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# A SHUTTLE DEVELOPMENT-FLIGHT-TEST VEHICLE STUDY

Robert W. Rainey, John J. Rehder, and Phillip J. Klich



June 1974





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# A SHUTTLE DEVELOPMENT-FLIGHT-TEST VEHICLE STUDY

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## SUMMARY

A study has been completed that identifies the potential performance capability of the production shuttle orbiter for powered flight tests using several propulsion systems following vertical takeoff and using J-2 rocket engines following air launch. Of the approaches considered, the air-launched orbiter equipped with J-2 rocket engines appeared to have the highest potential for early shuttle development flights. With this approach, Mach 4 appeared attainable using 45.5K kg (100K lb) of internal propellant. Several issues were identified that require resolution to prove feasibility.

## INTRODUCTION

A study has been recently completed at the Langley Research Center to identify the performance potential of a production shuttle orbiter retrofitted for development flights. To avoid delays in shuttle development tests which might result from waiting for space shuttle main engine (SSME) delivery, the J-2 liquid rocket engine and a modified 1205 solid rocket were considered as alternate propulsion systems. These were the only available propulsion systems that appeared to have potential compatibility with the orbiter to meet a development-flight-test objective of obtaining aerodynamic flight data into the supersonic regime. Emphasis was given to providing gradual penetration of the high-angle-of-attack regime and to taking at least an initial step in the demonstration of the angle-of-attack transition maneuver. Pilot familiarization and training as well as onboard system verification tests would also be accomplished inflight rather than by ground simulations. This approach to orbiter development flight testing encompasses the subsonic program presently envisioned; and, although more complex and costly than that program, it increases mission reliability in preparation for the first orbital flight following vertical launches. A detailed econometric and safety analysis will be required to identify the cost increase and shuttle schedule impact of this approach as well as the reduction in risk.

Two operational modes were investigated during this study: vertical takeoff (VTO) and air launched, the latter (suggested by Flight Research Center personnel) requiring a mothership to tow or carry the shuttle orbiter to altitude. Within the scope of this limited study, the primary emphasis was given to the ascent performance capability, vehicle weight, and systems layouts. Brief consideration was given to the launch site and test range requirements for the VTO mode of operation. It was assumed that the aerodynamic and maneuver capability and the thermal protection system of the orbiting version will accommodate the requirements for development and research flights. The Rockwell International shuttle orbiter with a dry weight of about 68K kg (150K lb) and a reference length, $\ell$, of 33.73 m (1,328 in) was used as the baseline vehicle. Where appropriate, all systems that were necessary for orbiting missions but had no function for shuttle development flight testing were removed; for several systems, such as the environmental control and reaction control systems still necessary for the development flights, the consumable weights were reduced to reflect the shorter time of operation. Although the baseline orbiter weights, subsystems, etc., are continually changing, it is believed that the results obtained herein are representative of the development flight test performance of the final shuttle orbiter, barring major changes in the orbiter configuration, its weight, or mode of operation.

## RESULTS AND DISCUSSION

In order to evaluate the potential of retrofitting the production orbiter for development flight tests and to calculate performance, the weights and center-of-gravity (c.g.) locations had to be estimated for the altered versions. This was accomplished utilizing the three primary modes of propulsion considered in the study: the 3.05 m (120-in.) diameter, five-segment solid rocket motor (1205 SRM), the J-2 liquid rocket engine, and the space shuttle main engines (SSME). In considering the weights and systems layouts using each of these propulsion modes, it was necessary to adapt the propulsion systems to the orbiter; and, in all cases (including the three SSMEs), some alteration of the aft part of the orbiter was necessary. This will be elaborated upon subsequently. Figure 1 summarizes the weights and c.g. locations of the development-flight-test versions in comparison with values for a baseline Rockwell International orbiter. For the 1205 SRM, the words "External" and "Internal" refer to the SRM mounted below the orbiter and within the interior of the orbiter, respectively. For the J-2 and SSME installations, "External" and "Internal" refer to the LOX/LH propellant tanks installed below the orbiter and within the payload bay, respectively. Because of the installation and operational complexities identified during the initial considerations of the 1205 SRM for the VTO mode, no trajectories were calculated, and no consideration was given to the use of the 1205 SRM in the air-launched mode.

### Vertical Takeoff (VTO)

Using the weights from figure 1 and Table I for each of the propulsion systems considered, the Program for Optimized Shuttle Trajectories (POST) (reference 1) was used to obtain the maximum burnout velocities at specified altitudes. Both maximum relative and ideal velocities are summarized in figure 2; the velocity increment between the two is approximately 1830 m/sec (6,000 ft/sec) for each system and is predominantly representative of gravity and aerodynamic-drag losses. For the trajectories calculated, the maximum total acceleration was 3.09; the maximum dynamic pressure (q) was 31.1K N/m$^2$ (650 lb/ft$^2$); and the maximum dynamic pressure times angle of attack ($q\alpha$) was 134K N/m$^2$-deg (2,800 lb/ft$^2$-deg). The maximum values of acceleration and $q$ were limited to those of the orbiting version, and the value of $q\alpha$ was approximately 2/3 of that used for ascent in gusty air and wind-shear conditions along the baseline ascent trajectory.

Details of each VTO concept follow:

Solid Rocket Powered.- Two installations of the United Technology 1205 SRM (used on the Titan IIIC launch vehicle) were considered. On the externally-mounted version (figure 3(a)), with a 10$^\circ$ nozzle cant angle, the thrust is directed between the liftoff and burnout c.g. locations; approximately ±4-1/2$^\circ$ of thrust vectoring are required to track the center-of-gravity location from liftoff to burnout. The present 1205 SRM has ±6$^\circ$ thrust vector control available through liquid injection, and it is questionable whether this would be sufficient to provide ascent trajectory and flight control for all wind conditions. Early thrust termination may be provided through blowout patches. Relatively large negative angles of attack would result during ascent. Following staging, the center of gravity is at approximately 0.60$\ell$ (figure 1(a)); thus, about 11.3K kg (25K lbs) of ballast aft of the orbiter thrust structure would be required to shift the c.g. to 0.65$\ell$ (the forward limit of the baseline orbiter). For the internally-mounted SRM (figure 3(b)), major thrust structure alteration is necessary. No thrust termination is believed feasible, and the orbiter entry weight is about 18K kg (40K lbs) greater than for the baseline (figure 1(a)). No ascent trajectories were calculated for either of these solid rocket motor configurations, and they are not recommended for further consideration.

J-2 Powered.- Because of the smaller overall dimensions of the J-2 engines relative to those of the SSME, modifications to the baseline engine mounts are necessary to move the exit planes of the J-2s aft (figure 4). With the J-2 installation, thrust termination is potentially available for 1, 2, or 3 engines. In addition, about 20 percent thrust reduction can be achieved by reducing the mixture ratio from 5.5 to 4.5 with an increase in specific impulse, Isp. Therefore, a considerable variation of total thrust may be achieved by combinations of mixture control and engine shutoff. The estimated center-of-gravity at liftoff is at approximately 0.50$\ell$ with a gross liftoff weight of 156K kg (344K lb). Burnout weight is about 67K kg (147K lb) with the c.g. at 0.64$\ell$; some ballast is necessary to shift the center of gravity back to about 65 or 66 percent $\ell$ for compatibility with the orbiting version. For the three J-2-powered, VTO concept, the available sea-level thrust was 219K kg (483K lb). The ascent propellant weight was based on providing a liftoff thrust-to-weight ratio (T/W) of 1.40. The maximum relative velocity predicted for this J-2 installation was about 1675 m/sec (5500 ft/sec) at a burnout altitude of about 45.7 km (150K feet) (figure 5). Basic trajectory parameters for this burnout altitude are presented in figure 6. For the three trajectories calculated, the maximum angle of attack never exceeded $\pm 1^\circ$ resulting in $q\alpha < 2400$ N/m$^2$-deg (50 lb/ft$^2$-deg). During the ascent when the acceleration reached 3, one engine was shut down; although no thrust modulation via mixture-ratio variation was used, this approach could provide fine-tuning of the trajectories. For all trajectories calculated, the flight path angle at burnout increased with altitude reaching a value of 39$^\circ$ at a burnout altitude of 61K m (200K feet); this parameter, along with several others, is important in establishing the initial conditions for the glide flight. Rocket burn time was approximately 135 seconds; the range at burnout for the three trajectories varied between 46 and 63 km (25 and 34 n.mi.).

The J-2 engines were designed for high-altitude start and operation with the S-II and S-IV stages of the Saturn V launch vehicle. In considering their use at low altitudes, transient lateral loads resulting from asymmetric flow separation in the nozzle during thrust buildup and termination must be considered. Personnel of Rocketdyne Division of Rockwell International have stated through private communications that these transients are significant and damage can result to the actuation and gimbal mechanisms. For starting, a nozzle support structure attached to the launch pad, similar to the present ground-testing arrangement, could be used. After the full chamber pressure of $4.7 \times 10^6$ N/m$^2$ (680 lb/in$^2$) is reached, this support structure may be released, and no further separation problems during start is anticipated. The transient shutdown loads must be reckoned with, however. Rocketdyne personnel have also warned of the problem of excessive LOX head at the pump inlets which, for this installation, would very likely require suppression valves. The ability to start and operate the J-2, including the impact of accelerations to 3 g's, requires study.

SSME Powered.- With two SSMEs and internal propellant tanks (figure 7), the gross liftoff and the propellant weights were slightly higher than for the J-2 installation because the SSMEs operate at a mixture ratio of 6.0 (compared to 5.5 for the J-2s). A maximum relative velocity of about 2225 m/sec (7300 ft/sec) was achieved at a burnout altitude of 61K m (200K ft) without exceeding the afore-mentioned maximum values of acceleration, q, and $q\alpha$ (see figure 8). Ninety-one percent of maximum sea level thrust was used at liftoff; at an altitude of 1220 m (4000 ft), both engines were throttled to 80 percent with subsequent throttling to 50 percent to constrain acceleration to < 3. Burnout flight-path angle increased with burnout altitude, reaching 33$^\circ$ at 61 km (200K ft).

With SSME installations (2 or 3 engines) and with internal propellant tanks, the engine gimbal points were shifted upward so that with the thrust directed through the center of gravity and with full gimbal capability, the lower portion of the exit plane of the nozzle cleared the body flap. No additional weight to account for this modification was included. With the removal of the upper SSME, the center of gravity moved forward resulting in the entry and landing c.g. locations at about 0.64$\ell$ (figure 1(a)); again, use of ballast would be anticipated to shift the c.g. aft. The entry and landing weights of 66.7K kg (147K lb) and 64.9K kg (143K lb), respectively, were about 4.5K kg (10K lb) below those of the baseline orbiter without payload.

Using three SSMEs at liftoff with full sea level thrust and a slightly lower liftoff T/W (~1-1/4), the allowable GLOW was 401.4K kg (885K lb). The majority of the GLOW increase went into ascent propellant. Since the payload-bay volume of approximately 283 m$^3$ (~10K ft$^3$) accommodated only about 90.7K kg (200K lb) of propellant, all of the ~318K kg (~700K lb) of LOX-hydrogen was contained in an external tank 43.3 m (142 ft) long (figure 9); an off-loaded baseline external tank could also be considered. With this propellant loading, a relative velocity at burnout in excess of 4875 m/sec (16K ft/sec) was achieved (figure 5). The burnout weight (including external tank) was 85.3K kg (188K lb). Burn time was approximately 266 seconds, and burnout occurred approximately 445 km (240 n.mi.) downrange for the three burnout altitudes. Because of the low c.g. location, angles of attack were positive throughout the ascents and never exceeded 15$^\circ$; the maximum $q\alpha$ was constrained to 120K N/m$^2$-deg (2500 lb/ft$^2$-deg). The flight-path angle at burnout never exceeded 3$^\circ$, and the maximum dynamic pressure was less than 24K N/m$^2$ (500 lb/ft$^2$; trajectory details for burnout at 76.2 km are presented in figure 10. The landing weight of 65.8K kg (145K lb) was approximately the same as for the two-SSME, internal-tank version (figure 1(a)). With the third SSME installed, the center-of-gravity location of 0.67$\ell$ was the same as for the baseline orbiter.

In considering the use of a smaller external tank (in lieu of an off-loaded baseline tank), the cost advantage resulting from easier recovery, inspection, and reuse for the tank must be balanced against the requirement for separate design, construction, and man-rating. Obviously, a trade study would be required prior to making a decision on tank size.

## Air Launched

The performance capability of the air-launched orbiter retrofitted with a single J-2 rocket engine and propellant tanks within the payload bay (figure 11) is summarized in figure 12 and Table II with the trajectory details presented in figures 13 to 15. This mode of operation was believed to be the prime candidate for an early, supersonic flight-test program using the production orbiter. Air launch and rocket-engine ignition was assumed at an altitude of 9145 km (30K ft), a velocity of 122 m/sec (400 ft/sec), and a flight path angle of 0$^o$. Using POST, trajectories were calculated using 22.7K, 45.4K, and 68.0K kg (50K, 100K, and 150K lb) of propellant with burnout altitudes of 22.9, 30.5, and 38.1 km (75K, 100K, and 125K ft), respectively. Relative velocities at burnout of about 610, 1220, and 1650 m/sec (1950, 3950, and 5360 ft/sec) were obtained for the three propellant loadings. The maximum dynamic pressure was 12,690 N/m$^2$ (265 lb/ft$^2$), and maximum $q\alpha$ was about 124,500 N/m$^2$-deg (2600 lb/ft$^2$-deg). The maximum flight path angle decreased as the propellant loading increased and never exceeded 57$^o$ with a maximum value at burnout of -8$^o$. Using one J-2, the accelerations were $< 2$ g's. If two J-2s were used, the increase in T/W would reduce the drag and gravity losses which constituted a significant portion of the ideal velocity increment (figure 12). A higher burnout velocity would probably be obtained at the same burnout altitude; however, the maximum acceleration of three g's would be exceeded unless thrust modulation was utilized.

The same limitations and concerns expressed previously with regard to the start, operation, and shutdown of the J-2 apply to the air-launched version; the starting and shutdown loads must be taken by the orbiter structure and would be less than for sea-level starts because of reduced back pressure.

