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FULL-SCALE AIRCRAFT CRASH TESTS OF MODIFIED JET FUEL

Robert H. Ahlers

JULY 1977

FINAL REPORT

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16. Abstract

Crash tests were conducted with two A3 and two RB66 aircraft under impact-survivable crash conditions. The wing tanks in the first RB66 aircraft contained Jet A fuel modified with an 0.7-percent polymeric additive. The aircraft was crash tested into the specially constructed test site at 104.6 knots. The fuel mist generated by the fuel released from four crash-inflicted openings in the front wing spar was not ignited by the array of ignition sources. The wing tanks in the second RB66 aircraft were filled with JET A fuel modified with 0.5-percent of the same polymeric additive. The aircraft was crashed into the test site at 102.4 knots. The test conditions for the second RB66 test were made more severe by increasing the fuel temperature, partially drilling out areas in the front spar to increase the opened fuel spillage area, and by adding four fuel release openings under the wing, larger ignition sources, and operating the engines. The fuel mist burst into flame and followed the aircraft down the test site, continuing to burn until extinguished by the firefighting crew. These full-scale tests indicate that modified fuels have a potential for reducing the postcrash fire hazard and that small-scale tests should be conducted which are representative of full-scale crash conditions to determine the additive concentration to be used in any future crash tests.

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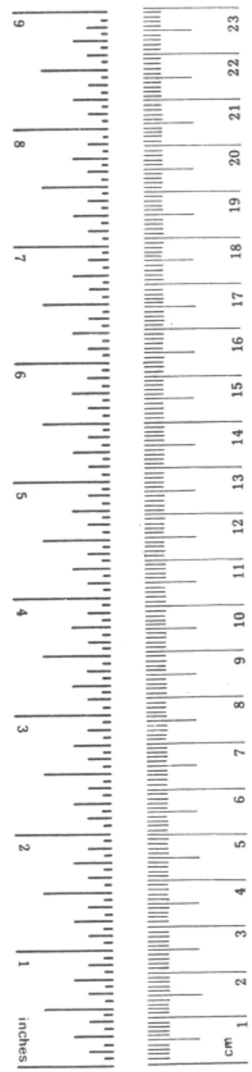
METRIC CONVERSION FACTORS

Approximate Conversions to Metric Measures

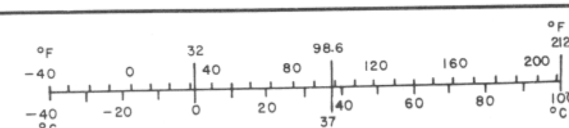
Symbol	When You Know	Multiply by	To Find	Symbol
		LENGTH		
in	inches	*2.5	centimeters	cm
ft	feet	30	centimeters	cm
yd	yards	0.9	meters	m
mi	miles	1.6	kilometers	km
		AREA		
in^2	square inches	6.5	square centimeters	cm^2
ft^2	square feet	0.09	square meters	m^2
yd^2	square yards	0.8	square meters	m^2
mi^2	square miles	2.6	square kilometers	km^2
	acres	0.4	hectares	ha
		MASS (weight)		
oz	ounces	28	grams	g
lb	pounds	0.45	kilograms	kg
	short tons (2000 lb)	0.9	tonnes	t
		VOLUME		
tsp	teaspoons	5	milliliters	ml
Tbsp	tablespoons	15	milliliters	ml
fl oz	fluid ounces	30	milliliters	ml
c	cups	0.24	liters	l
pt	pints	0.47	liters	l
qt	quarts	0.95	liters	l
gal	gallons	3.8	liters	l
ft^3	cubic feet	0.03	cubic meters	m^3
yd^3	cubic yards	0.76	cubic meters	m^3
		TEMPERATURE (exact)		
°F	Fahrenheit temperature	5/9 (after subtracting 32)	Celsius temperature	°C

*1 in = 2.54 (exactly). For other exact conversions and more detailed tables, see NBS Misc. Publ. 286, Units of Weights and Measures, Price $2.25, SD Catalog No. C13.10:286.



