent limit. Approximately 50 percent to 60 percent of the climb in flights B and C was at Mach numbers of 0.95 to 1.0, whereas only 5 percent of the climb in flight A was at these Mach numbers. Thus inaccuracies in the predicted airplane aerodynamics and in the flight-test data in this transonic region, coupled with the length of time spent at these Mach numbers, could have been responsible for the errors encountered at the higher Mach numbers. Consequently, the measured performance of the airplane and the computed performance of the model were compared for the supersonic portions of the flight profiles presented in figures 8(b) and 8(c). The comparisons (figs. 9(a) and 9(b)) show good agreement of the performance data, which verifies the basic assumptions and points out that improved predictions of aerodynamic data, flight-test measurements, and test techniques are needed if the model coefficients in the transonic Mach number region are to be defined more accurately. Still, the validity of the modeling technique has been demonstrated, and the time and effort required to produce this model has been substantially reduced.

Sensitivity of the Coefficients

From the preceding section and equations (6) and (9) it may be inferred that if the model performance exactly matches the measured performance of the test airplane, the in-flight thrust, fuel flow, and drag of the test airplane are defined accurately. Therefore, errors in the values of in-flight thrust, fuel flow, and drag are due to the errors in the final set of coefficients. The errors in the coefficients result from instrumentation, measurement, and data analysis errors in the values of fuel flow, altitude, velocity, rate of change of specific energy, and normal load factor. The sensitivity of the model coefficients to variations in these parameters was determined by increasing each parameter individually by 1 percent and determining the resulting percentage of change in the coefficients. The results of the error determination are presented in figure 10. As shown in figure 10(a), the coefficient K_D is sensitive to errors in fuel flow, normal load factor, and rate of change of specific energy, especially at Mach numbers less than 1.0. For example, a 1 percent error in fuel flow at Mach 0.75 results in a 4.5 percent change in the value of K_D. The results are sensitive to altitude at subsonic and supersonic speeds.

Because of the order in which the parameters were used in the modeling analysis, fuel flow was the only parameter which influenced the value of the model coefficient K_F. The variation in K_F caused by a 1 percent increase in the fuel flow (fig. 10(b)) shows that it is sensitive to error in this parameter. Thus the quality of the model coefficients depends on the accuracy with which the flight-test fuel flow, normal load factor, and altitude can be measured and on the accuracy with which the rate of change of specific energy can be calculated.

Extension of Modeling Technique

The results of the validation show that a realistic performance model of an airplane can be defined with the modeling technique used in this study and that the assumptions made in the development of the technique were valid. However, to develop a complete performance model of an airplane, the modeling techniques must be extended to cover both partial power and maneuvering flight conditions. Also, the technique was demonstrated only with a fixed-wing, fixed-inlet-geometry airplane. For a complete evaluation it should be extended to airplanes with variable wing sweep, variable inlet geometry, or both. Application of the modeling technique to any airplane is dependent on the ability to account for variations in the aerodynamic and propulsion system parameters resulting from variations in test-day temperatures and aircraft weight.

CONCLUDING REMARKS

A technique for defining an airplane performance model in terms of thrust and drag from flight-test data and predicted aerodynamics and propulsion system characteristics of the airplane was developed. With the modeling technique, a nominal 1g performance model was defined for the entire flight envelope of an F-104G airplane from approximately 31 minutes of flight-test data. Use of the technique could substantially reduce the time required for performance flight testing and produce a clear definition of the thrust and drag of an aircraft.

The study showed the sensitivity of the model coefficients to errors in measured and calculated flight-test parameters. The model coefficient for drag was sensitive to errors in fuel flow, normal load factor, altitude, and rate of change of specific energy, and the model coefficient for thrust was sensitive to errors in fuel flow. The accuracy of the performance model defined was therefore dependent on the accuracy with which these flight-test parameters were measured and calculated.

The performance modeling technique was evaluated with flight-test data from the F-104G airplane. The model performance matched the flight-test performance within 5 percent except where large portions of a trajectory were flown at transonic Mach numbers. This lack of agreement pointed out the need for improved flight-test measurements, test techniques, and predicted aerodynamic data if the model coefficients in the transonic Mach number region are to be defined more accurately.

The modeling technique was demonstrated only with a fixed-wing, fixed-inlet-geometry airplane. It should be extended to airplanes with variable wing sweep, variable inlet geometry, or both, for complete evaluation. Also, to develop a complete performance model of an airplane, the modeling technique must be extended to cover partial power and maneuvering flight conditions.

Flight Research Center,
National Aeronautics and Space Administration,
Edwards, Calif., June 28, 1972.

APPENDIX

PERFORMANCE MODELING DATA ANALYSIS PROCEDURES

The performance modeling techniques and data analysis procedures used to define a performance model for an F-104G airplane from level acceleration flight-test data are outlined in the following discussion.

The model coefficients K_F and K_D were computed from flight-test data, predicted aerodynamic and propulsion system characteristics given in references 7 and 8, and equations (A1 to A12). The flight-test parameters used to compute the coefficients were test-day true velocity, V_t, pressure altitude, h_t, Mach number, M_t, angle of attack, α, total fuel flow, W_{f_t}, normal load factor, N, and ambient temperature at altitude, T_{a_t}.

The instantaneous specific energy (or total energy per pound) of the airplane was computed for each data point as follows:

$$he_t = h_t + \frac{V_t^2}{2g} \tag{A1}$$

The rate of change of the specific energy was then computed by determining the slope at each test point of a smooth curve fitted to a time history of he_t.

The engine and afterburner fuel-used values for the test airplane were computed for each data point and summed to obtain the instantaneous total fuel used, FU, for the airplane. Instantaneous airplane weight was determined by subtracting the total fuel used at each data point from the airplane weight at engine start. The flight-test fuel flow was then obtained by determining the slope at each data point of a curve fitted to a time history of the total fuel used for each level acceleration with the following equation:

$$W_{f_t} = \frac{d(FU)}{dt} \tag{A2}$$

The predicted standard-day values of thrust and fuel flow for the test airplane were obtained from curves given in reference 7 for a maximum afterburner power setting by using the expressions

$$F_p = f(M,\ h_t) \tag{A3}$$

$$W_{f_p} = f(M,\ h_t) \tag{A4}$$

To compute the predicted test-day values of thrust and fuel flow, the changes in thrust,

ΔF, and fuel flow, ΔW_f, resulting from nonstandard day temperatures were computed. Values of ΔF and ΔW_f were obtained from plots of $\frac{\Delta F/\delta_{t_2}}{T_s - T_t}$ and $\frac{\Delta W_f/\delta_{t_2}}{T_s - T_t}$ versus test-day total temperature, respectively. The data for these plots were obtained from reference 8. The test-day total temperature at altitude was determined from the Mach number and ambient temperature, and the change in thrust and fuel flow was computed as follows:

$$\Delta F = \frac{\frac{\Delta F}{\delta_{t_2}}}{T_s - T_t}\left(T_s - T_t\right)\left(\delta_{t_2}\right) \tag{A5}$$

$$\Delta W_f = \frac{\frac{\Delta W_f}{\delta_{t_2}}}{T_s - T_t}\left(T_s - T_t\right)\left(\delta_{t_2}\right) \tag{A6}$$

The model coefficient K_F was calculated for each data point with the expression

$$K_F = \frac{W_{f_t}}{\left(W_{f_{p_s}} + \Delta W_f\right)} \tag{A7}$$

The flight-test thrust of the airplane was then computed by using the expression

$$F_t = \left(F_p + \Delta F\right) K_F \tag{A8}$$

The total flight-test drag of the airplane was computed as follows:

$$D_t = F_t - \left[\left(\frac{dhe}{dt}\right)_t \frac{g_c W_t}{V_t}\right] \tag{A9}$$

To compute the model coefficient K_D, the predicted flight-test drag of the airplane was calculated. The predicted drag coefficient was then obtained from the aerodynamic data presented in reference 7 by using the equation

$$C_{D_p} = f\left(C_{L_t}, M\right) \tag{A10}$$

where

$$C_{L_t} = \frac{1}{qS}\left[\left(\frac{N}{\cos\alpha} + \frac{(F - D)}{g_c W_t}\tan\alpha\right) g_c W_t - F\sin(\alpha - \lambda)\right]$$

The predicted drag of the test airplane for each data point was computed as follows:

$$D_p = C_{D_p} qS \tag{A11}$$

The model coefficient K_D was computed for each data point using the results of equations (A9) and (A11) and the expression

$$K_D = D_t/D_p \tag{A12}$$

The values of the model coefficients K_F and K_D were plotted against Mach number for each altitude at which a level acceleration maneuver was performed. Then a curve was faired through each set of coefficients. These fairings were combined with the predicted aerodynamic and propulsion system characteristics of the airplane in a two-dimensional trajectory analysis program to define the performance model of the test airplane.