## FLIGHT-TEST RANGE CONSIDERATIONS

A preliminary analysis of the use of the Flight Research Center test range was conducted. For the vertical takeoff case, a launch complex was assumed at Rogers Lake; and, because of potentially high flight velocities and long distances, flights up the test range toward Bonneville were assumed. An indication of the extent of the focused sonic-boom footprint during the pitchover maneuver following VTO is presented in figure 16 for the J-2 ascent. Overpressures (excluding plume effects) in excess of 100 $N/m^2$ (2.1 $lb/ft^2$) occur about 55 km (30 miles) downrange, and care would have to be exercised to avoid the populated areas by use of trajectory and/or azimuth adjustments.

Because of the operational complexity and installation cost and time to provide facilities, the vertical takeoff concept may be impractical for shuttle development. The air-launched concept requires a "mother" aircraft; prior programs at FRC have demonstrated the versatility of this approach in utilizing the test range for both nominal and aborted missions. Launch location and altitude can be selected dependent upon mission requirements (range, abort landing site locations, etc.). Gradual maximum performance buildup is possible with this approach, and higher relative velocities are reached for a given propellant weight; also, the focused sonic boom issue is avoided since no pitchover is required. This mode is preferred over the vertical takeoff.

## ISSUES IDENTIFIED

Several issues have been identified that were beyond the scope of this study but warrant attention in a detailed feasibility evaluation. Because internal tanks in the payload bay contain large amounts of cryogenic propellant well in excess of the maximum payload weight for which the orbiter was designed, a complete structural analysis would have to be made. For the air-launch mode, normal accelerations during pitchup following engine ignition result in load distributions and moments in the orbiter structure unlike the baseline shuttle ascent. The safety aspects of internally-located cryogenic propellants would have to be carefully considered, also.

The mounting and installation of J-2 engines upon a thrust structure designed for SSMEs requires modification. Because the length of the J-2 is less than the SSME, extensions are envisioned for J-2 installation to retain the nozzle exit planes in locations similar to that for the SSMEs. For SSME operation with internal propellant, raising the SSMEs to clear the body flap appears desirable, dependent upon gimbal requirements.

The starting and operational capability of the liquid rockets (J-2 and SSME) in these off-design conditions must be considered for the installations envisioned. Of particular significance are the J-2 low-altitude starting and shutdown transient loads resulting from asymmetric flow separation within the J-2 nozzles.

Ascent flight-control requirements and the magnitude of thrust vectoring would have to be determined. The use of aerodynamic control during ascent should be considered to alleviate the thrust vector requirements.

## CONCLUDING REMARKS

A study has been completed that identifies the potential performance capability of the production shuttle orbiter for powered flight tests using several propulsion systems following vertical takeoff and using J-2 rocket engines following air launch. Of the approaches considered, it was concluded that the air-launched orbiter equipped with J-2 rocket engines has the highest potential to provide a capability of early shuttle development flight tests. A potential of Mach 4 using 45.4K kg (100K lb) of internal propellant was shown. Several issues were identified that require resolution to prove feasibility.

## REFERENCE

1. Cornick, Douglas E.; Stevenson, Robert; Brauer, Garry L.; and Steinhoff, R. Terry: POST - Program to Optimize Shuttle Trajectories. NASA CR-125584, 1971.

TABLE I
WEIGHT SUMMARY
(a) S.I. Units (kg)

| SYSTEM | R.I. ORBITER | V.T.O. 3 J-2's | V.T.O. 2 SSME's | V.T.O. 3 SSME's | AIR LAUNCHED, 3 J-2's: 22.7K kg PROP | AIR LAUNCHED, 3 J-2's: 45.4K kg PROP | AIR LAUNCHED, 3 J-2's: 68.0K kg PROP |
|---|---|---|---|---|---|---|---|
| WING | 5084 | 5084 | 5084 | 5084 | 5084 | 5084 | 5084 |
| TAIL | 1210 | 1210 | 1210 | 1210 | 1210 | 1210 | 1210 |
| BODY | 16125 | 16397 | 16125 | 16125 | 16398 | 16398 | 16397 |
| INDUCED ENVIR PROT | 11414 | 11414 | 11414 | 11414 | 11414 | 11414 | 11414 |
| LND, DOCK, REC. | 4310 | 4310 | 4310 | 4310 | 4310 | 4310 | 4310 |
| PROP ASCENT | 11994 | 9755 | 10943 | 11994 | 7370 ** | 8147 ** | 8963 ** |
| PROP AUXILLARY | 2369 | 1689 | 1689 | 1689 | 1689 | 1689 | 1689 |
| PRIME POWER | 1540 | 877 | 877 | 877 | 877 | 877 | 877 |
| ELECT CONVER & DISTR<br>HYDR CONVER & DISTR | 2840 | 2433 | 2433 | 2433 | 2433 | 2433 | 2433 |
| SURFACE CONTROLS | 833 | 833 | 847 | 847 | 833 | 833 | 833 |
| AVIONICS | 2051 | 1369 | 1369 | 1369 | 1369 | 1369 | 1369 |
| ENVIR CONTROL | 1940 | 1134 | 1134 | 1134 | 1134 | 1134 | 1134 |
| PERSONNEL PROV. | 612 | 612 | 612 | 612 | 612 | 612 | 612 |
| PAYLOAD ACCOM. | 22 | 0 | 0 | 0 | 0 | 0 | 0 |
| MARGIN | 5696 | 5221 | 5172 | 4875 | 4982 | 5059 | 5142 |
| DRY WEIGHT | 68040 | 62338 | 63219 | 63973 | 59715 | 60569 | 61467 |
| PERSONNEL | | 553 | 553 | 553 | 553 | 553 | 553 |
| ACPS RESERVES | | 91 | 91 | 91 | 91 | 91 | 91 |
| RESIDUALS | | 2280 | 1134 | 1179 | 2280 | 2280 | 2280 |
| LANDING WEIGHT | 70217 | 65262 | 64997 | 65796 | 62639 | 63493 | 64391 |
| ACPS PROP (ENTRY) | 1270 | 1635 | 1635 | 1635 | 1635 | 1635 | 1635 |
| ENTRY WEIGHT | 71487 | 66897 | 66632 | 67431 | 64274 | 65128 | 66026 |
| INFLIGHT LOSSES | | 454 | 454 | 680 | 116 | 231 | 340 |
| ASCENT PROP | | 88892 | 93634 | 333273* | 22680 | 45360 | 68040 |
| GROSS WEIGHT | | 156243 | 160720 | 401384 | 87070 | 110719 | 134406 |

* Includes Weight of External Tank 17730 kg.

** Includes Weight of 3 J-2's or 1 J-2 + 3133 kg Ballast.

TABLE I
WEIGHT SUMMARY
(b) U.S. Customary Units (lb)

| SYSTEM | R.I. | V.T.O. | | | AIR LAUNCHED, 3 J-2's | | |
|---|---|---|---|---|---|---|---|
| | 150K ORBITER | 3 J-2's | 2 SSME's | 3 SSME's | 50K LB. PROP. | 100K LB. PROP. | 150K LB PROP |
| WING | 11208 | 11208 | 11208 | 11208 | 11208 | 11208 | 11208 |
| TAIL | 2669 | 2669 | 2669 | 2669 | 2669 | 2669 | 2669 |
| BODY | 35540 | 36140 | 35548 | 35548 | 36148 | 36148 | 36148 |
| INDUCED ENVIR PROT | 25164 | 25164 | 25164 | 25164 | 25164 | 25164 | 25164 |
| LND, DOCK, REC. | 9501 | 9501 | 9501 | 9501 | 9501 | 9501 | 9501 |
| PROP ASCENT | 26441 | 21505 | 24125 | 26441 | 16247** | 17960** | 19760** |
| PROP AUXILIARY | 5223 | 3723 | 3723 | 3723 | 3723 | 3723 | 3723 |
| PRIME POWER | 3395 | 1934 | 1934 | 1934 | 1934 | 1934 | 1934 |
| ELECT CONVER & DISTR<br>HYDR CONVER & DISTR | 6260 | 5364 | 5364 | 5364 | 5364 | 5364 | 5364 |
| SURFACE CONTROLS | 1637 | 1637 | 1637 | 1637 | 1637 | 1637 | 1637 |
| AVIONICS | 4522 | 3019 | 3019 | 3019 | 3019 | 3019 | 3019 |
| ENVIR CONTROL | 4276 | 2500 | 2500 | 2500 | 2500 | 2500 | 2500 |
| PERSONNEL PROV. | 1350 | 1350 | 1350 | 1350 | 1350 | 1350 | 1350 |
| PAYLOAD ACCOM. | 48 | 0 | 0 | 0 | | 0 | 0 |
| MARGIN | 12550 | 11507 | 11399 | 10746 | [illegible] | 11153 | 11333 |
| DRY WEIGHT | 150000 | 137429 | 139371 | 141034 | 131646 | 133530 | 135510 |
| PERSONNEL | | 1220 | 1220 | 1220 | 1220 | 1220 | 1220 |
| ACPS RESERVES | | 200 | 200 | 200 | 200 | 200 | 200 |
| RESIDUALS | | 5026 | 2500 | 2600 | 5026 | 5026 | 5026 |
| LANDING WEIGHT | 154800 | 143875 | 143291 | 145054 | 138092 | 139976 | 141956 |
| ACPS PROP (ENTRY) | 2800 | 3605 | 3605 | 3605 | 3605 | 3605 | 3605 |
| ENTRY WEIGHT | 157600 | 147480 | 146896 | 148659 | 141697 | 143581 | 145561 |
| INFLIGHT LOSSES | | 1000 | 1000 | 1500 | 255 | 510 | 750 |
| ASCENT PROP | | 195970 | 206424 | 734728* | 50000 | 100000 | 150000 |
| GROSS WEIGHT | | 344450 | 354320 | 884887 | 191952 | 244091 | 296311 |

* Includes Weight of External Tank 39088 lb.

** Includes Weight of 3 J-2's or 1 J-2 + 6908 lb Ballast.

## TABLE II

### BURNOUT CONDITIONS FOLLOWING AIR LAUNCH ONE J-2 ROCKET, INTERNAL TANKS

| | BURNOUT CONDITIONS | | |
|---|---|---|---|
| PROPELLANT WEIGHT | ALTITUDE | RELATIVE VELOCITY | FLIGHT-PATH ANGLE, DEG. |
| 22.68K kg<br>50K lb | 22.9 km<br>75K ft | 594 m/sec<br>1948 ft/sec | -0.2 |
| 45.36K kg<br>100K lb | 30.5 km<br>100K ft | 1240 m/sec<br>3952 ft/sec | -8.3 |
| 68.04K kg<br>150K lb | 38.1 km<br>125K ft | 1635 m/sec<br>5363 ft/sec | -7.2 |

| | R. I. Orbiter | 1205 SRM External | 1205 SRM Internal | 3 J-2's Internal Tanks | 2 SSME's Internal Tanks | 3 SSME's External Tanks |
|---|---|---|---|---|---|---|
| GLOW, kg | | 287.5K | 287.9K | 156.2K | 160.7K | 401.4K |
| GLOW, lbs. | | 633.8K | 634.6K | 344.5K | 354.3K | 884.9K |
| Entry wt., kg | 71.5K | 56.1K | 89.3K | 66.9K | 66.6K | 67.4K |
| Entry wt., lbs. | 157.6K | 123.7K | 196.8K | 147.5K | 146.9K | 148.7K |
| Entry c.g., x/l | 0.670 | 0.604 | 0.683 | 0.641* | 0.643* | 0.667 |
| Landing wt., kg | 70.2K | 54.5K | 87.6K | 65.3K | 65.0K | 65.8K |
| Landing wt., lbs. | 154.8K | 120.1K | 193.2K | 143.9K | 143.3K | 145.1K |
| Landing c.g., x/l | 0.669 | 0.598 | 0.681 | 0.641* | 0.639* | 0.670 |

*Approximately 15K lb. at rear of vehicle will shift c.g. aft about 0.03 x/L

(a) Vertical takeoff.

Figure 1.- Development flight-test vehicle weight/c.g. summary.

| | R. I. Orbiter | 50K lb Propellant | 100K lb Propellant | 150K lb Propellant |
|---|---|---|---|---|
| GLOW, kg | | 87.1K | 110.7K | 134.4K |
| GLOW, lb | | 192.0K | 244.1K | 296.3K |
| Entry wt., kg | 71.5K | 64.3K | 65.1K | 66.0K |
| Entry wt., lb | 157.6K | 141.7K | 143.6K | 145.4K |
| Entry c.g., $x/l$: | 0.670 | .64 | 0.64 | 0.64 |
| Landing wt., kg. | 70.2K | 62.6K | 63.5K | 64.4K |
| Landing wt., lb | 154.8K | 138.1K | 140.0K | 142.0K |
| Landing c.g., $x/l$: | .669 | 0.64 | 0.64 | 0.64 |

(b) Airlaunched, J-2 powered.

Figure 1.- Concluded.