Approximate Conversions from Metric Measures

Symbol	When You Know	Multiply by	To Find	Symbol
		LENGTH		
mm	millimeters	0.04	inches	in
cm	centimeters	0.4	inches	in
m	meters	3.3	feet	ft
m	meters	1.1	yards	yd
km	kilometers	0.6	miles	mi
		AREA		
cm^2	square centimeters	0.16	square inches	in^2
m^2	square meters	1.2	square yards	yd^2
km^2	square kilometers	0.4	square miles	mi^2
ha	hectares (10,000 m^2)	2.5	acres	
		MASS (weight)		
g	grams	0.035	ounces	oz
kg	kilograms	2.2	pounds	lb
t	tonnes (1000 kg)	1.1	short tons	
		VOLUME		
ml	milliliters	0.03	fluid ounces	fl oz
l	liters	2.1	pints	pt
l	liters	1.06	quarts	qt
l	liters	0.26	gallons	gal
m^3	cubic meters	35	cubic feet	ft^3
m^3	cubic meters	1.3	cubic yards	yd^3
		TEMPERATURE (exact)		
°C	Celsius temperature	9/5 (then add 32)	Fahrenheit temperature	°F



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INTRODUCTION

PURPOSE.

The purpose of this project was to demonstrate the performance of modified fuels when subjected to full-scale crash conditions in the presence of positive ignition sources and to determine the tendency of the fuels to mist and burn when forcibly expelled from ruptured airplane fuel tanks.

BACKGROUND.

In relatively minor aircraft accidents where airplane structures and occupants are subjected to forces that are not seriously destructive, volatile fuels are spilled and cause postcrash fires that result in severe property damage and loss of life. In an effort to reduce the destruction and loss of life caused by postcrash fires in survivable accidents, the Federal Aviation Administration (FAA) has been investigating various ways to either contain the fuel after a crash, or modify the fuel characteristics to reduce or eliminate the tendency to mist under impact conditions and burst into flame after contacting ignition sources such as friction sparks, hot surfaces, or flames that might be encountered during an actual crash of an air carrier airplane.

For the work reported herein, a reduction in postcrash fires was attempted by using modified fuels in the wing tanks of the test aircraft to reduce the tendency of the fuel to mist after being released under impact. In small-scale test work where 1- or 2-gallon quantities of fuel were used, it was discovered that the addition of as little as 0.3 percent of a particular additive would reduce the flammability of the fuel to a negligible amount when compared with normal jet-engine fuel, such as JET A. When larger quantities of modified fuel were catapulted into ignition sources, the flammability and flame propagation of the modified fuels was so slight, compared with unmodified fuels under the same test conditions, that it was decided to study the performance of modified fuels in a full-scale test environment.

DISCUSSION

TEST AIRPLANES AND LOCATION.

The RB66 airplane was chosen as the test vehicle over other available surplus aircraft because it had integral wing fuel tanks and jet engines that hung below the wing on pods similar to a typical air carrier jet airplane, such as the Boeing 707, and therefore the fuel spillage under crash conditions would be closely comparable.

Facilities for accelerating a 50,000-pound airplane to a test velocity of 110 knots were not available at NAFEC, so Agreement No. DOT FA72NA-AP-17 was established between NAFEC and the Naval Air Test Facility, (NATF) Lakehurst, New Jersey, to perform the tests under the technical guidance of NAFEC personnel.

Four airplanes were used in the program; two Navy, type A3, airplanes, and two Air Force, type RB66, airplanes. The RB66 is actually an Air Force version of the A3 with different engines and landing gear, but these differences did not enter into the test results. The primary purpose for using the two A3 airplanes was to verify the severity of the test site conditions and to predict whether or not the tests with the RB66 airplanes would remain within the limits of survivability during actual testing of the modified fuels.

TEST DESCRIPTION.

Before actual full-scale crash tests could be performed using the selected RB66 as the test vehicle, a suitable crash site environment had to be established. From available results on previous similar tests, it was decided to start with a test site configuration as shown in figure 1, using a 6° sloped, earthen mound followed by a 2-foot-high pillow of earth. Preliminary calculations, as shown in appendix A, indicated that the airplane would become airborne for approximately 300 feet after leaving the 6° slope. However, since several assumptions had to be incorporated into the mathematics of appendix A, it was necessary to actually perform a full-scale test run.

A Navy, type A3, airplane was available for the first test run. This airplane had been used by NATF for several tests where the airplane was arrested by flexible nylon nets. The airplane had sustained slight damage from this work, and the engines had been removed. Surplus J57 jet engines were obtained and mounted on the airplane, since the engines were a necessary part of the airplane in duplicating typical crash characteristics.