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TABLE 1

VALUES OF LIFT COEFFICIENT FOR THE F-104G AIRPLANE AT SPECIFIC ANGLES OF ATTACK AND MACH NUMBERS

	C_L for an M of –									
α, deg	0.4	0.6	0.8	0.9	1.0	1.2	1.4	1.6	1.8	2.0
0	0.010	0.010	0.010	0.010	-0.010	-0.020	0.040	0.032	0.020	0.010
2	.100	.100	.118	.145	.140	.125	.155	.132	.110	.090
4	.210	.215	.270	.300	.300	.255	.275	.235	.200	.175
6	.318	.330	.415	.465	.450	.380	.390	.340	.290	.252
8	.430	.445	.535	.610	.600	.515	.500	.440	.380	.330
10	.540	.555	.630	.720	.740	.650	.618	.540	.470	.409
12	.655	.659	.710	.800	.868	.765	.720	.639	.559	.480
14	.743	.748	.775	.870	.980	.870	.820	.730	.642	.560

TABLE 2

VALUES OF DRAG COEFFICIENT FOR THE F-104G AIRPLANE AT SPECIFIC MACH NUMBERS AND LIFT COEFFICIENTS

	C_D for a C_L of –									
M	0	0.10	0.15	0.20	0.25	0.30	0.35	0.40	0.45	0.50
0	0.0160	0.0175	0.0195	0.0222	0.0258	0.0308	0.0378	0.0462	0.0562	0.0673
.800	.0160	.0175	.0195	.0222	.0258	.0308	.0378	.0462	.0562	.0673
.850	.0163	.0180	.0197	.0220	.0260	.0315	.0387	.0475	.0575	.0683
.880	.0170	.0184	.0200	.0228	.0269	.0329	.0402	.0491	.0591	.0700
.900	.0170	.0185	.0203	.0233	.0278	.0335	.0408	.0495	.0600	.0718
.925	.0190	.0200	.0219	.0250	.0295	.0357	.0431	.0521	.0632	.0755
.950	.0230	.0240	.0268	.0305	.0355	.0417	.0488	.0578	.0685	.0802
.975	.0310	.0338	.0357	.0383	.0428	.0488	.0562	.0652	.0752	.0880
1.000	.0380	.0401	.0420	.0455	.0509	.0575	.0650	.0738	.0852	.0983
1.050	.0470	.0485	.0512	.0555	.0606	.0670	.0763	.0870	.0990	.1118
1.100	.0500	.0520	.0550	.0595	.0653	.0740	.0820	.0913	.1040	.1175
1.150	.0500	.0522	.0555	.0599	.0661	.0739	.0830	.0940	.1059	.1210
1.200	.0490	.0519	.0553	.0602	.0669	.0750	.0850	.0959	.1088	.1235
1.400	.0468	.0495	.0535	.0589	.0663	.0758	.0870	.1000	.1145	.1305
1.600	.0440	.0470	.0515	.0579	.0663	.0770	.0895	.1038	.1193	.1360
1.800	.0430	.0472	.0523	.0591	.0685	.0798	.0930	.1085	.1250	.1435
2.000	.0422	.0463	.0519	.0590	.0689	.0815	.0960	.1130	.1315	.1515

TABLE 3

VALUES OF NET THRUST FOR THE F-104G AIRPLANE AT SPECIFIC ALTITUDES AND MACH NUMBERS

	F, in N (lb), for an M of –									
h, m (ft)	0	0.4	0.6	0.8	1.0	1.2	1.4	1.6	1.8	2.0
0 (0)	49,820 (11,200)	60,673 (13,640)	66,945 (15,050)	74,596 (16,770)	84,961 (19,100)	95,859 (21,550)	106,534 (23,950)	117,210 (26,350)	127,886 (28,750)	138,561 (31,150)
3,048 (10,000)	34,251 (7,700)	44,704 (10,050)	51,154 (11,500)	58,716 (13,200)	67,168 (15,100)	76,509 (17,200)	89,409 (20,100)	102,309 (23,000)	115,653 (26,000)	128,998 (29,000)
6,096 (20,000)	20,684 (4,650)	30,559 (6,870)	36,920 (8,300)	43,948 (9,880)	51,822 (11,650)	60,940 (13,700)	71,972 (16,180)	81,180 (18,250)	89,409 (20,100)	97,860 (22,000)
9,144 (30,000)	15,124 (3,400)	23,042 (5,180)	26,912 (6,050)	31,137 (7,000)	37,810 (8,500)	45,372 (10,200)	54,224 (12,190)	62,186 (13,980)	70,415 (15,830)	75,842 (17,050)
12,192 (40,000)	13,478 (3,030)	16,903 (3,800)	18,593 (4,180)	20,462 (4,600)	23,575 (5,300)	28,246 (6,350)	34,918 (7,850)	41,947 (9,430)	48,218 (10,840)	54,401 (12,230)
15,240 (50,000)	7,562 (1,700)	10,008 (2,250)	11,209 (2,520)	12,544 (2,820)	14,457 (3,250)	17,215 (3,870)	21,440 (4,820)	26,111 (5,870)	29,936 (6,730)	33,273 (7,480)
18,288 (60,000)	1,779 (400)	4,671 (1,050)	6,183 (1,390)	7,562 (1,700)	9,030 (2,030)	10,809 (2,430)	13,300 (2,990)	16,014 (3,600)	18,682 (4,200)	20,551 (4,620)

TABLE 4

VALUES OF FUEL FLOW RATE FOR THE F-104G AIRPLANE AT SPECIFIC ALTITUDES AND MACH NUMBERS

	Fuel flow rate, in kg/sec (lb/sec), for an M of –									
h, m (ft)	0	0.4	0.6	0.8	1.0	1.2	1.4	1.6	1.8	2.0
0 (0)	3.23 (7.13)	3.97 (8.75)	4.42 (9.75)	5.01 (11.05)	5.86 (12.91)	6.83 (15.05)	7.80 (17.19)	8.77 (19.33)	9.74 (21.47)	10.71 (23.61)
3,048 (10,000)	2.23 (4.91)	2.89 (6.38)	3.27 (7.22)	3.74 (8.25)	4.36 (9.62)	5.14 (11.33)	5.92 (13.05)	6.80 (15.00)	7.68 (16.94)	8.56 (18.88)
6,096 (20,000)	1.52 (3.36)	2.05 (4.52)	2.34 (5.16)	2.68 (5.91)	3.13 (6.91)	3.75 (8.27)	4.47 (9.86)	5.30 (11.69)	6.01 (13.25)	6.71 (14.80)
9,144 (30,000)	1.01 (2.22)	1.43 (3.16)	1.65 (3.63)	1.89 (4.16)	2.21 (4.88)	2.63 (5.80)	3.15 (6.94)	3.73 (8.22)	4.30 (9.47)	4.79 (10.55)
12,192 (40,000)	.50 (1.11)	.91 (2.00)	1.11 (2.44)	1.32 (2.91)	1.55 (3.41)	1.81 (4.00)	2.15 (4.75)	2.55 (5.63)	2.93 (6.47)	3.20 (7.05)
15,240 (50,000)	.34 (.75)	.63 (1.38)	.77 (1.69)	.91 (2.00)	1.06 (2.33)	1.22 (2.69)	1.43 (3.16)	1.67 (3.69)	1.90 (4.19)	2.05 (4.52)
18,288 (60,000)	.04 (.08)	.30 (.66)	.43 (.94)	.55 (1.22)	.68 (1.50)	.80 (1.77)	.94 (2.08)	1.08 (2.38)	1.22 (2.69)	1.31 (2.88)