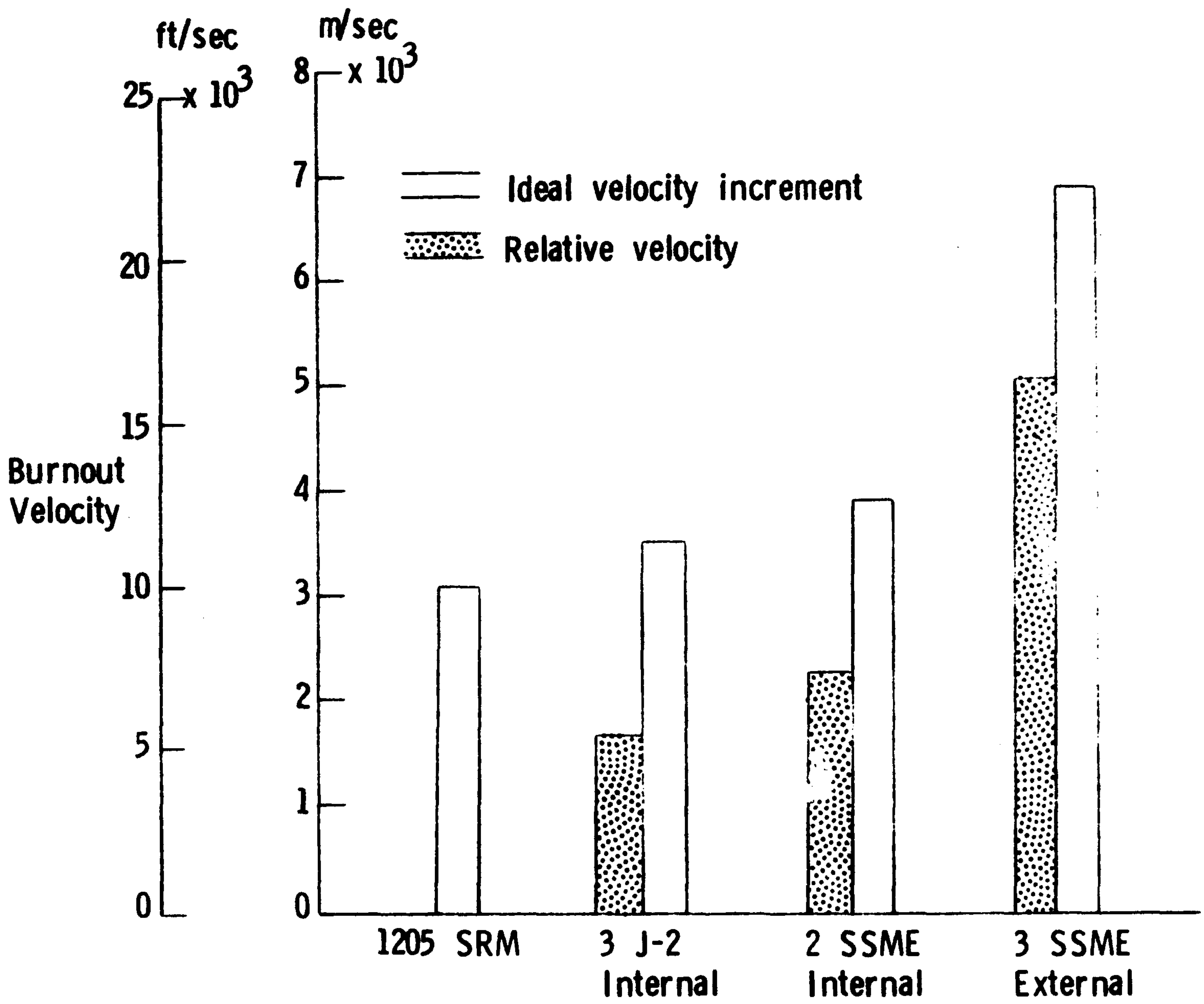
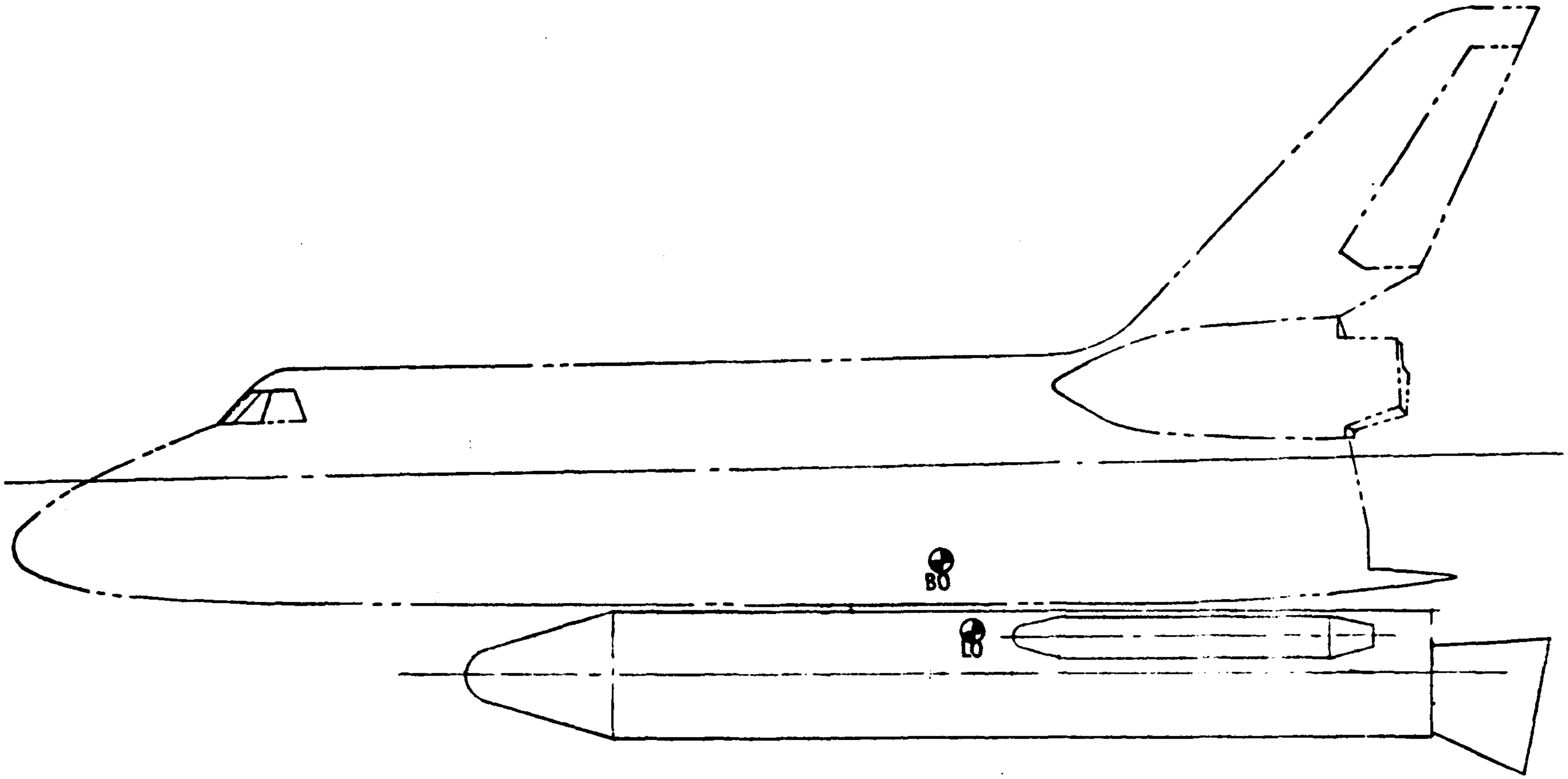


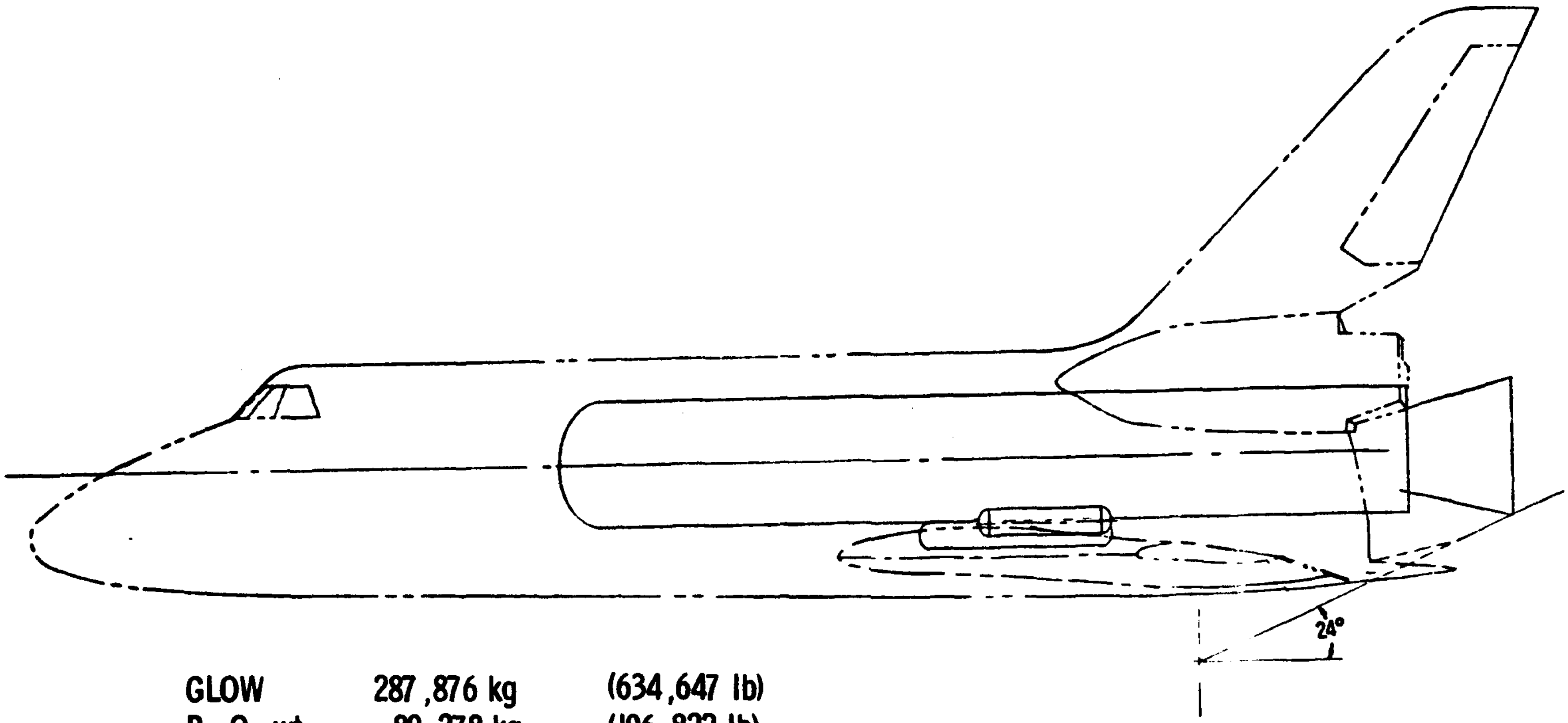

Figure 2.- Vertical takeoff performance.




| | | |
|---|---|---|
| GLOW | 287,474 kg | (633,760 lb) |
| B. O. wt. | 88,876 kg | (195,935 lb) |
| Entry wt. | 56,118 kg | (123,718 lb) |

(a) Externally-mounted.

Figure 3.- Vertical takeoff, 1205 SRM pov




| | | |
|---|---|---|
| GLOW | 287,876 kg | (634,647 lb) |
| B. O. wt. | 89,278 kg | (196,822 lb) |
| Entry wt. | 89,278 kg | (196,822 lb) |

(b) Internally-mounted.

Figure 3.- Concluded.

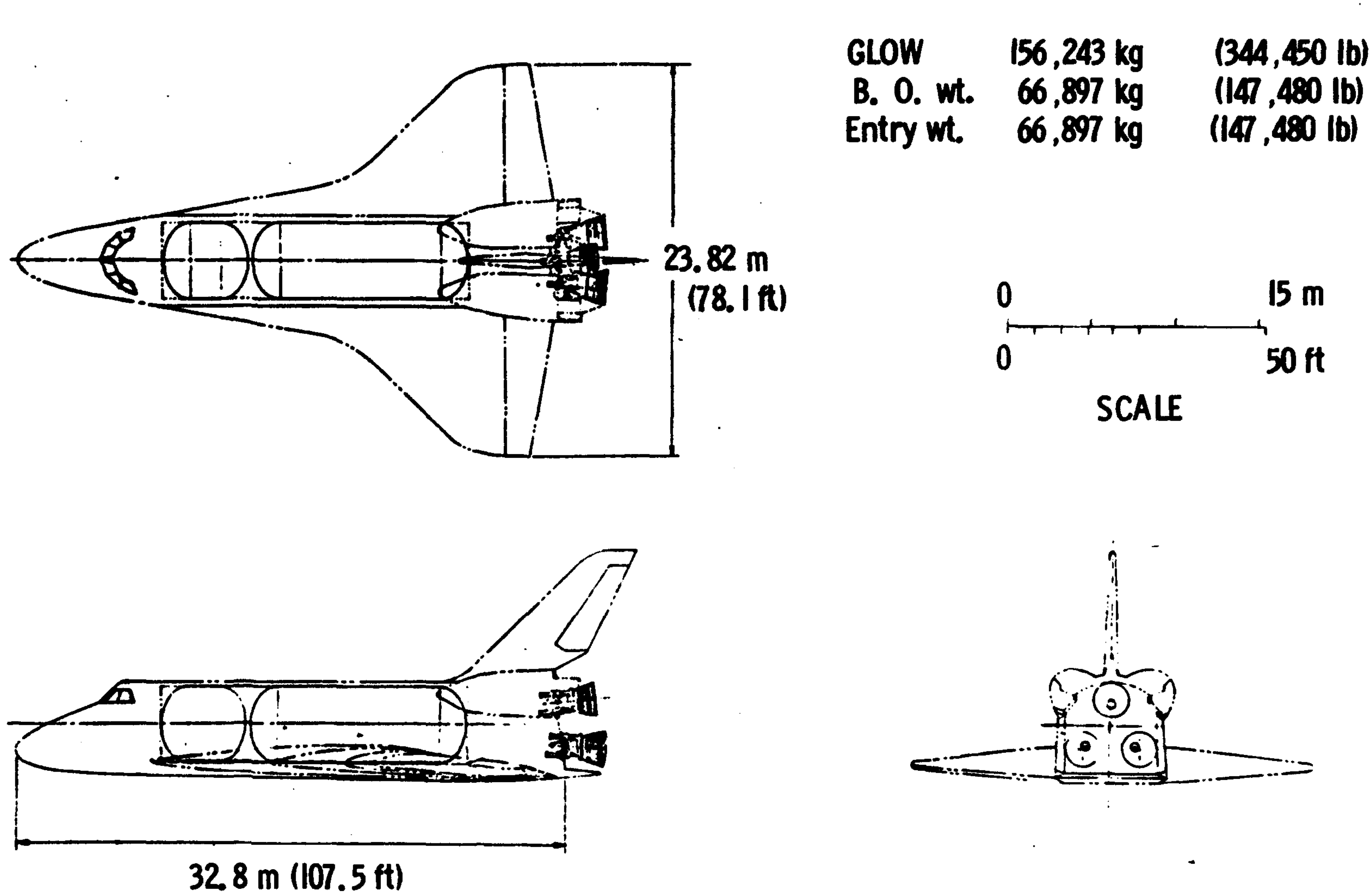



Figure 4.- Vertical takeoff, J-2 powered.