All four aircraft tested were launched on Recovery System Test Site No. 1 (RSTS No. 1). The aircraft were propelled down the track by a four-engined jet car with centrifugal-flow jet engines that developed a thrust of 5,000 pounds each, or a total thrust of 20,000 pounds for the four engines. The aircraft were constrained to follow the RSTS No. 1 track by a shuttle that, pushed by the jet car, pulled the aircraft, using its own tricycle landing gear, through flexible cables attached to the underside of the aircraft. Figure 2 shows the jet car, the shuttle, and the aircraft cables that connected the aircraft to the shuttle. Figure 3 is an overall view of one of the test airplanes ready to be propelled down the track by the jet car.

RSTS No. 1 track was 5,800 feet long, which was more than enough distance for the 20,000-pound thrust of the jet car to accelerate the test aircraft up to the desired speed of 102 to 115 knots. At a predetermined point on the track, the jet car was braked to a stop, and the airplane and shuttle coasted onward. Then the shuttle was braked to a stop, at which time a weak link in the cables holding the airplane to the shuttle gave way, and the momentum of the rolling airplane carried it into the test site area.

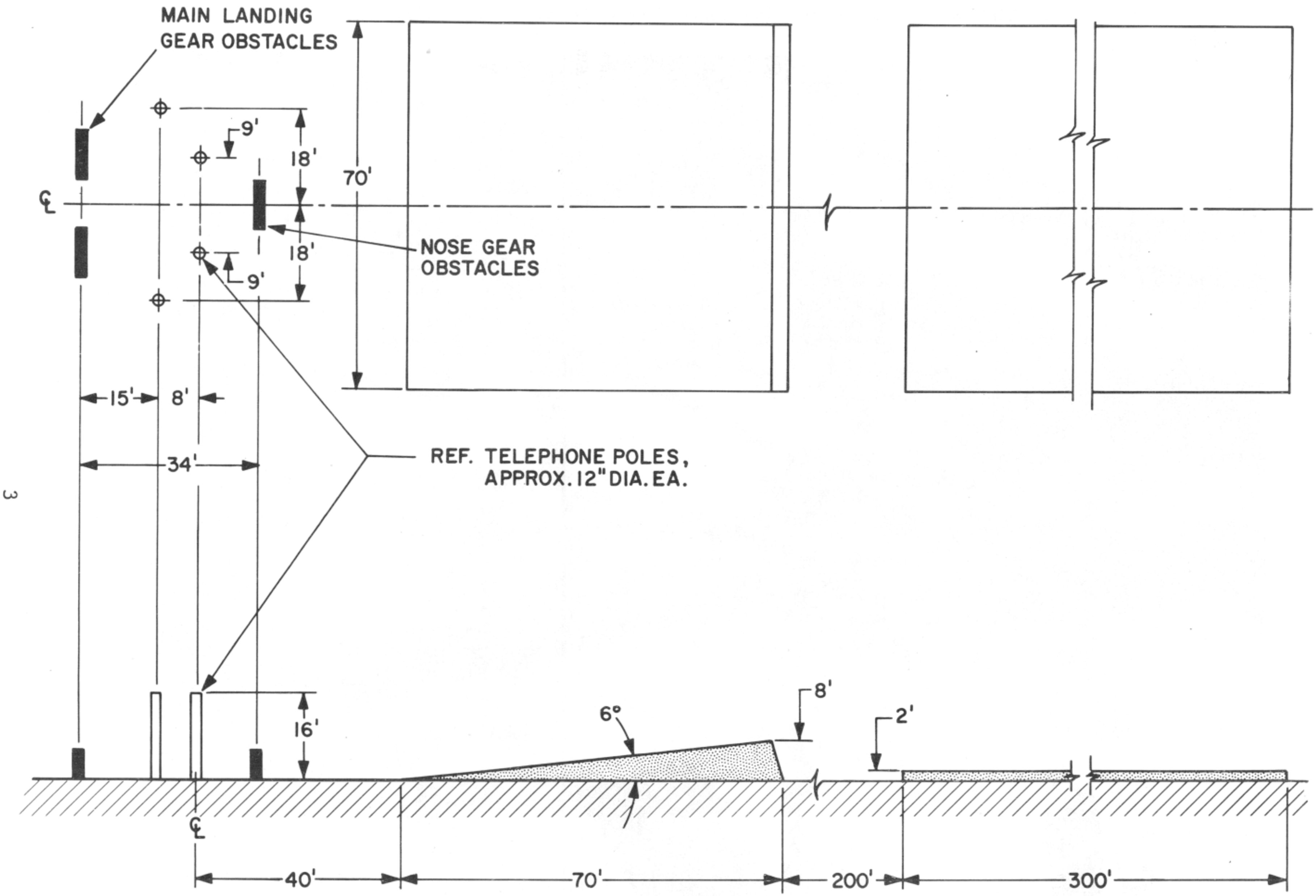

FIGURE 1. TEST SITE CONFIGURATION FOR TEST NO. 1

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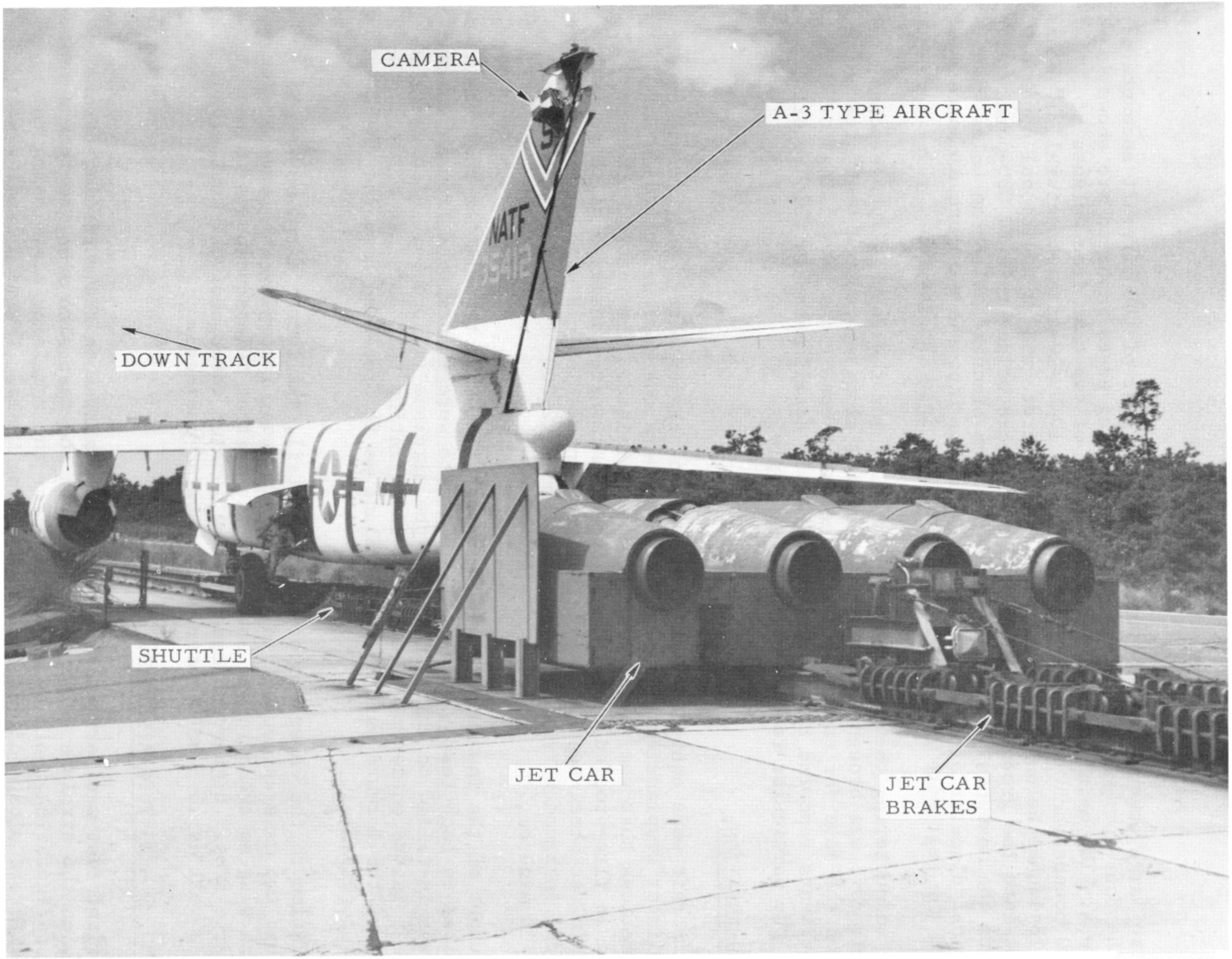
FIGURE 2. TEST AIRPLANE ATTACHMENT TO THE JET CAR SHUTTLE