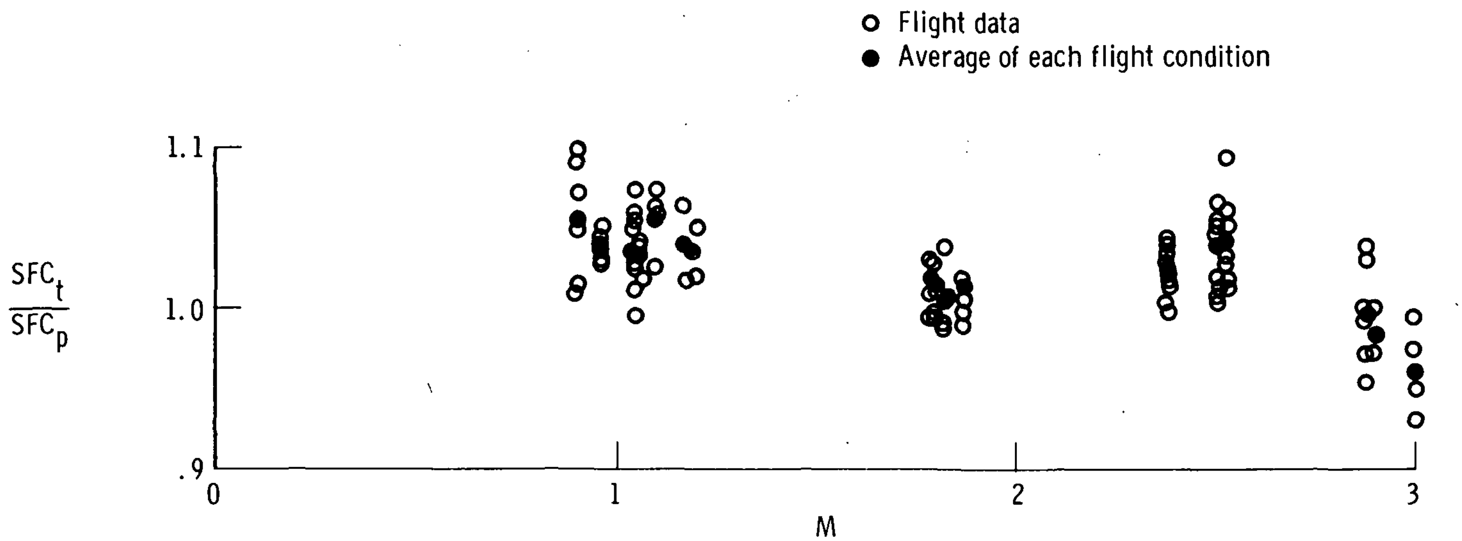


Figure 1. Ratio of measured to predicted specific fuel consumption for the XB-70 airplane at maximum power.

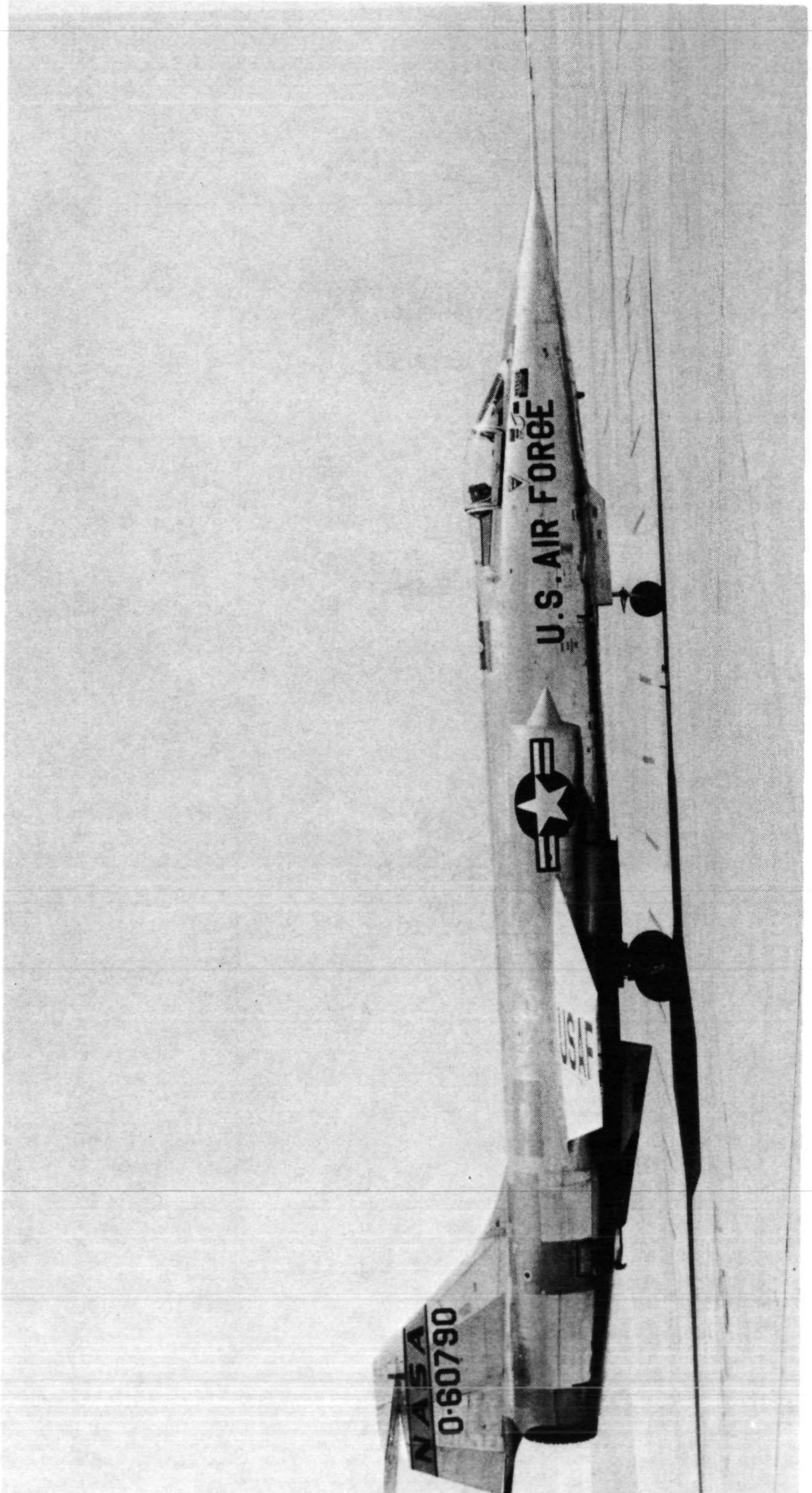


Figure 2. F-104G airplane.

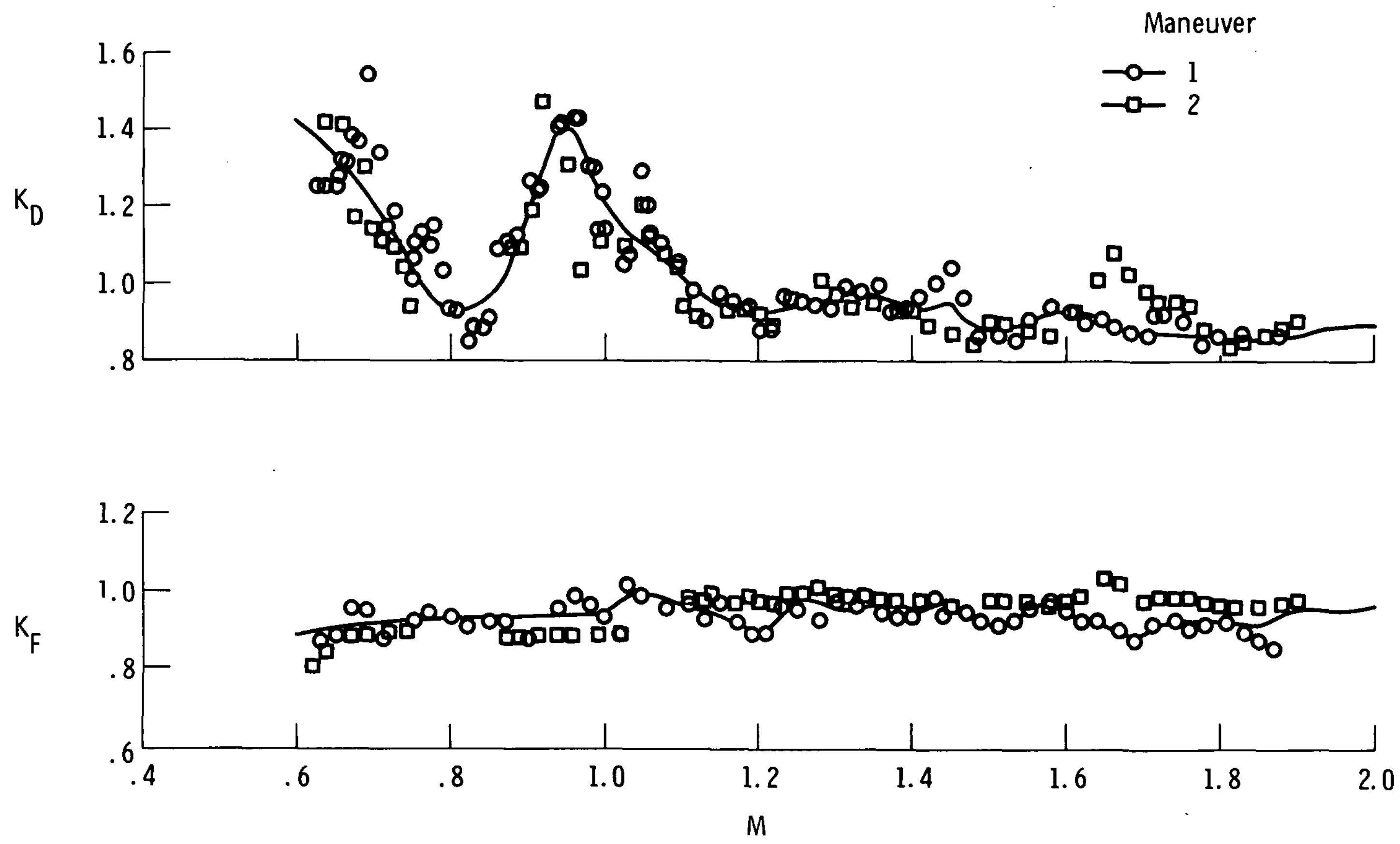


Figure 3. Model coefficients computed from maximum power level accelerations at an altitude of 9144 meters (30,000 feet) for an F-104G airplane.