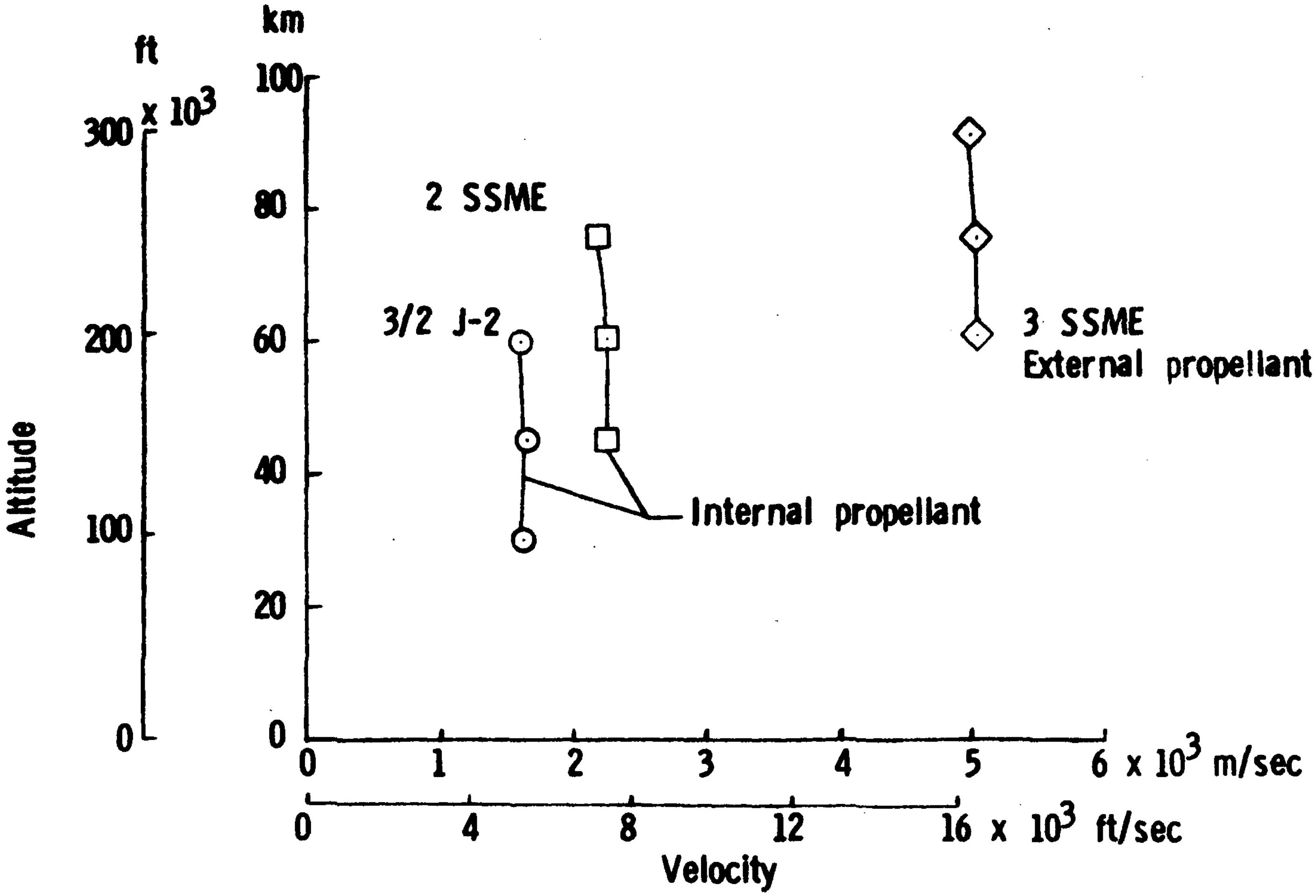
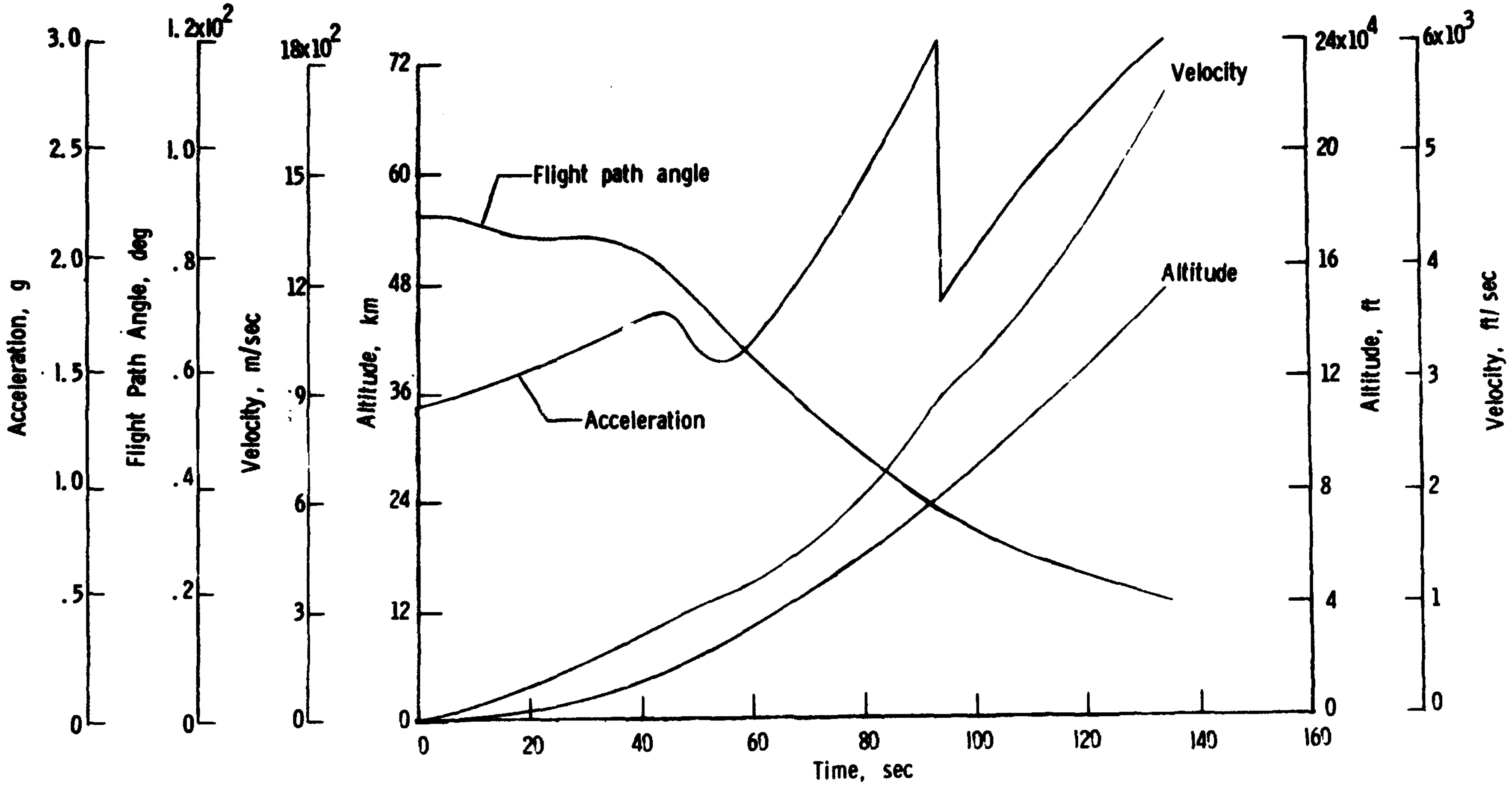
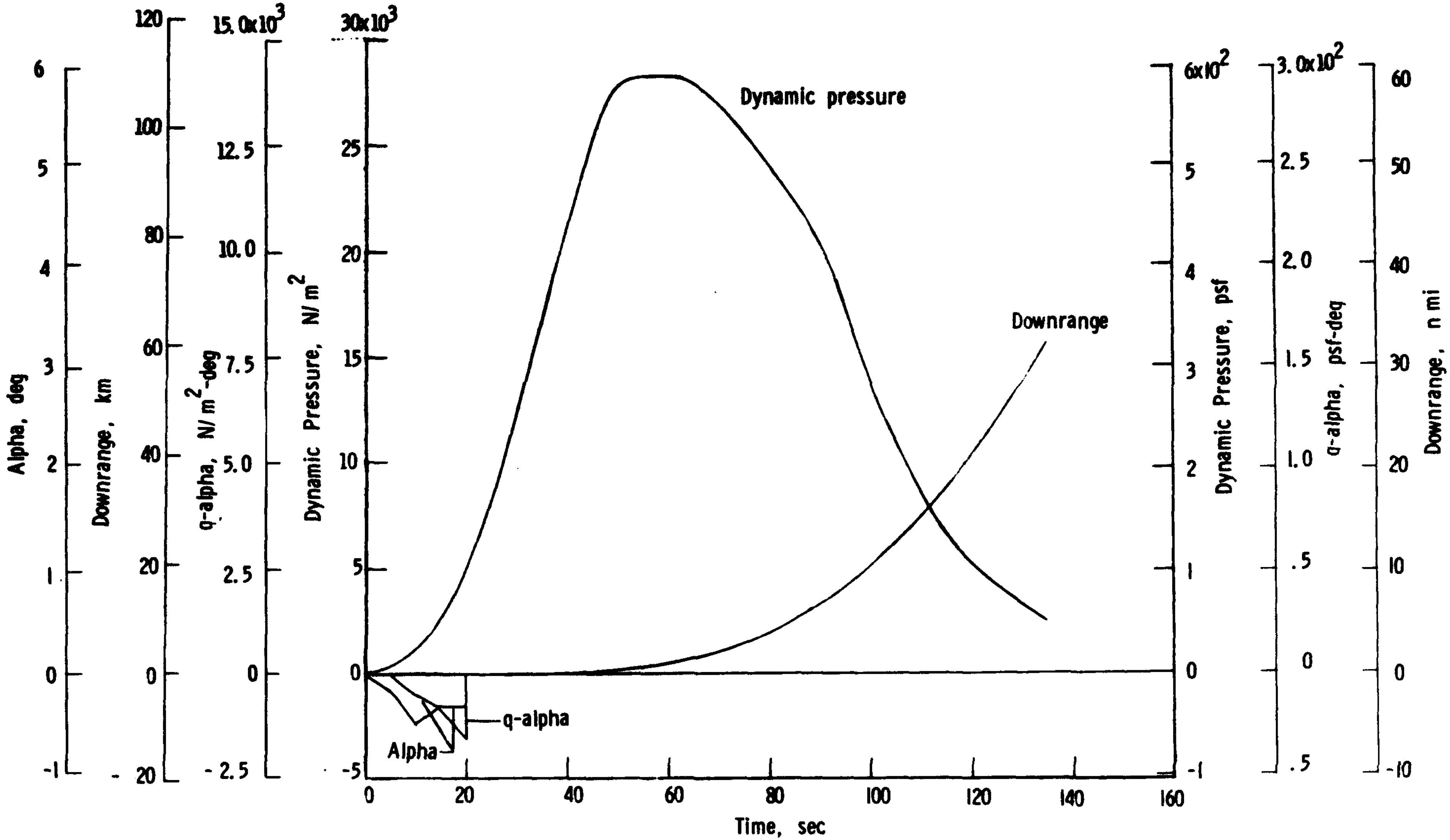



Figure 5.- Burnout conditions following vertical takeoff.




(a) Altitude, velocity, flight path angle, and total acceleration.

Figure 6.- Vertical takeoff with J2 engines.




(b) Angle of attack, dynamic pressure, q-alpha, and downrange.

Figure 6.- Concluded.