74-38-3

FIGURE 3. TEST AIRPLANE ATTACHED TO THE JET PUSHER CAR

The two large obstacles that the main landing gear of the aircraft encountered and the four upright poles that the wing leading edges impacted are shown in figure 4. The main gear obstacles consisted of two cast iron weights of 20,000 pounds each, which were reinforced by 10-by-10-inch structural "I" beams sunk 18 inches below the track surface in concrete and an additional depth of 3 feet into the ground. Similarly, I beams were welded to a 20,000-pound weight to form the nosewheel barrier. The poles were standard telephone poles with a nominal diameter of 12 inches, sunk 18 inches into the reinforced concrete pad of the test site and an additional 4 feet into the ground.

After encountering the landing gear obstacles and the four telephone poles, the aircraft was brought to a stop by sliding along a prepared test site which was not the same for each test run. The exact conditions prevailing for each of the tests reported herein are described in detail later in this report under the particular test for which they apply.

INSTRUMENTATION.

The aircraft were equipped with strain gauge accelerometers and, in one case, with linear displacement transducers to measure the destructive loads and deformations that the aircraft structure sustained. The instruments were housed in fire-proofed containers. These containers consisted of three layers of stainless steel separated by glass wool insulation and were designed to protect the instrument from temperatures of up to 2,000° Fahrenheit (F) for at least 30 minutes.

The output of the instruments was sent to a microwave transmitter that was also installed in each test airplane, and the data were transmitted from an antenna that was installed on the tail fin of the airplane (figure 5). Also shown on figure 5 is the motion picture camera installation that obtained color photographs of the forward part of the airplane as it proceeded down the track into the test site area.

In addition to the camera coverage in the tail, the fourth airplane tested also had two cameras mounted in the cockpit area, recording the leading edge of each wing as the airplane encounted the upright telephone poles in the test site area. High-speed cameras were located throughout the test site area as well as in a helicopter which hovered overhead.

TEST NO. 1 - A3 TYPE AIRPLANE.

This first test was performed to obtain information on the retarding forces acting on the airplane as it passed through the test site area, so that a test site configuration could be constructed for the remaining tests using the RB66 aircraft with modified fuel in the wing tanks. The main landing gear obstacles, the nosewheel obstacle, and the four telephone poles for rupturing the leading edge of the wing were installed, and a 6° sloped earthen mound followed by a 2-foot-high pillow of earth was constructed with dimensions as shown in figure 1.