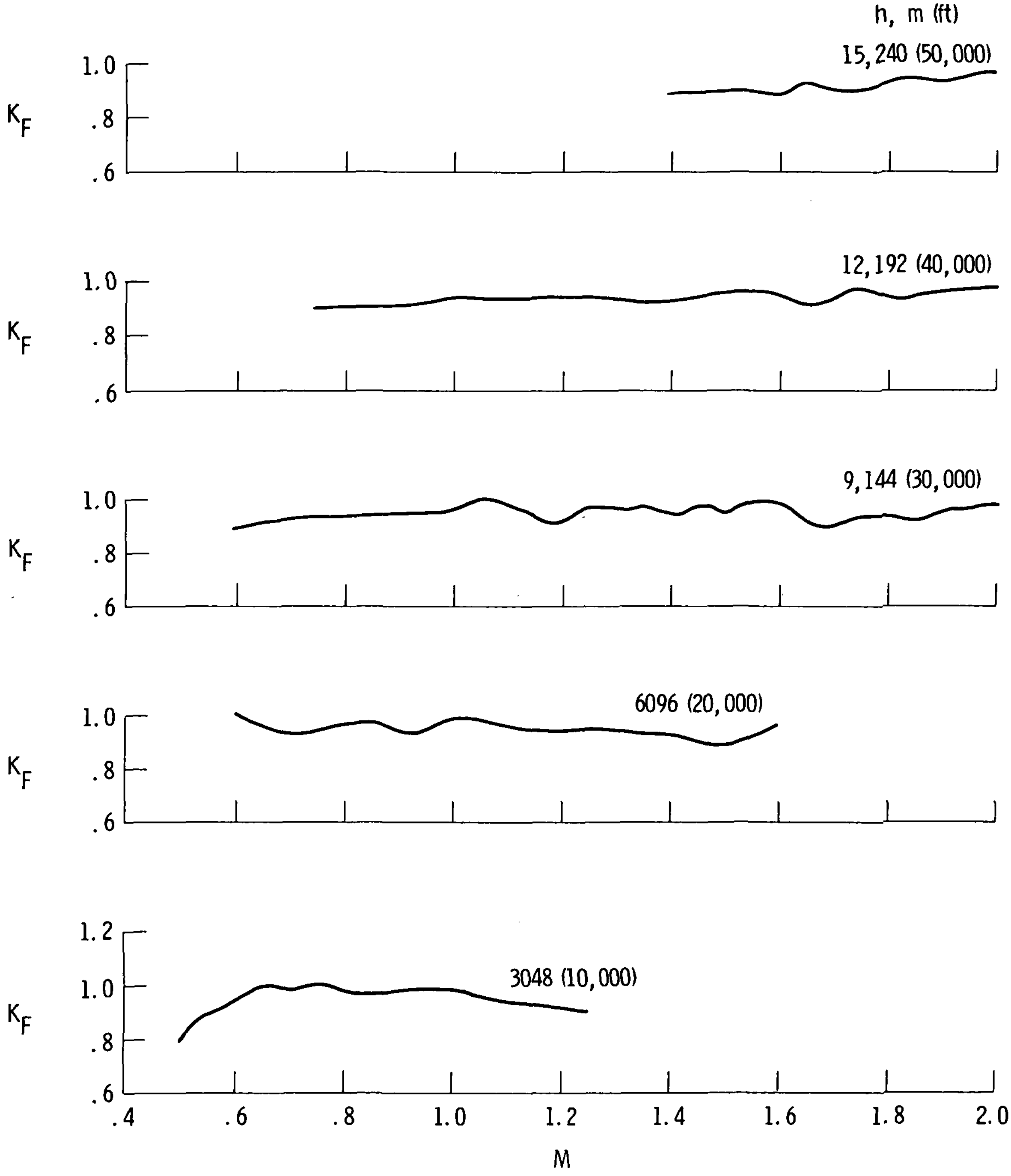


Figure 4. Computed values of model coefficient K_F for thrust and fuel flow for the flight envelope of an F-104G airplane.