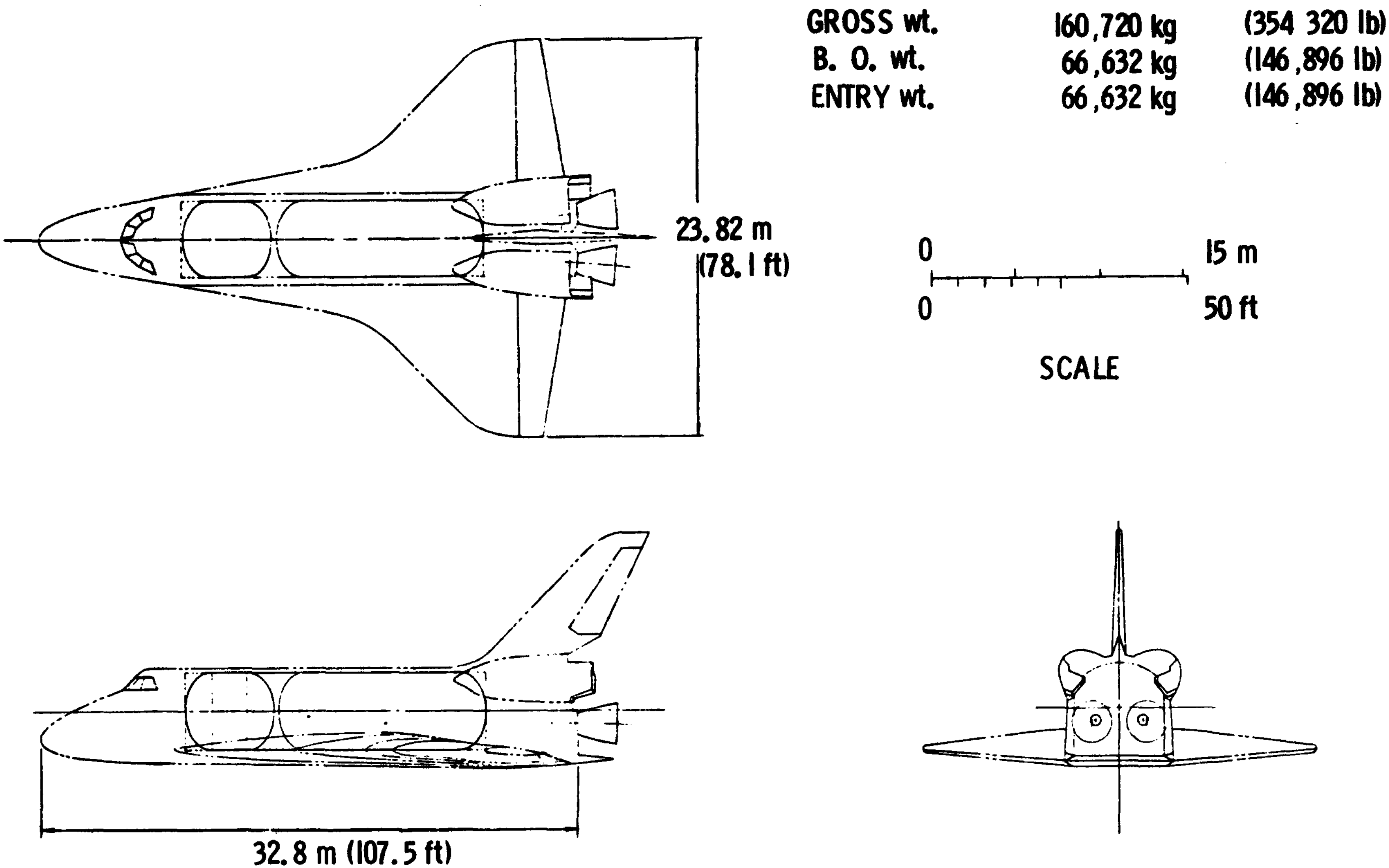
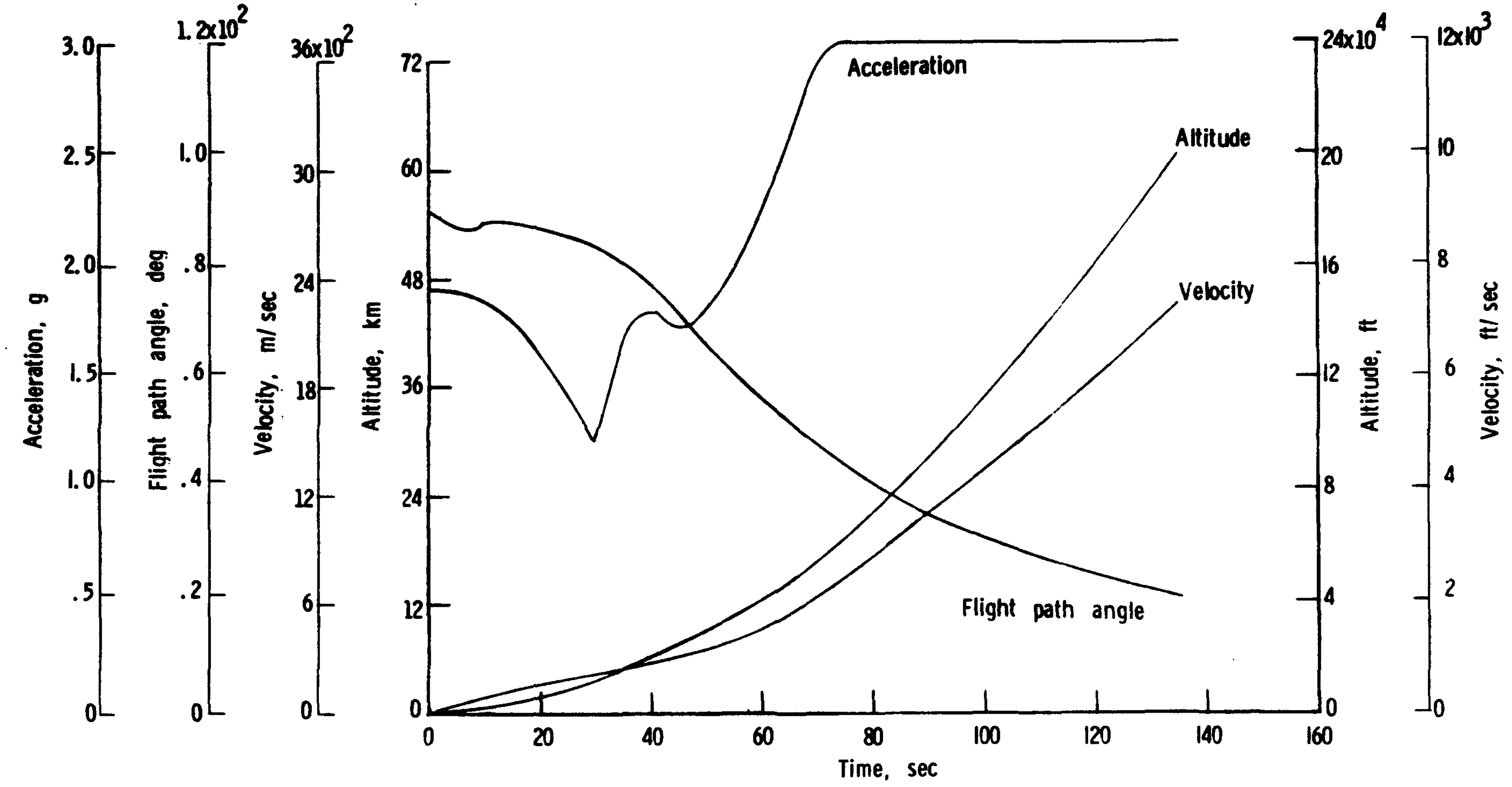
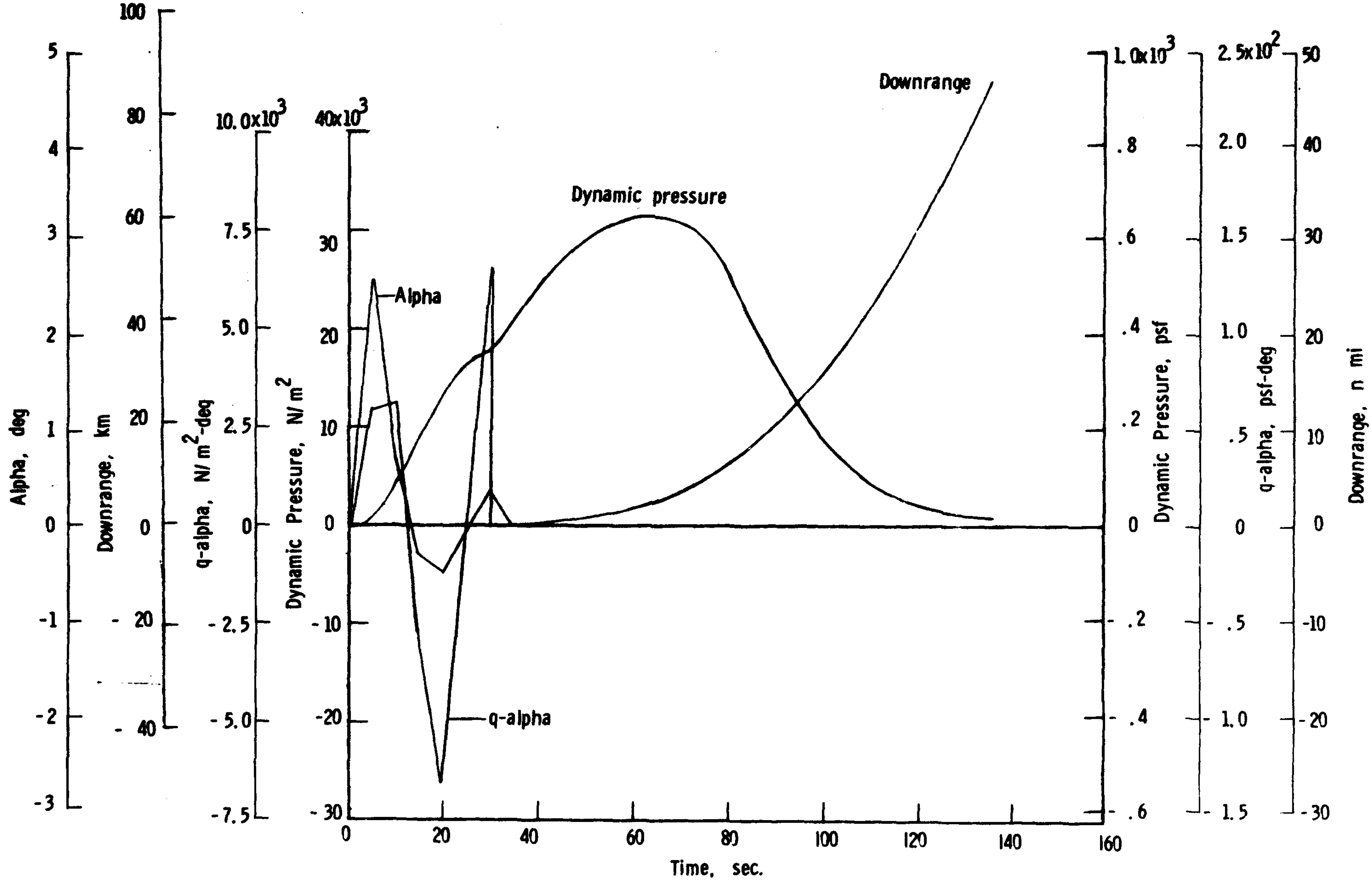



Figure 7.- Vertical takeoff, two SSME powered, internal tanks.




(a) Altitude, velocity, flight path angle, and total acceleration.

Figure 8.- Vertical takeoff with 2 SSME.




(b) Angle of attack, dynamic pressure, q-alpha, and downrange.

Figure 8.- Concluded.