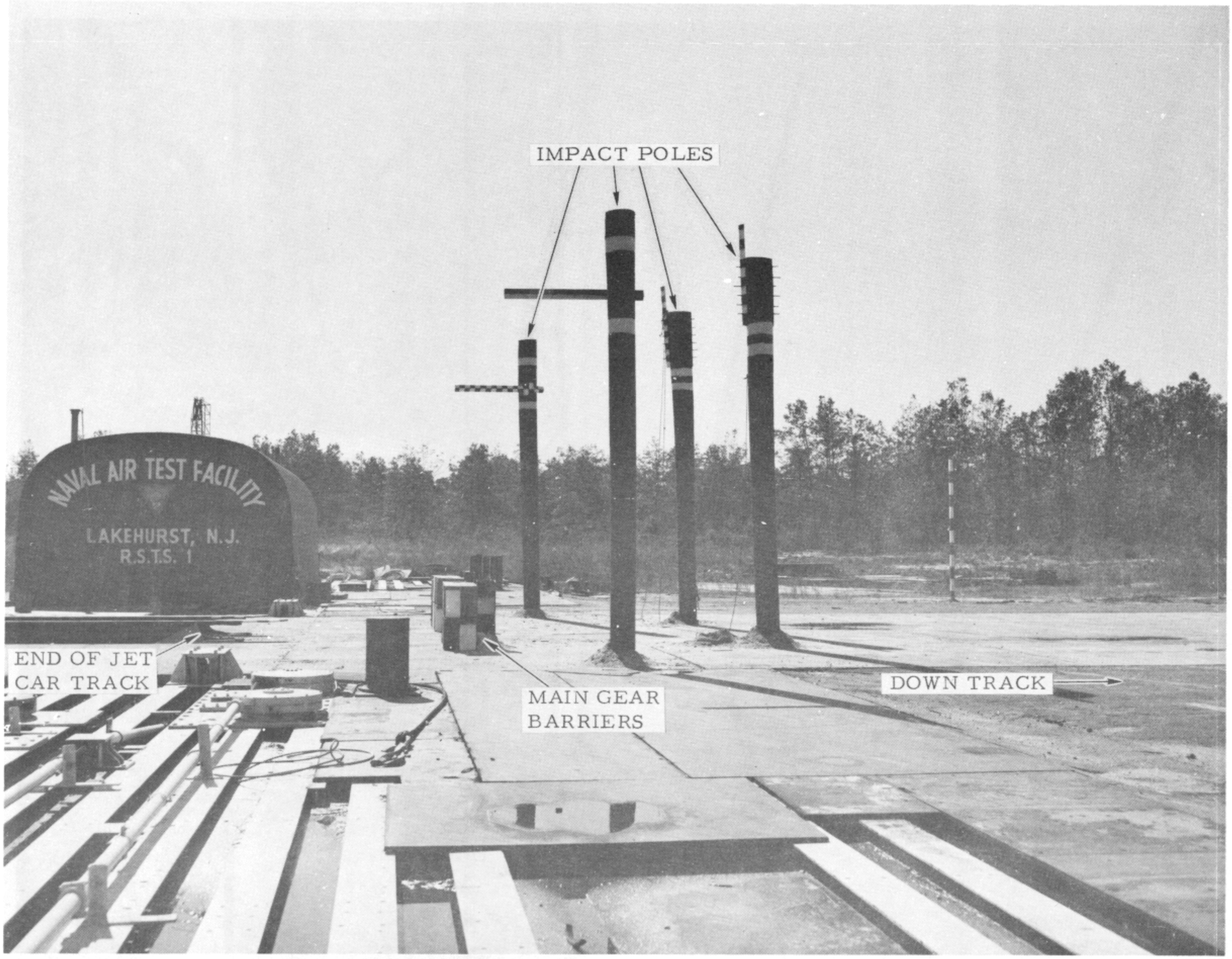
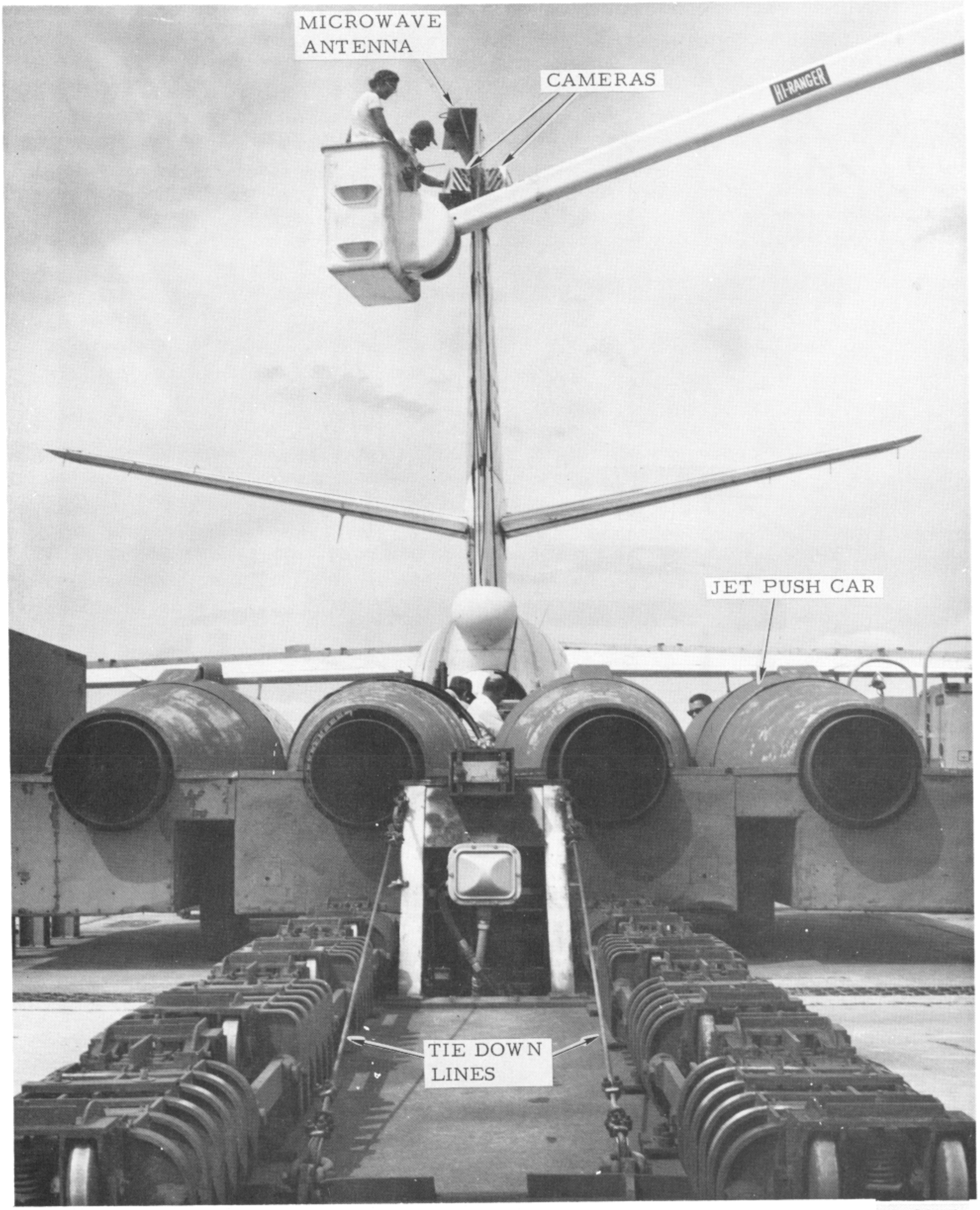