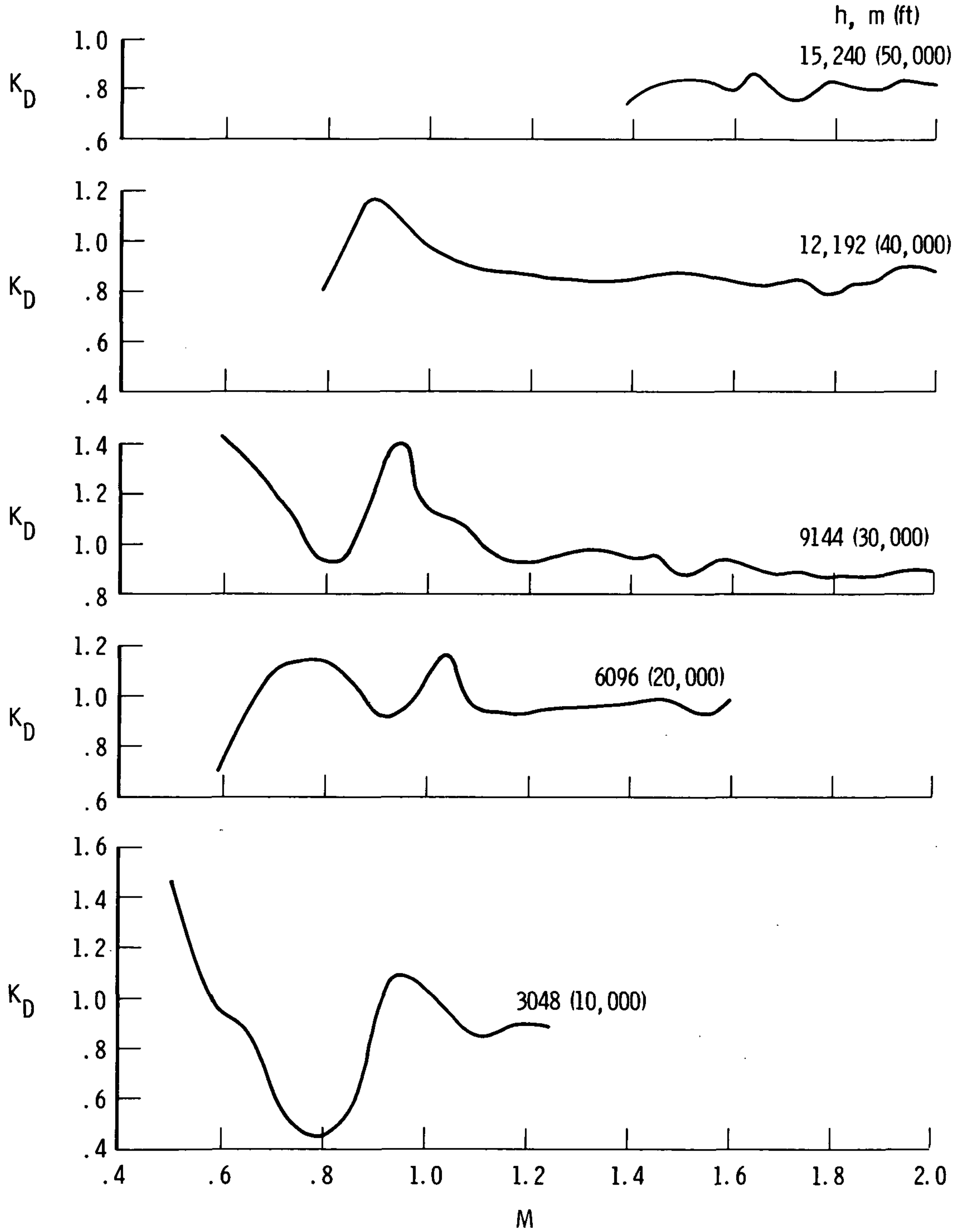
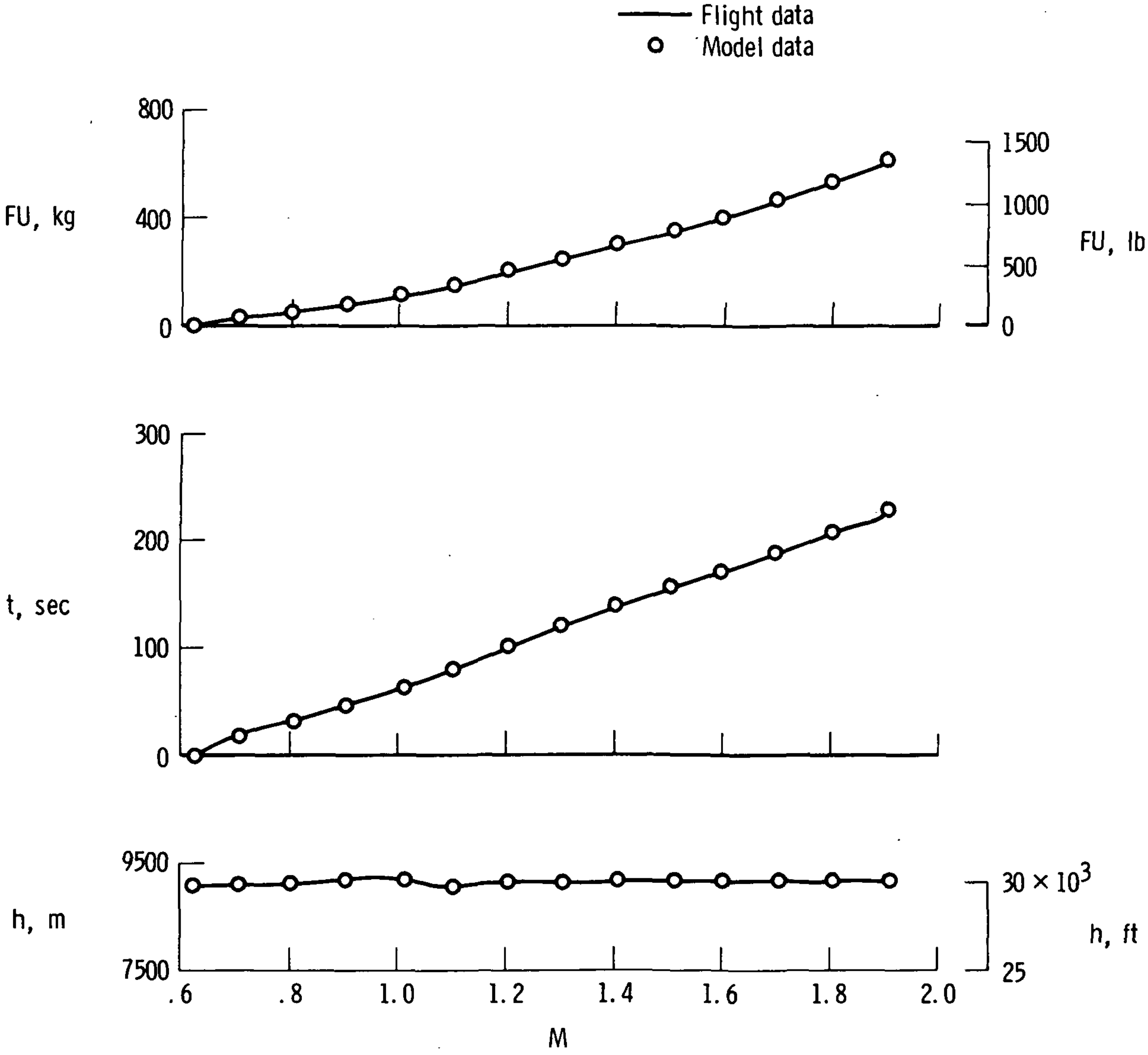
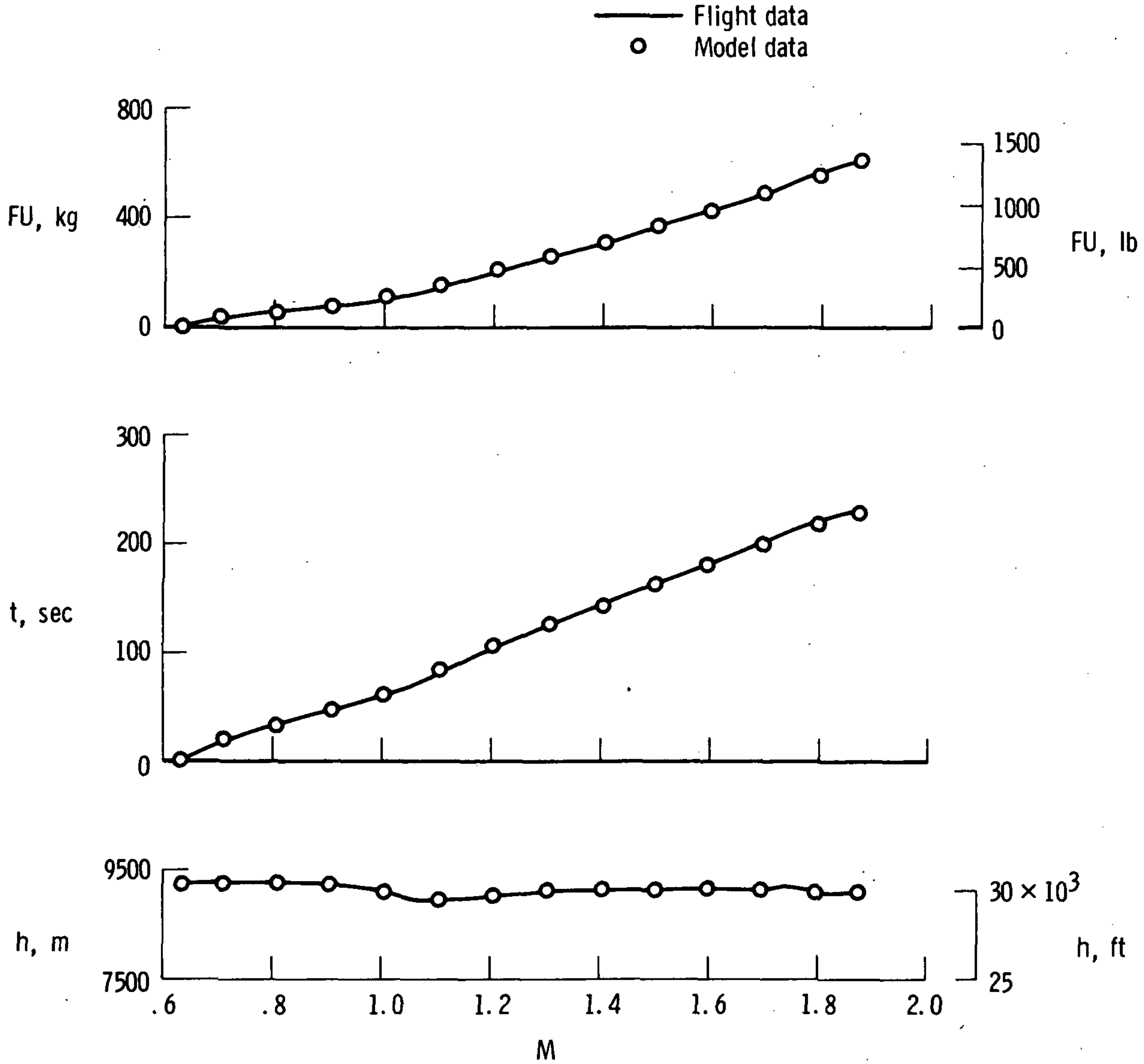


Figure 5. Computed values of model coefficient K_D for drag for the flight envelope of an F-104G airplane.