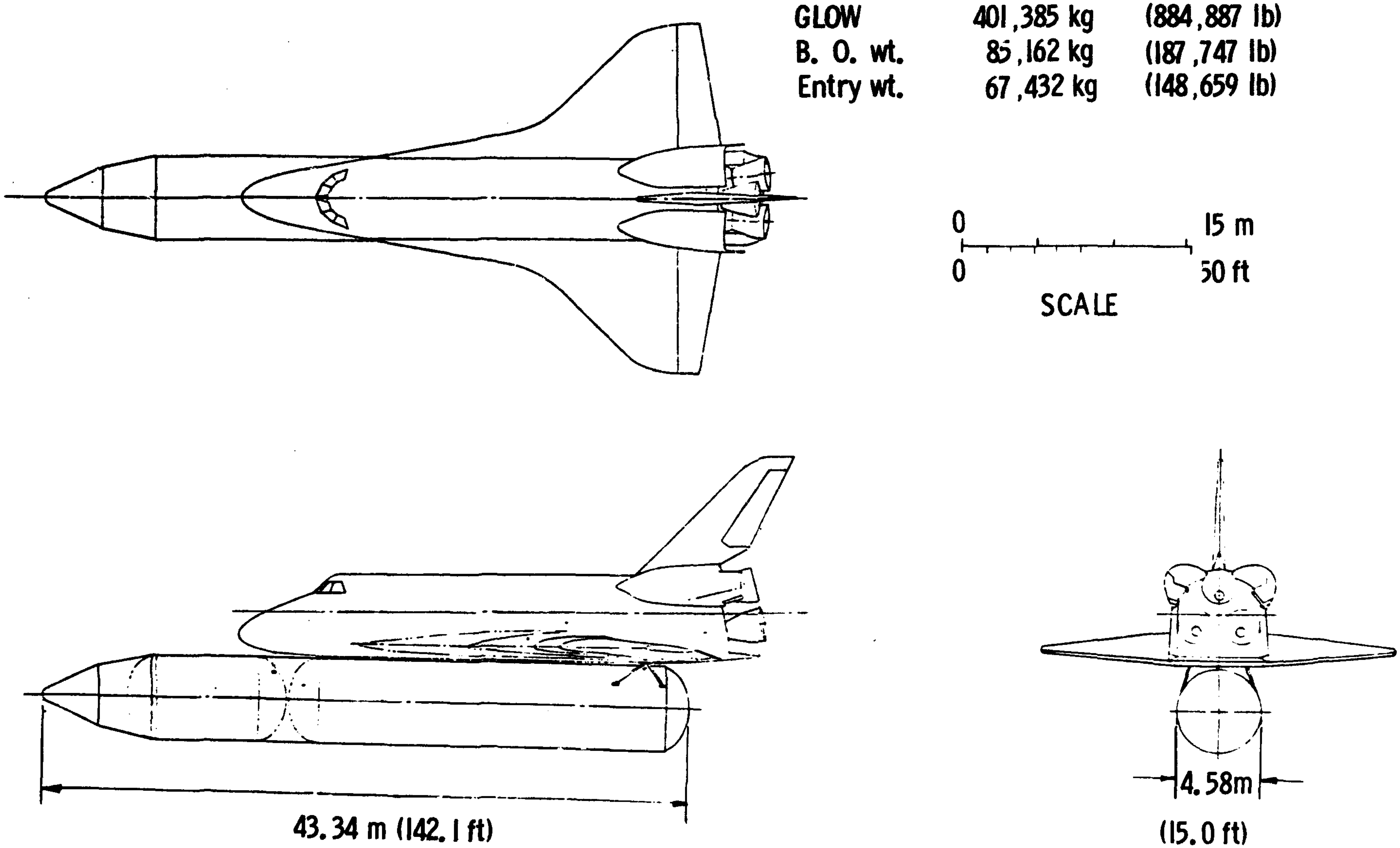
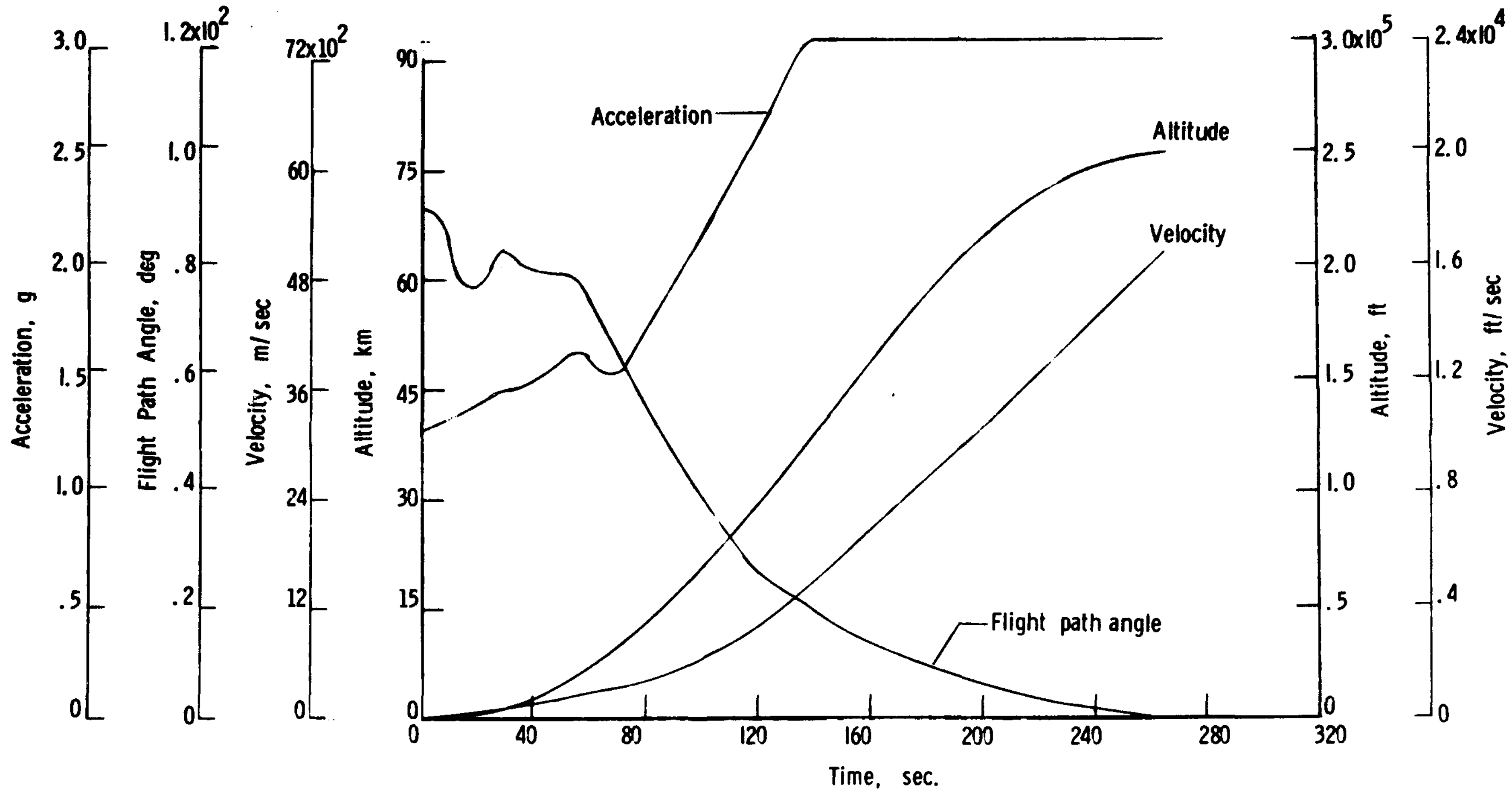
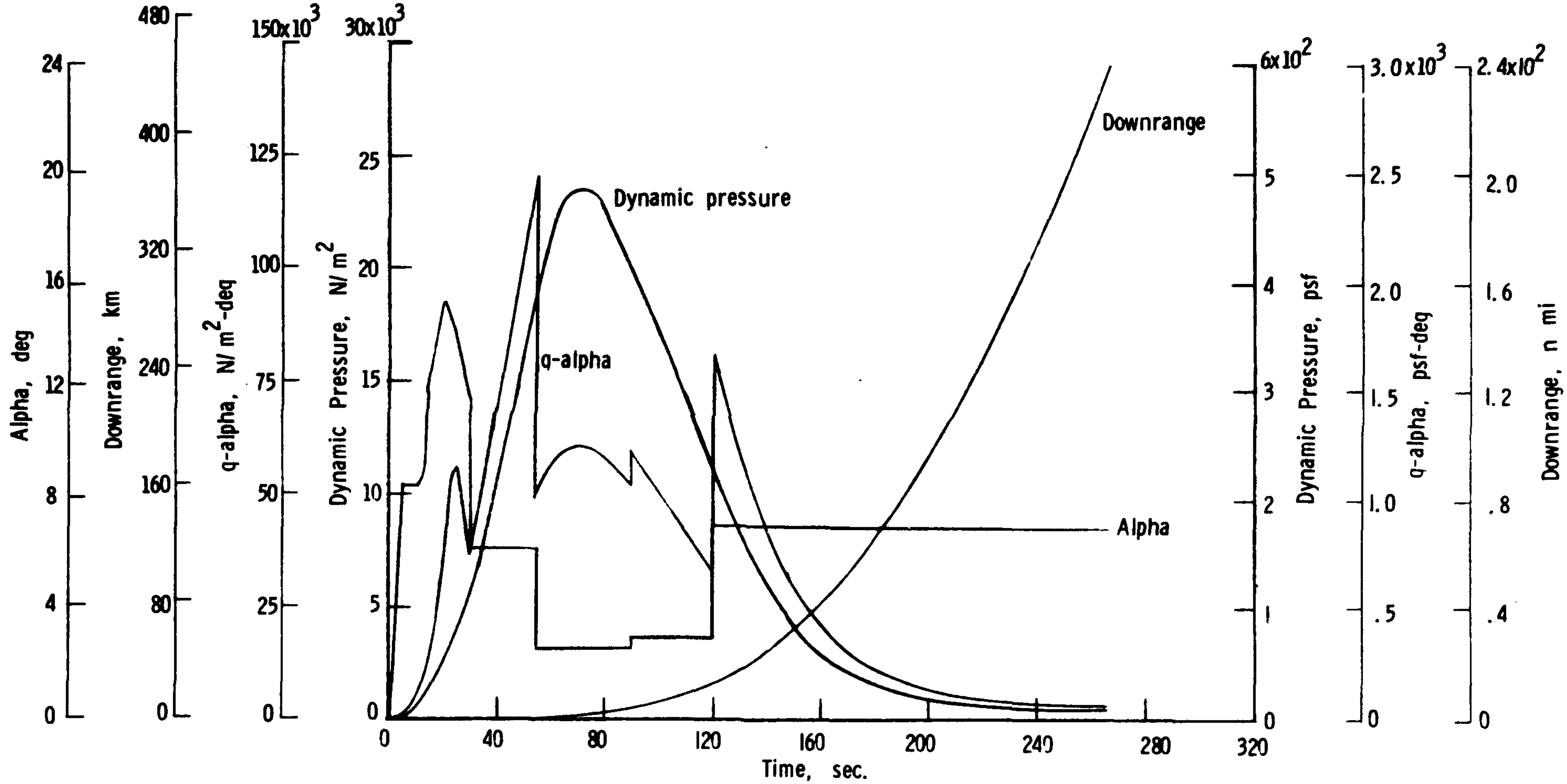



Figure 9.- Vertical takeoff, three SSME, external tanks.




(a) Altitude, velocity, flight path angle, and total acceleration.

Figure 10.- Vertical takeoff with 3 SSME and external tanks.




(b) Angle of attack, dynamic pressure, q-alpha, and downrange.

Figure 10.- Concluded.

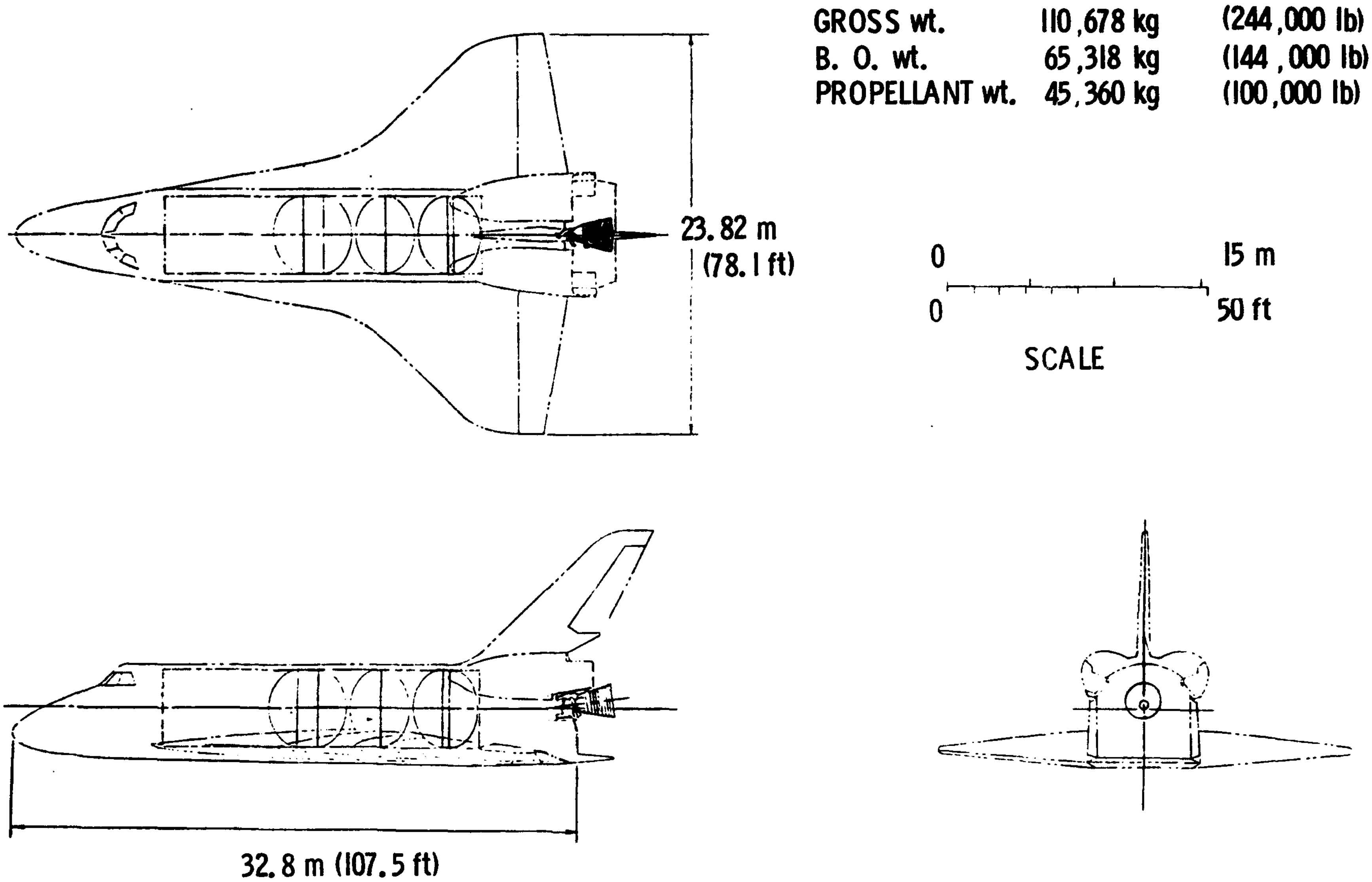



Figure 11.· Air-launched, J-2 powered.