FIGURE 4. TYPICAL TEST SITE AREA CONFIGURATION



74-38-5

FIGURE 5. INSTRUMENTATION INSTALLATION ON TEST AIRPLANE TAIL

The aircraft wing tanks were filled with dyed water. The instrumentation consisted of four triaxial accelerometers mounted as follows:

1. On the floor of cockpit area station 134,
2. On the floor of bomb bay area station 450,
3. On the floor of electronic equipment bay station 770, and
4. On the upper surface, port wing, 13.5 feet off centerline (W.L. 165).

Two 16-millimeter (mm) motion picture cameras were mounted on the vertical tail surface stabilizer recording the view forward as the airplane entered the test site area. Figure 6 is the view looking downtrack at the test site area showing the landing gear obstacles, telephone pole barrier, and 6° sloped earthen bank.

TEST NO. 2 - A3 TYPE AIRPLANE.

From the theoretical study contained in appendix A, the entire test site area was determined based on a 6° sloped bank followed by a 15° sloped hill. The purpose of test No. 1 was actually to determine where to construct the 15° sloped hill in relation to the 6° sloped bank, since several assumptions had to be made in the calculations of appendix A and could be verified only by performing a full-scale test. As it turned out, the predictions contained in appendix A were very accurate; however, test No. 1 revealed that the 6° sloped hill was too formidable a barrier for the airplane, and it was decided to reduce the slope from 6° to 3° for the next test. Also, from the experience gained in test No. 1, sufficient confidence was obtained to allow a decision on the relative location of the 15° sloped hill.

It was reasoned that the airplane would remain airborne for a shorter distance coming off the 3° sloped bank than it did off the 6° sloped bank. The 15° sloped hill would be constructed at the point where the airplane in test No. 1 came to rest. This would assure that the next test airplane would not be airborne when it encountered the 15° slope, and the airplane would, in all probability, lose all of its forward momentum before it reached the crest of the 15° sloped hill. By keeping the airplane from going over the crest of the 15° sloped hill, the crash forces would be kept below the upper limits of survivability.

Since these changes resulted in a significant deviation from the original test conditions, it was necessary to perform another test run before actually evaluating the performance of modified fuels in the RB66 airplanes. Therefore, this second test run was made using another A3-type airplane that had been used by NATF for barrier work and was no longer suitable for that type of work, but would make a satisfactory test vehicle for this project.