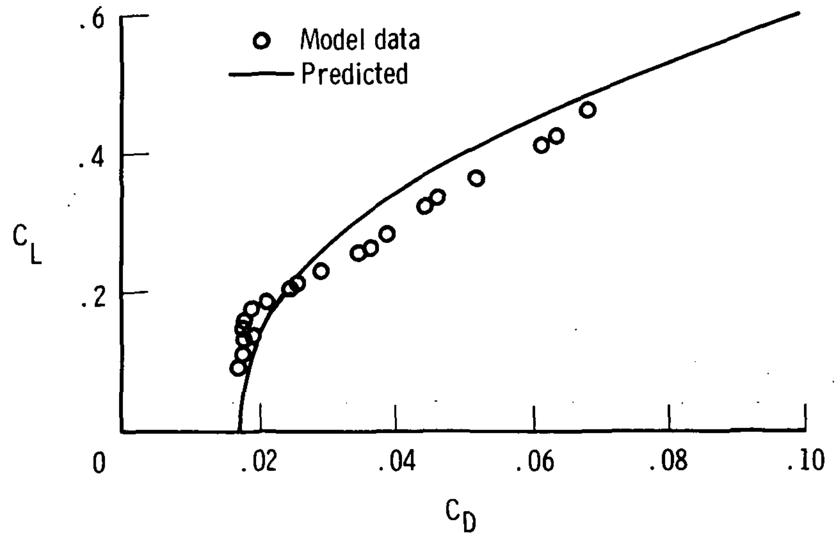
(a) Maneuver 1.

Figure 6. Comparison of flight-test data and model-derived data for a maximum power level acceleration of an F-104G airplane.

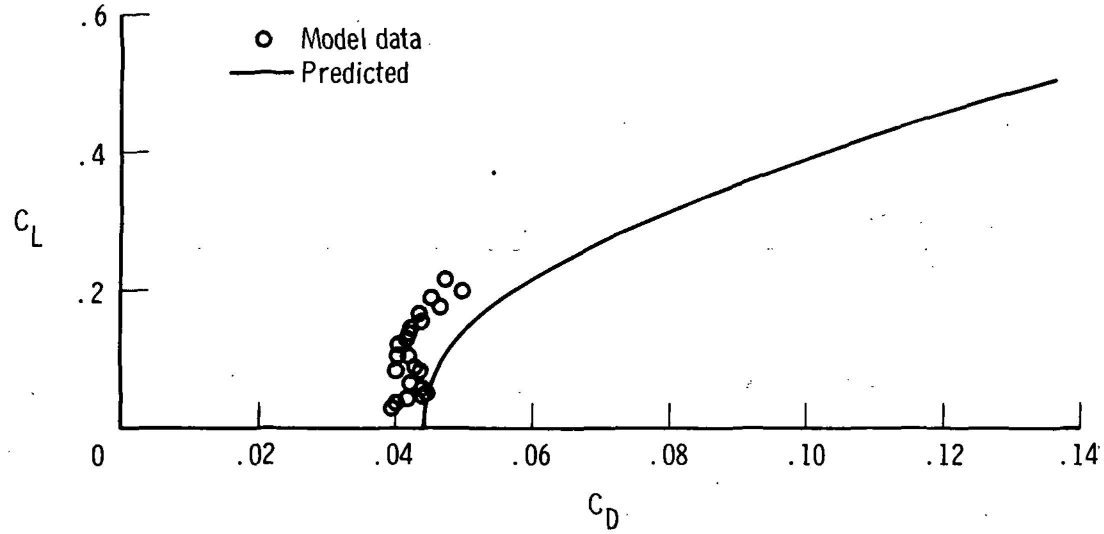


(b) Maneuver 2.

Figure 6. Concluded.

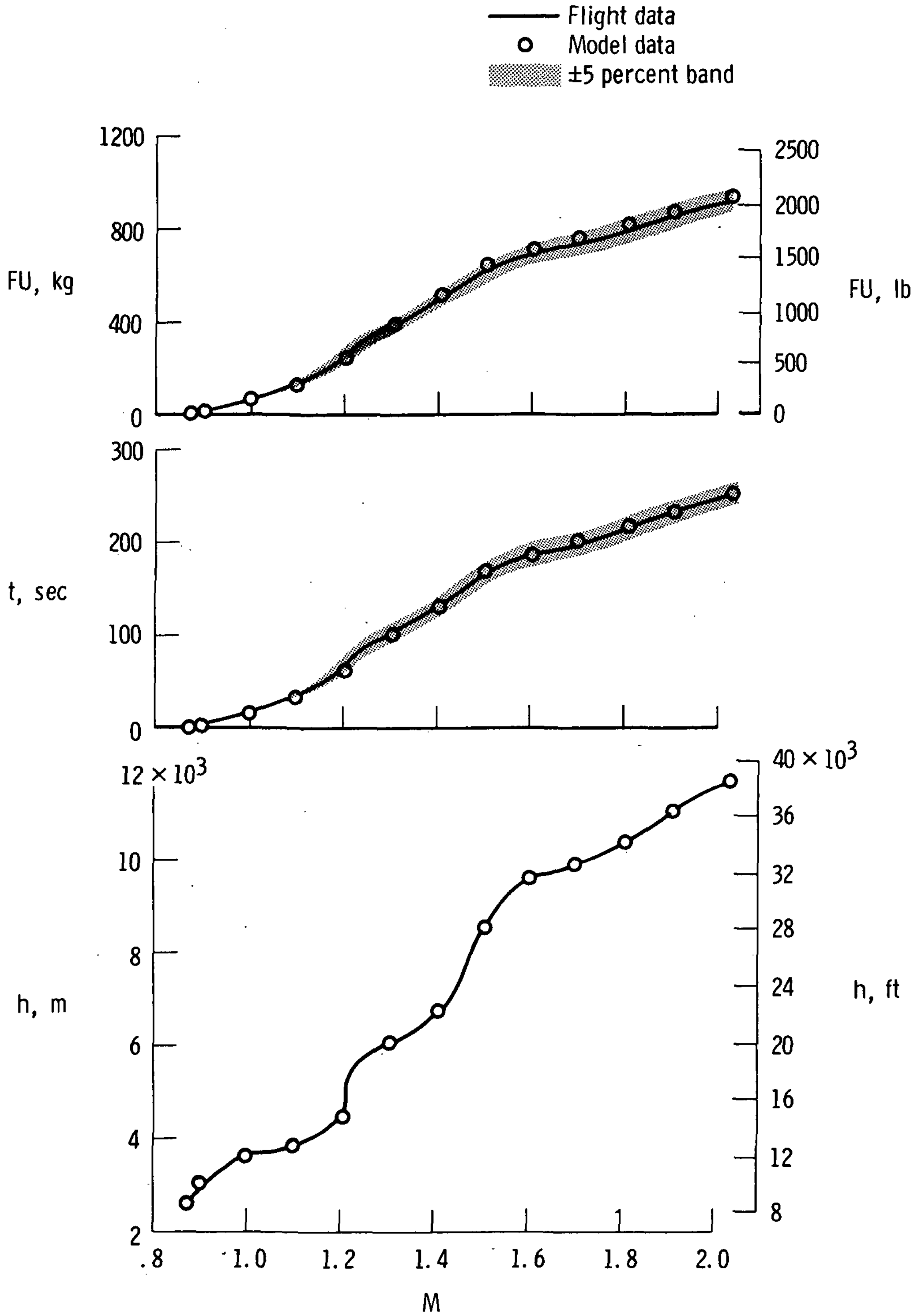


(a) M = 0.90.