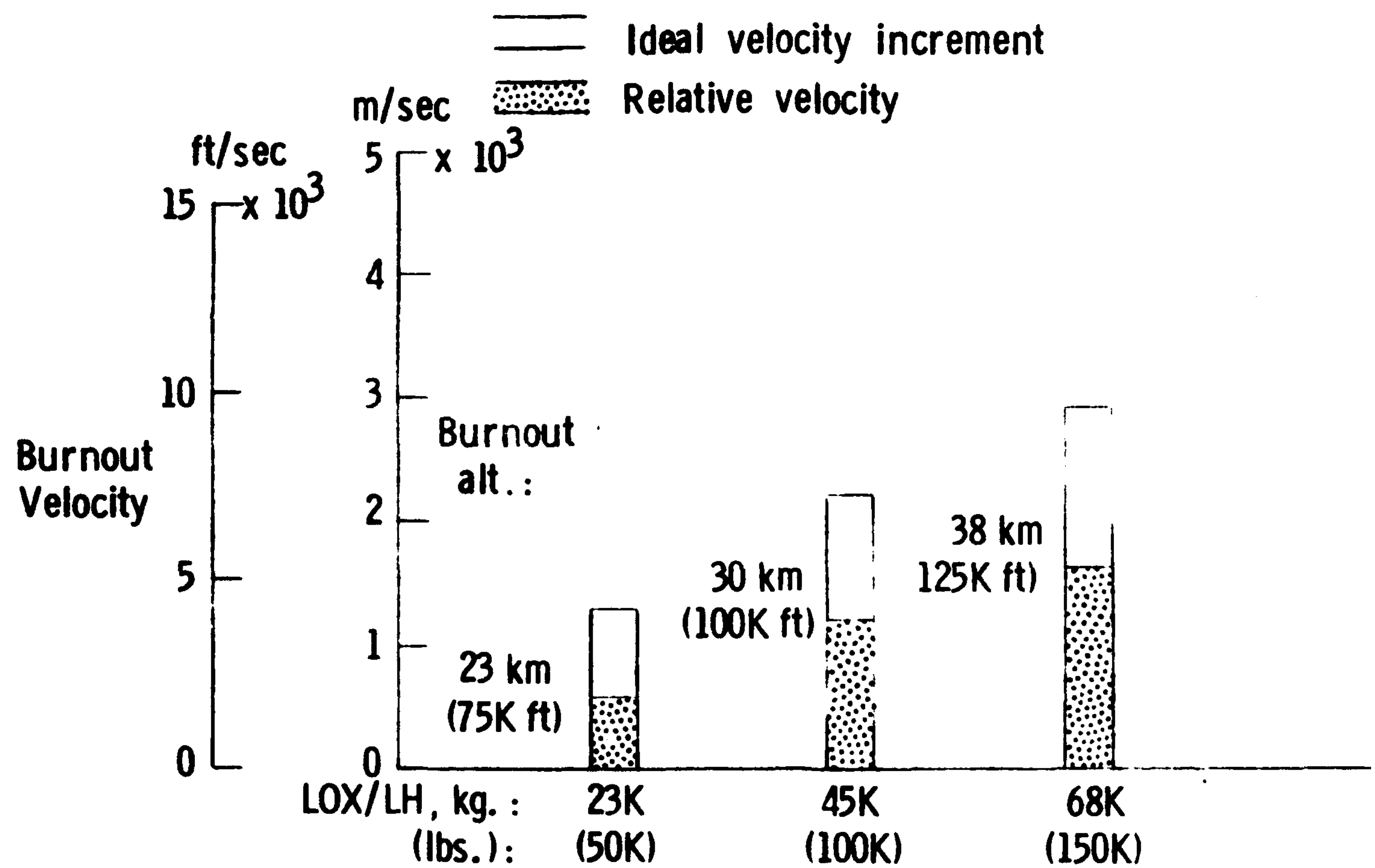
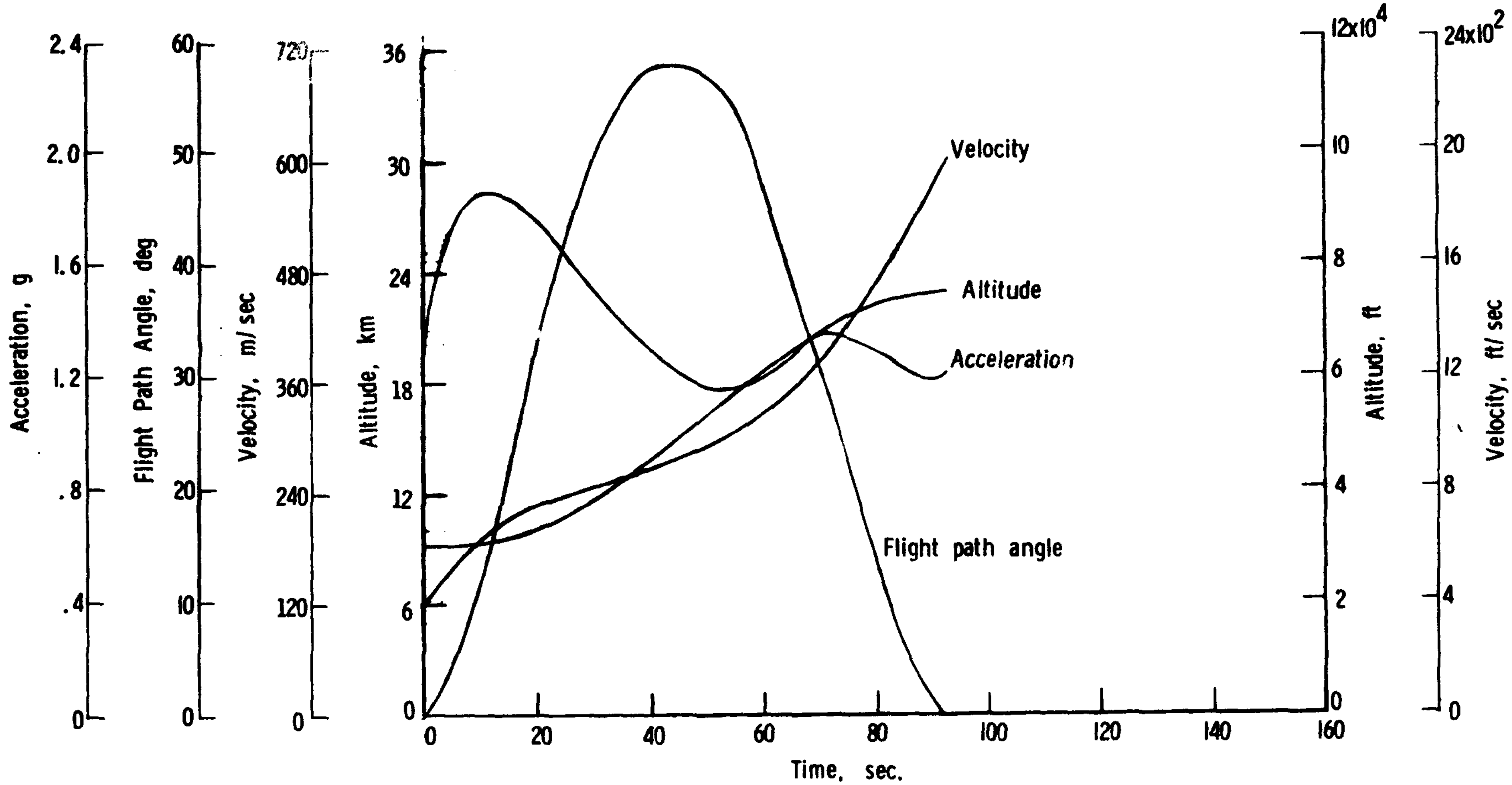
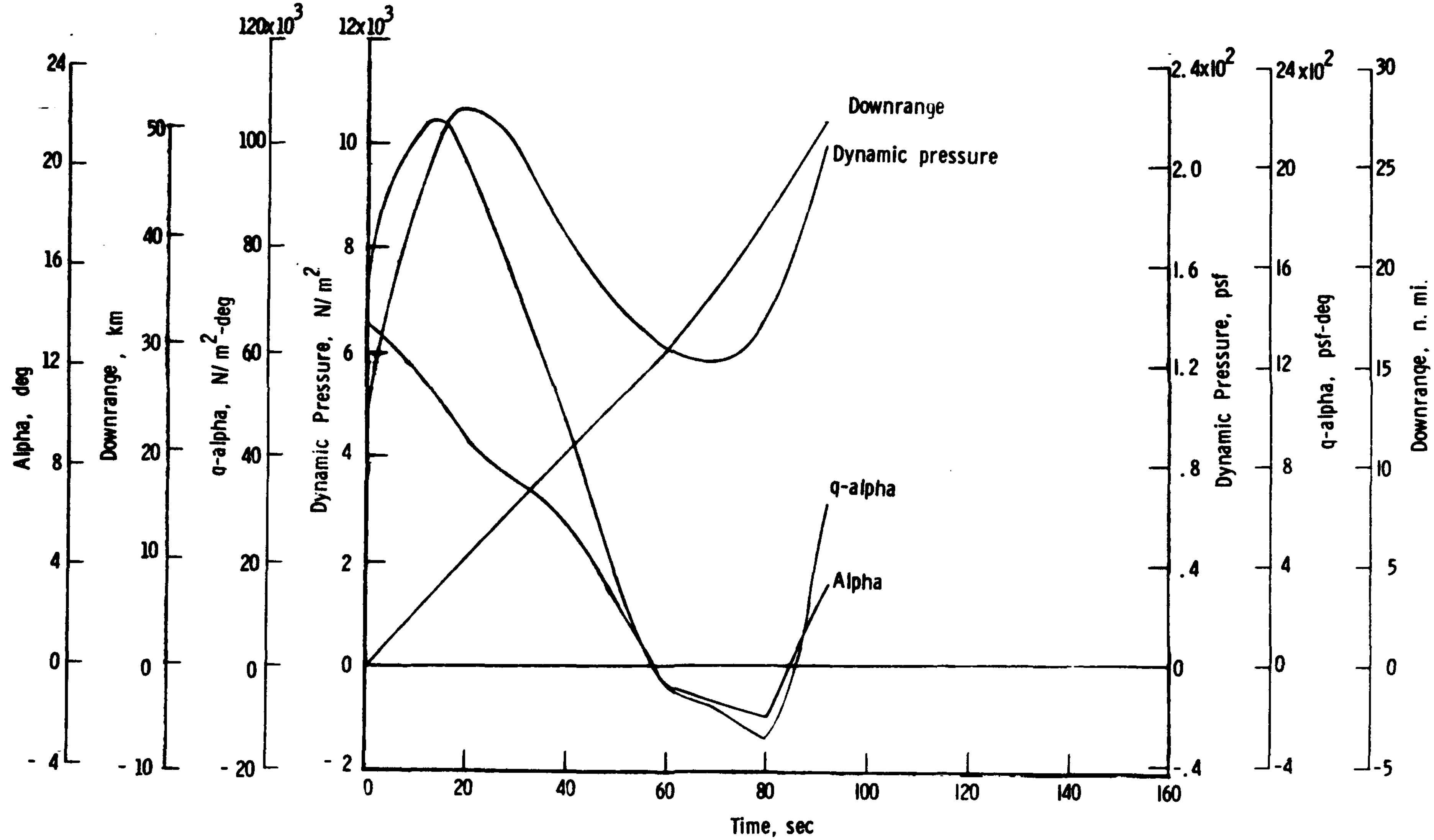



Figure 12.- Air-launched performance, one J-2 powered, internal tanks.



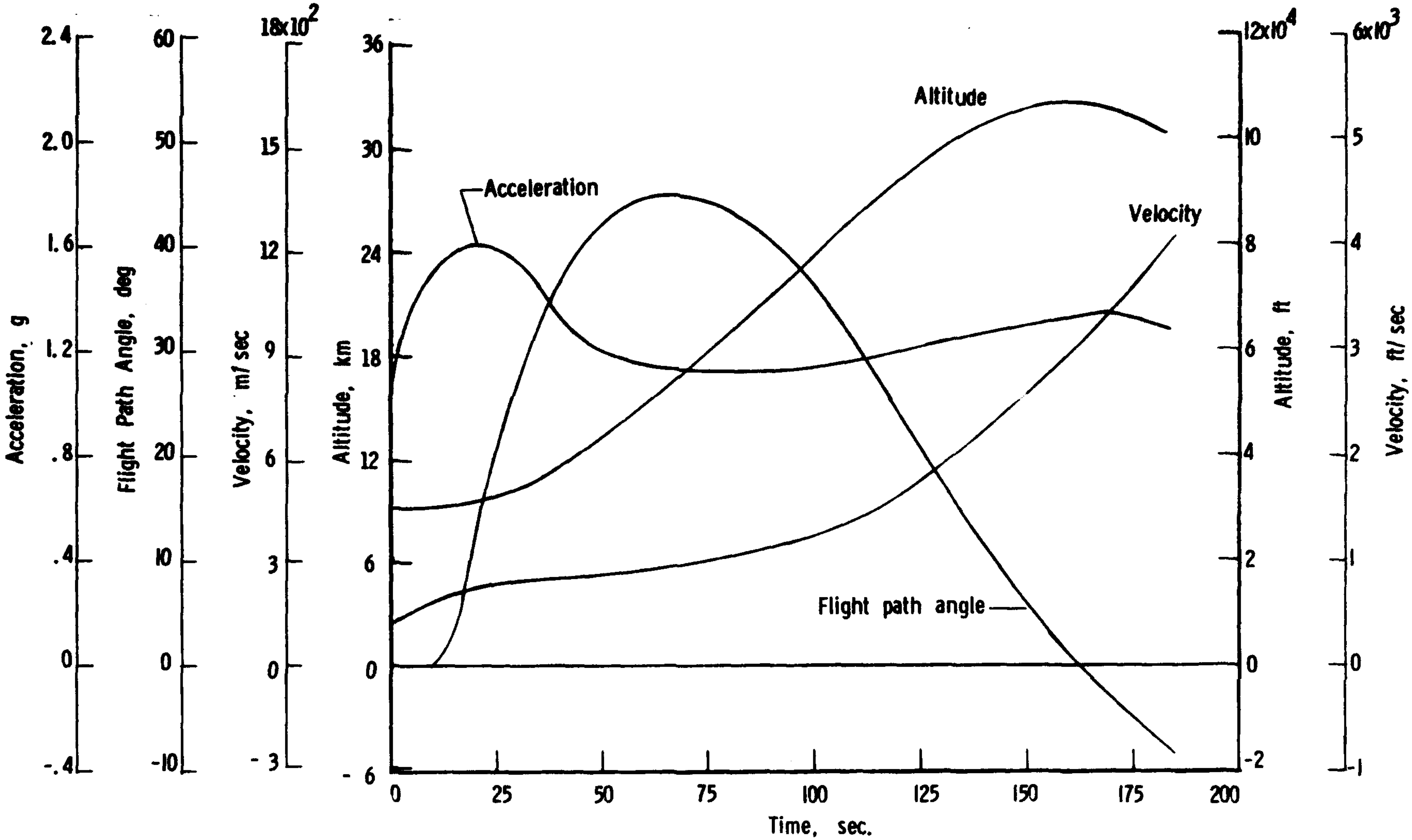

(a) Altitude, velocity, flight path angle, and total acceleration.

Figure 13.- Airlaunched with 22,680 kg (50,000 lb) of propellant, J-2 powered.



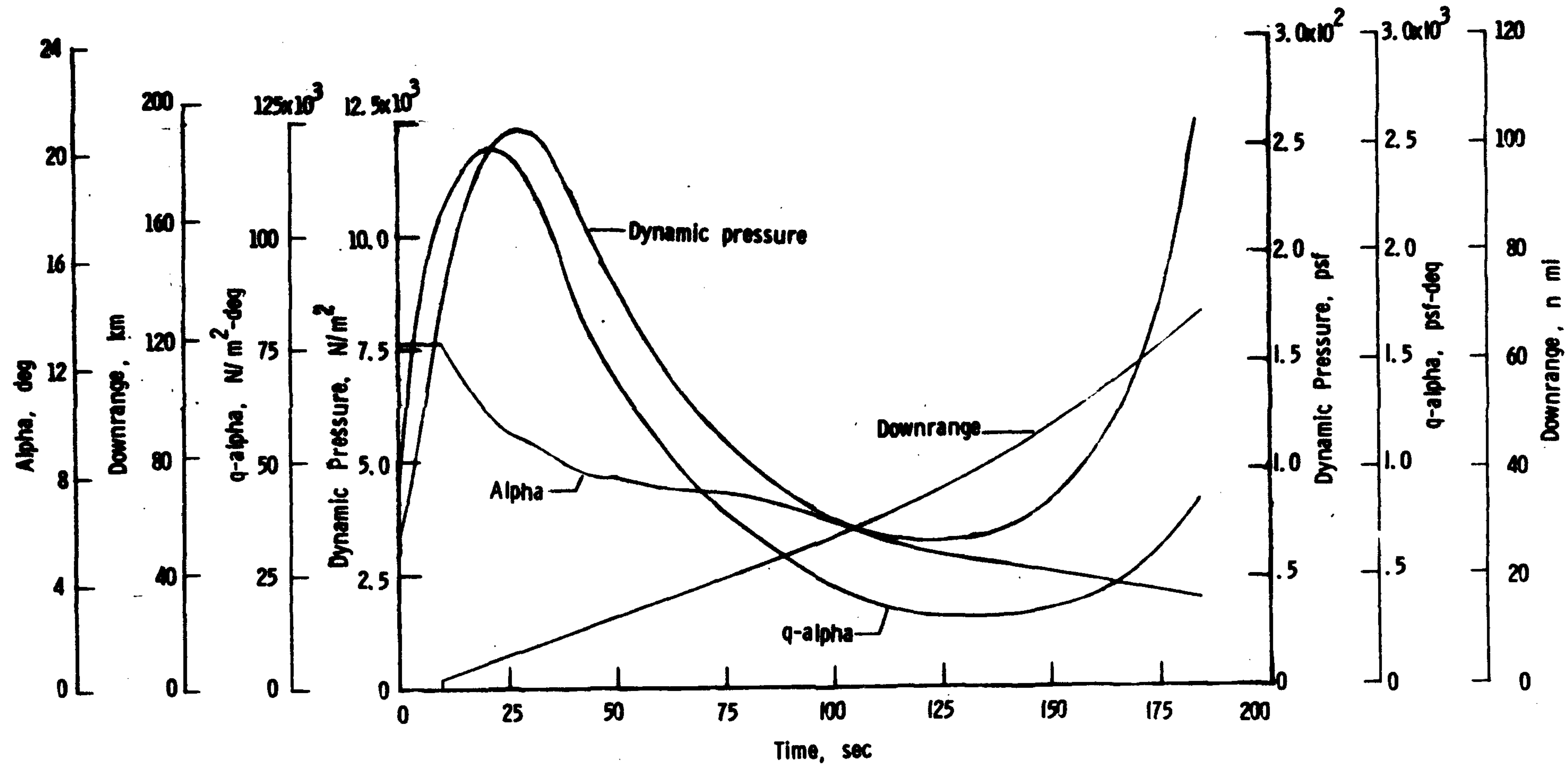

(b) Angle of attack, dynamic pressure, q-alpha, and downrange.

Figure 13.- Concluded.



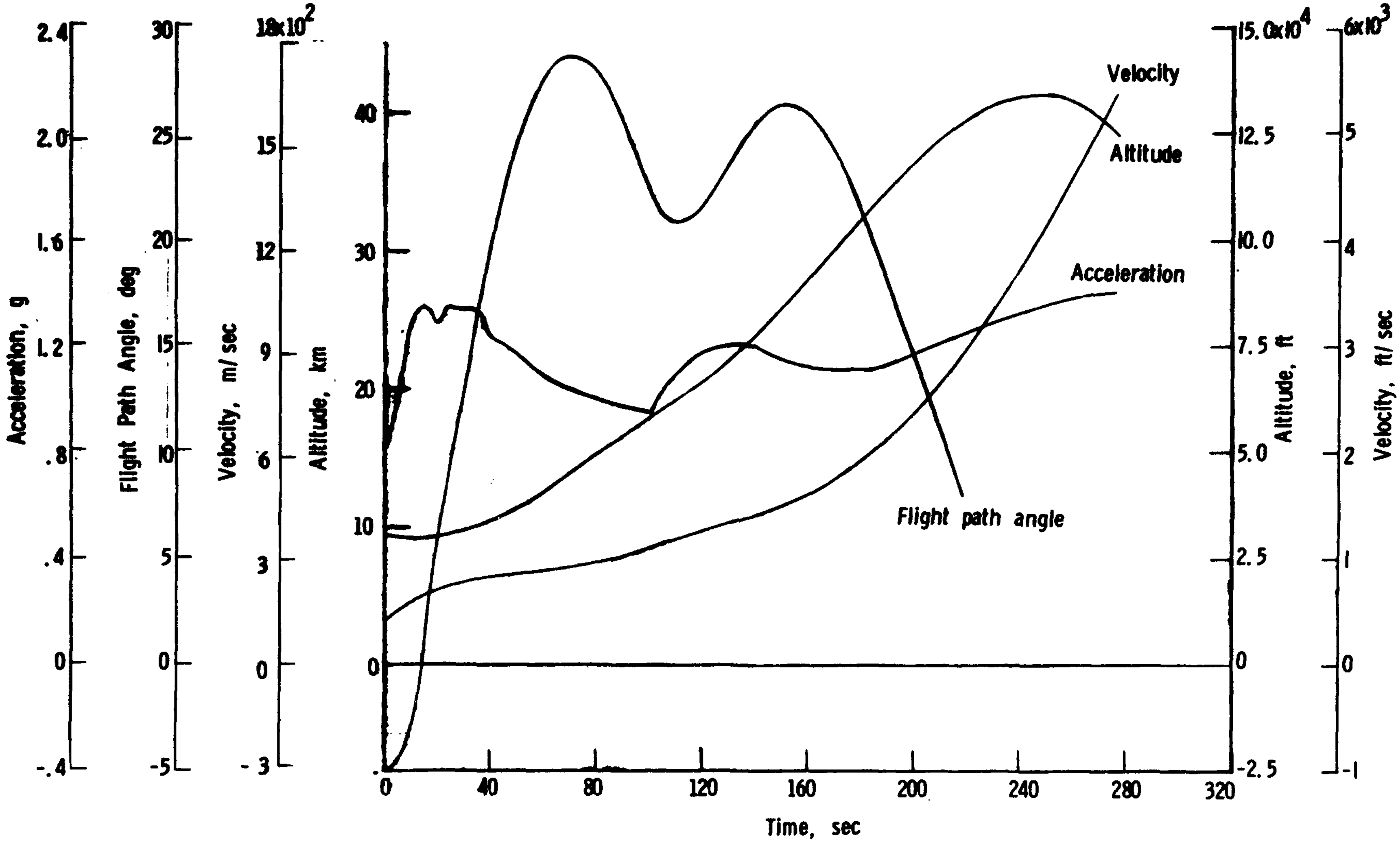

(a) Altitude, velocity, flight path angle, and total acceleration.

Figure 14.- Airlaunched with 45,360 kg (100,000 lb) of propellant, J-2 powered.



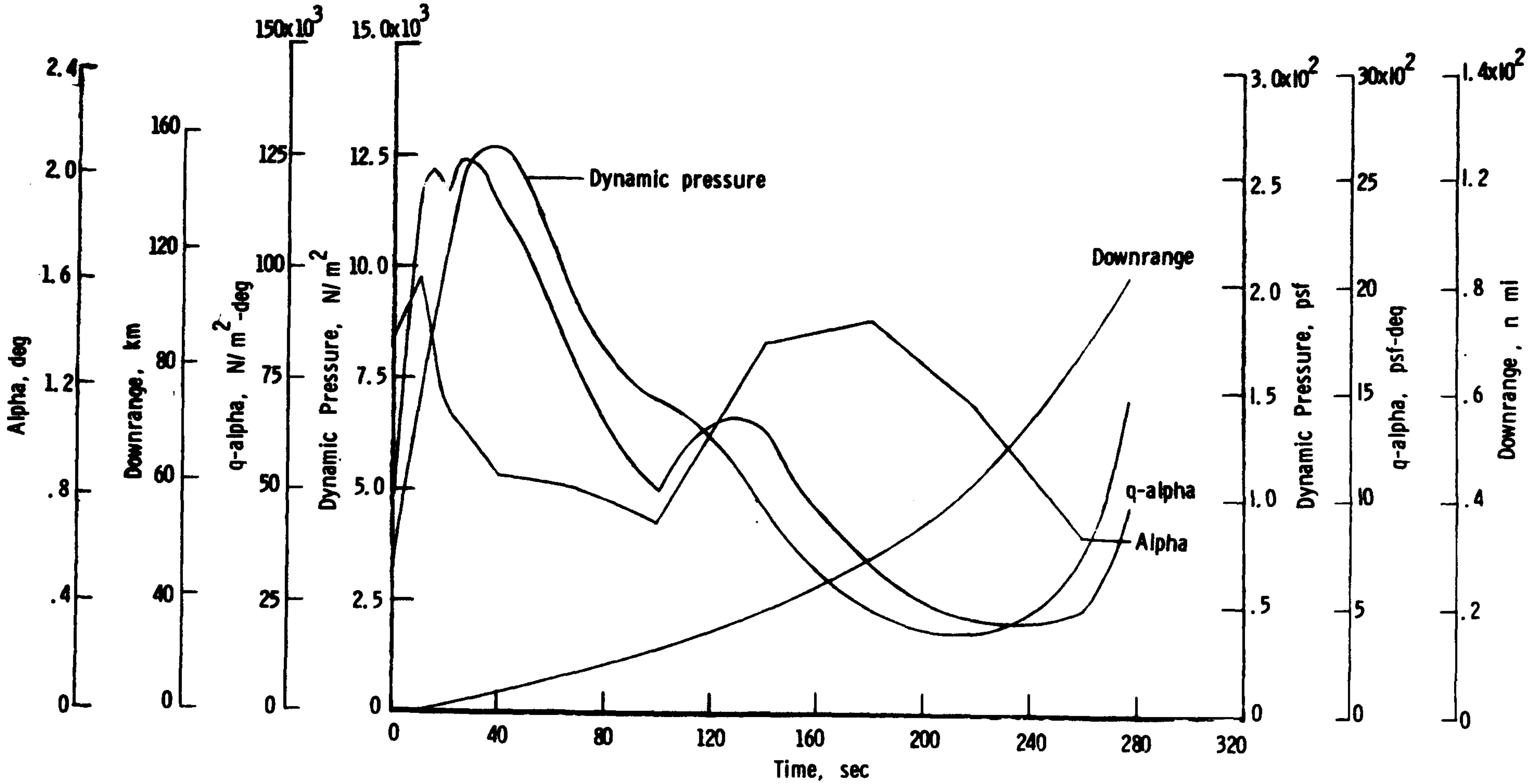

(b) Angle of attack, dynamic pressure, q-alpha, and downrange.

Figure 14.- Concluded.




(a) Altitude, velocity, flight path angle, and total acceleration.

Figure 15.- Airlaunched with 68,040 kg (150,000 lb) of propellant, J-2 powered.




(b) Angle of attack, dynamic pressure, q-alpha, and downrange.

Figure 15.- Concluded.

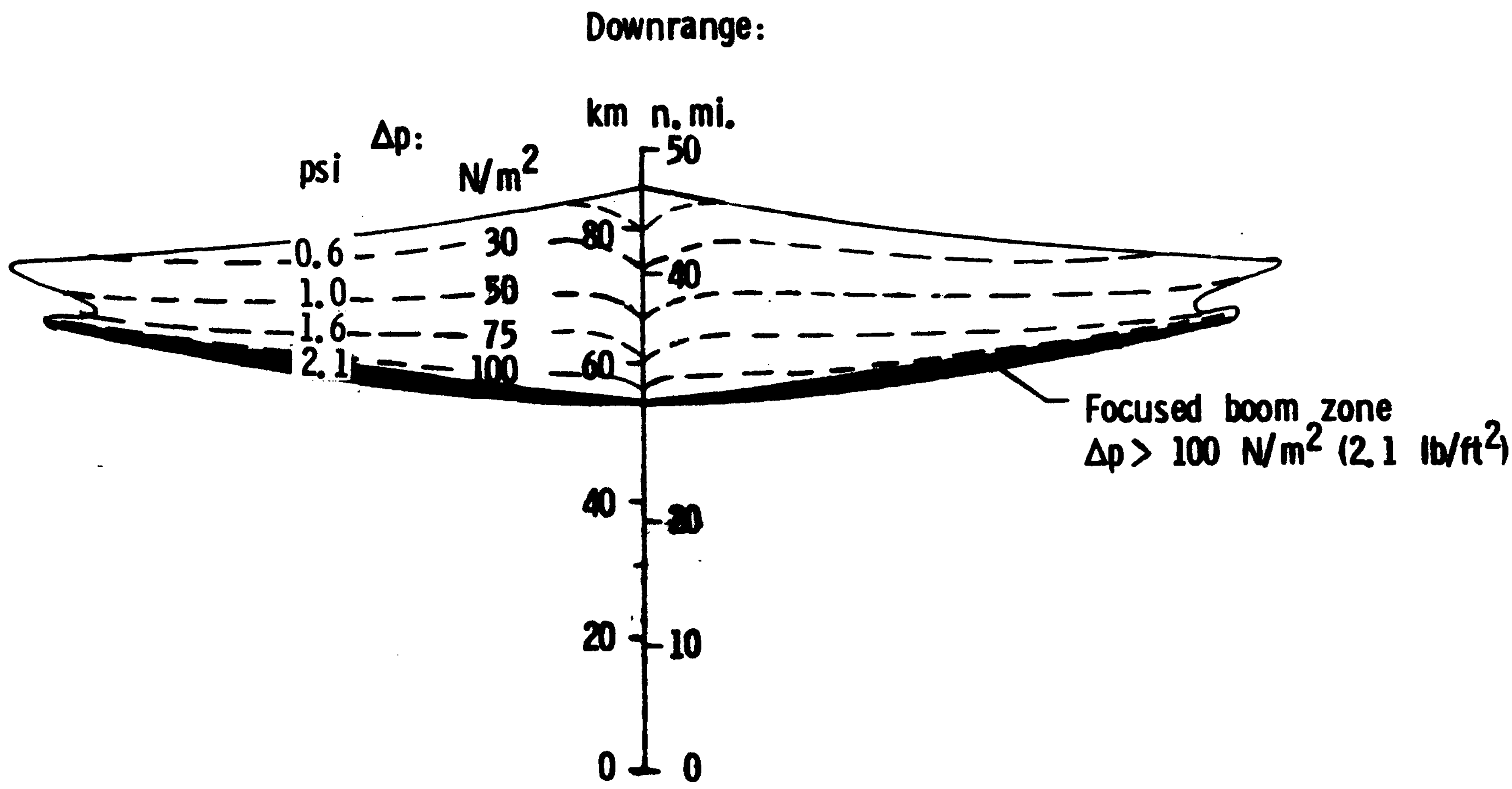



Figure 16. - Focused sonic-boom footprint.