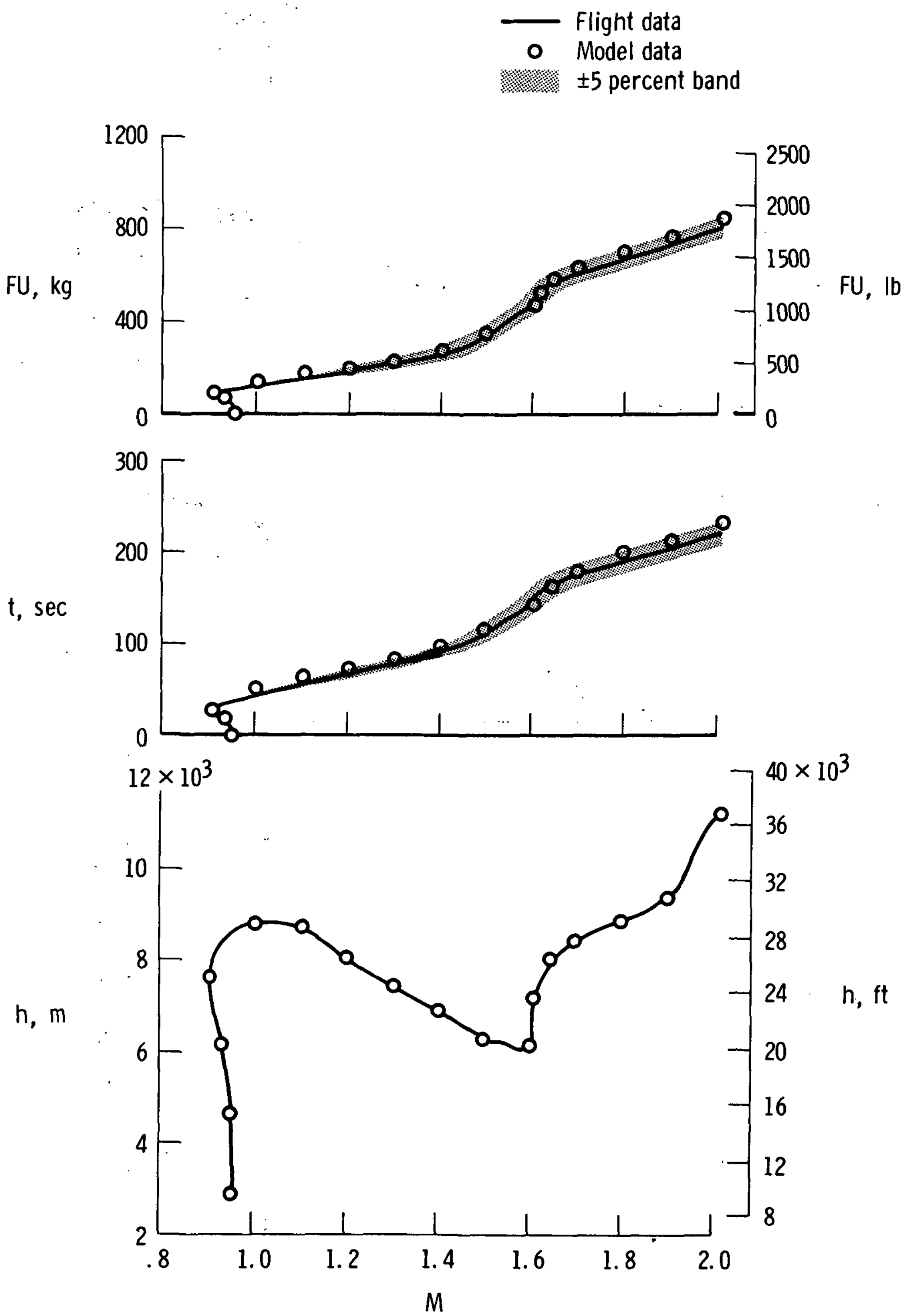
(b) M = 1.60.

Figure 7. Comparison of adjusted model and predicted drag polars for an F-104G airplane in a clean configuration.



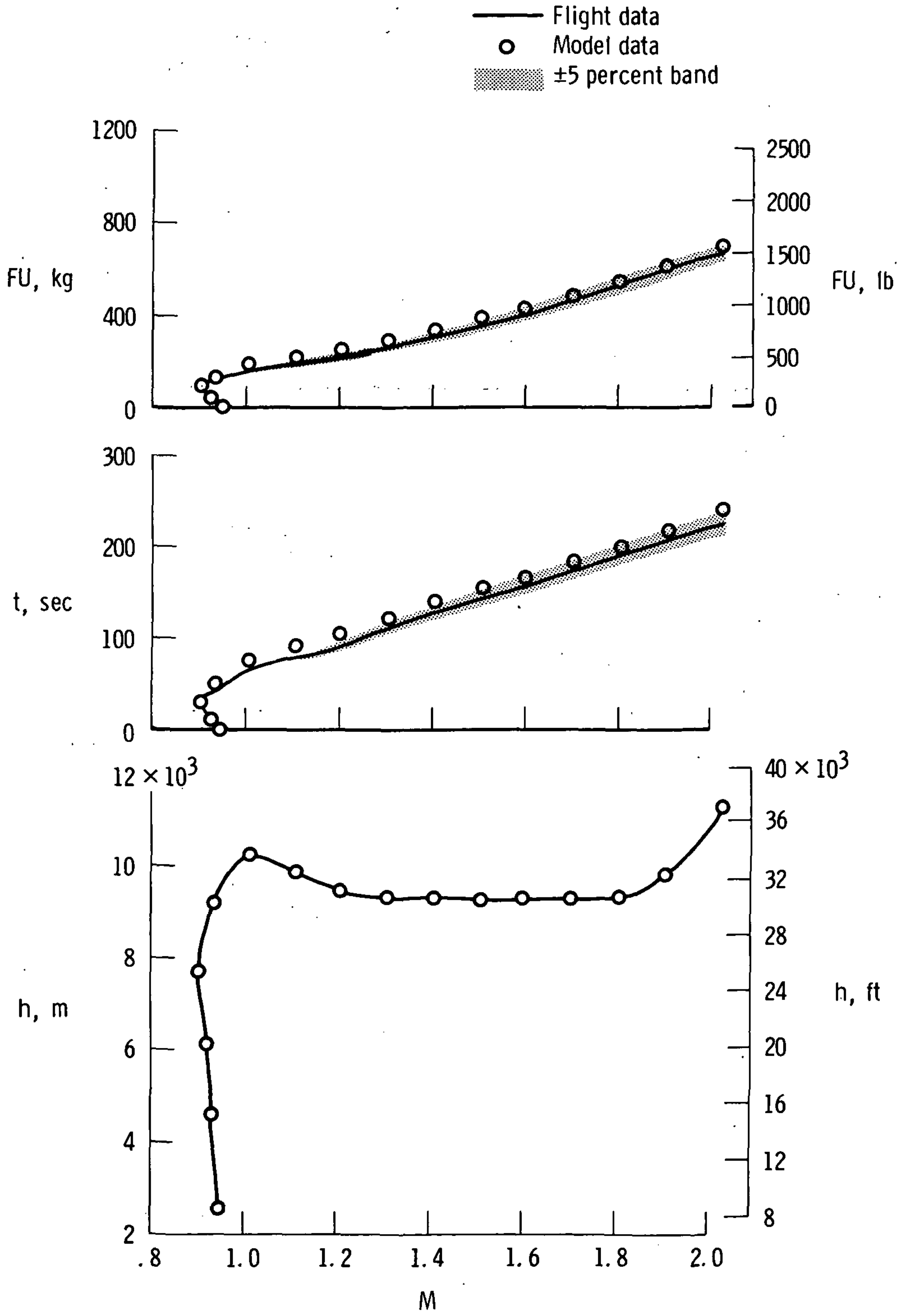
(a) Flight A.

Figure 8. Comparison of flight-test data and model-derived data for maximum power climb of an F-104G airplane.



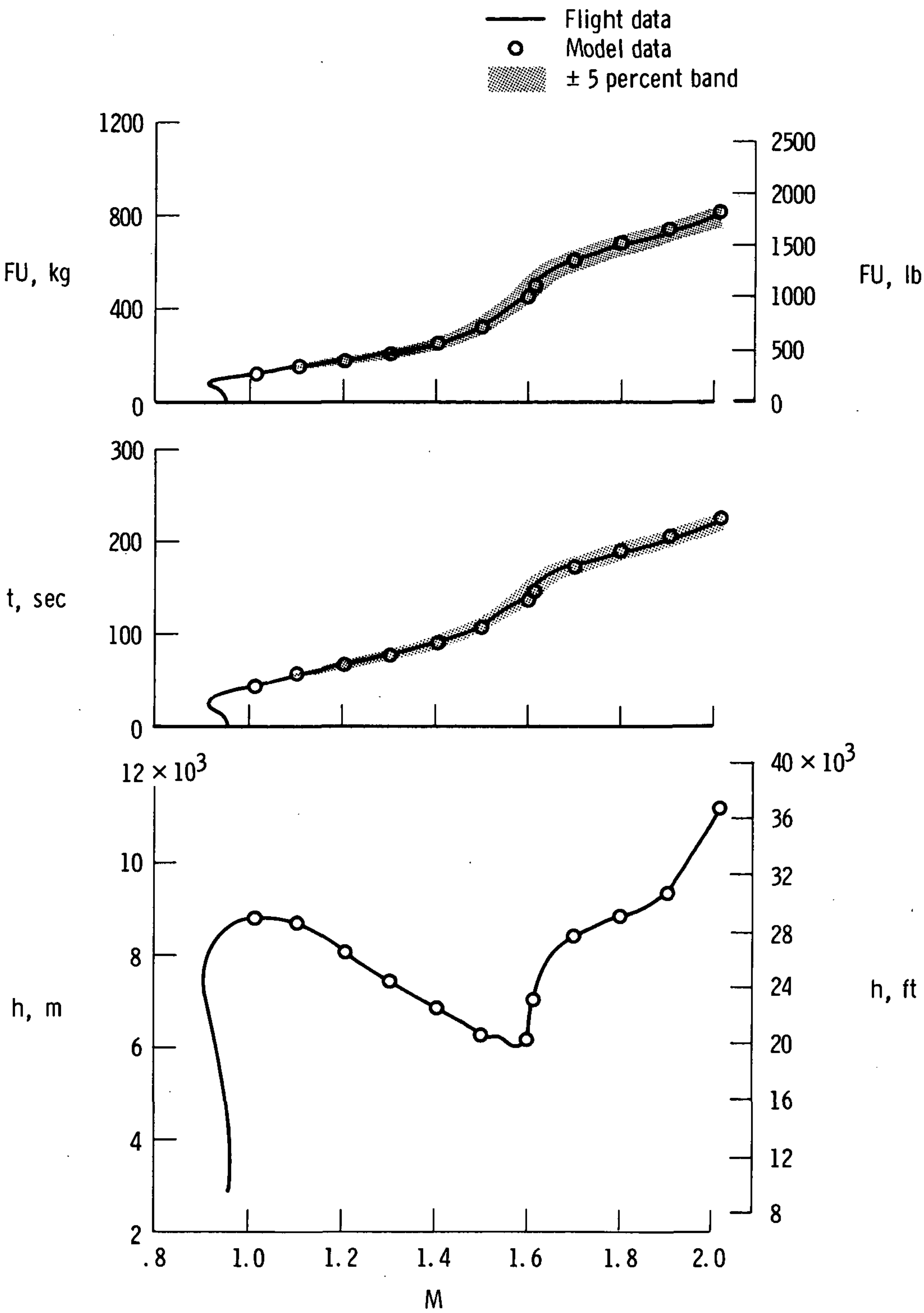
(b) Flight B.

Figure 8. Continued.



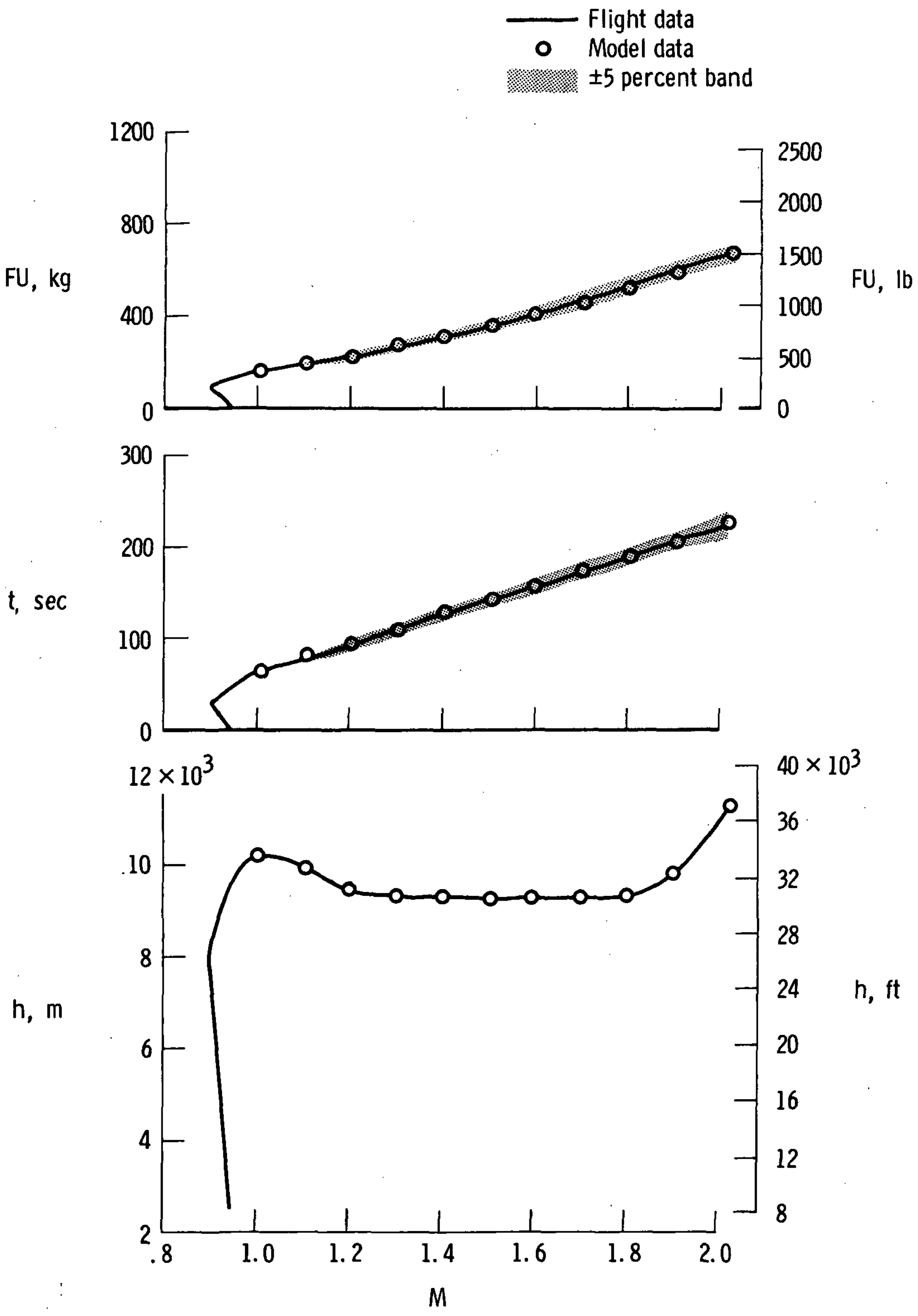
(c) Flight C.

Figure 8. Concluded.



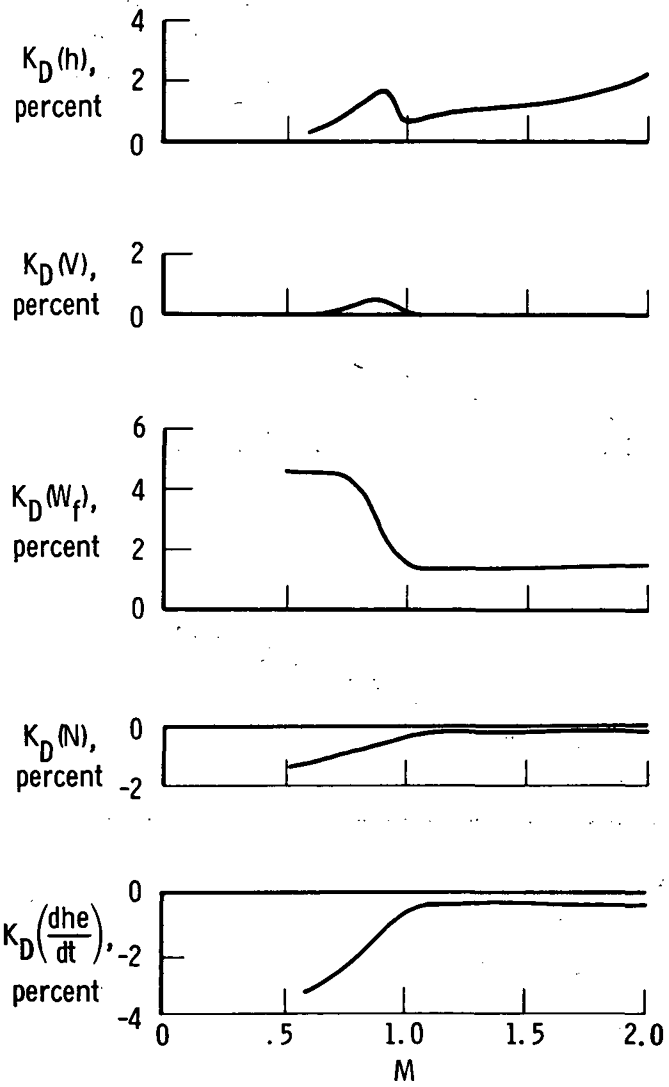
(a) Flight B.

Figure 9. Comparison of flight-test data and model-derived data for the supersonic portion of a flight of an F-104G airplane.

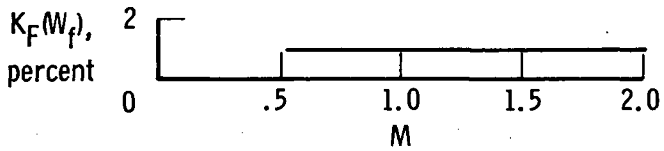


(b) Flight C.

Figure 9. Concluded.



(a) K_D.



(b) K_f.

Figure 10. Percentage of change in model coefficients K_D and K_F resulting from a 1 percent increase in important parameters.

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