The test site area for this second run was configured as shown in figures 7 and 8, except that no ignition sources were present. The main landing gear obstacles and the four telephone poles were the same as in test No. 1, but the obstacle for removing the nosewheel was moved downtrack to the base of the 3° sloped bank. By leaving the nosewheel intact until the airplane

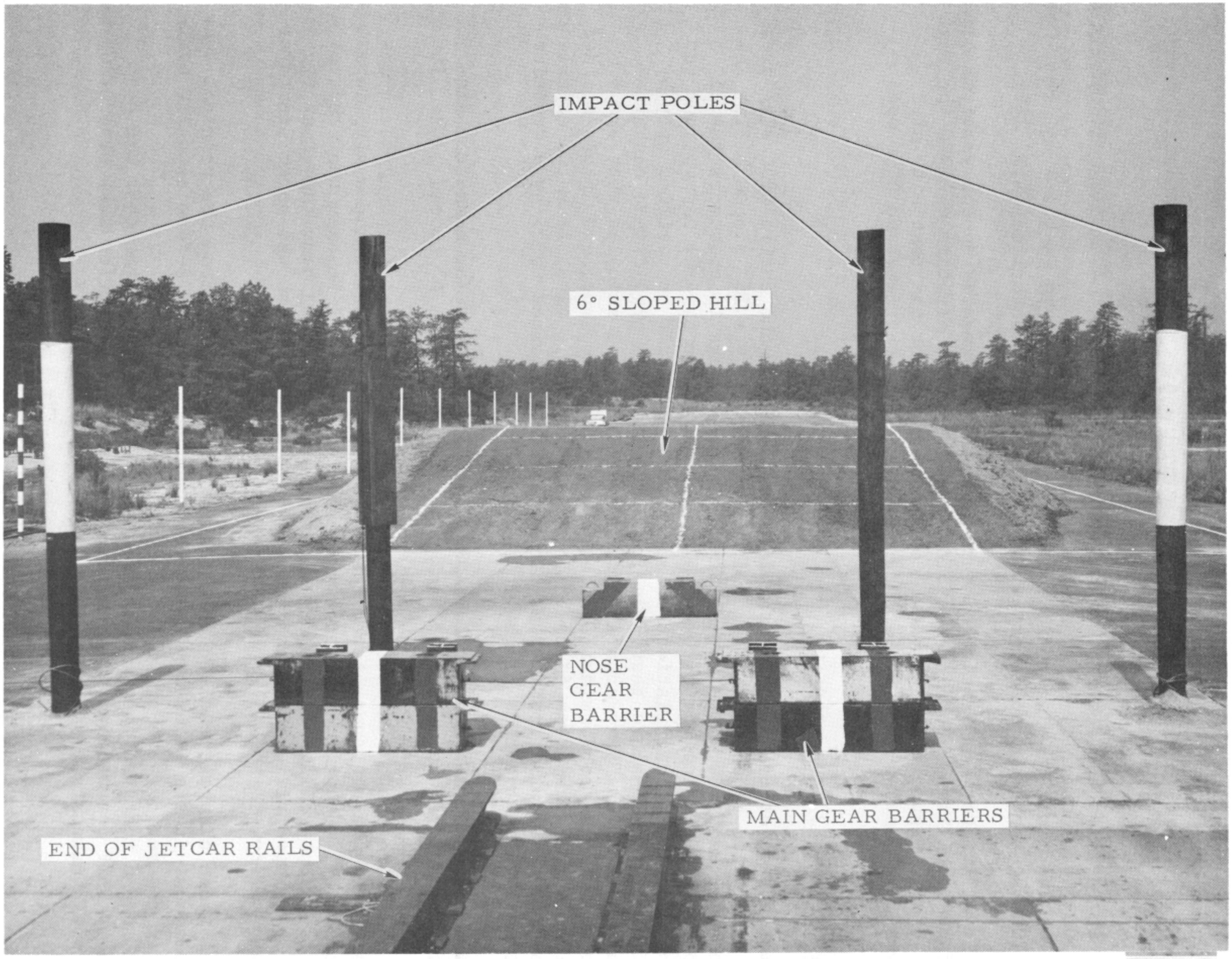


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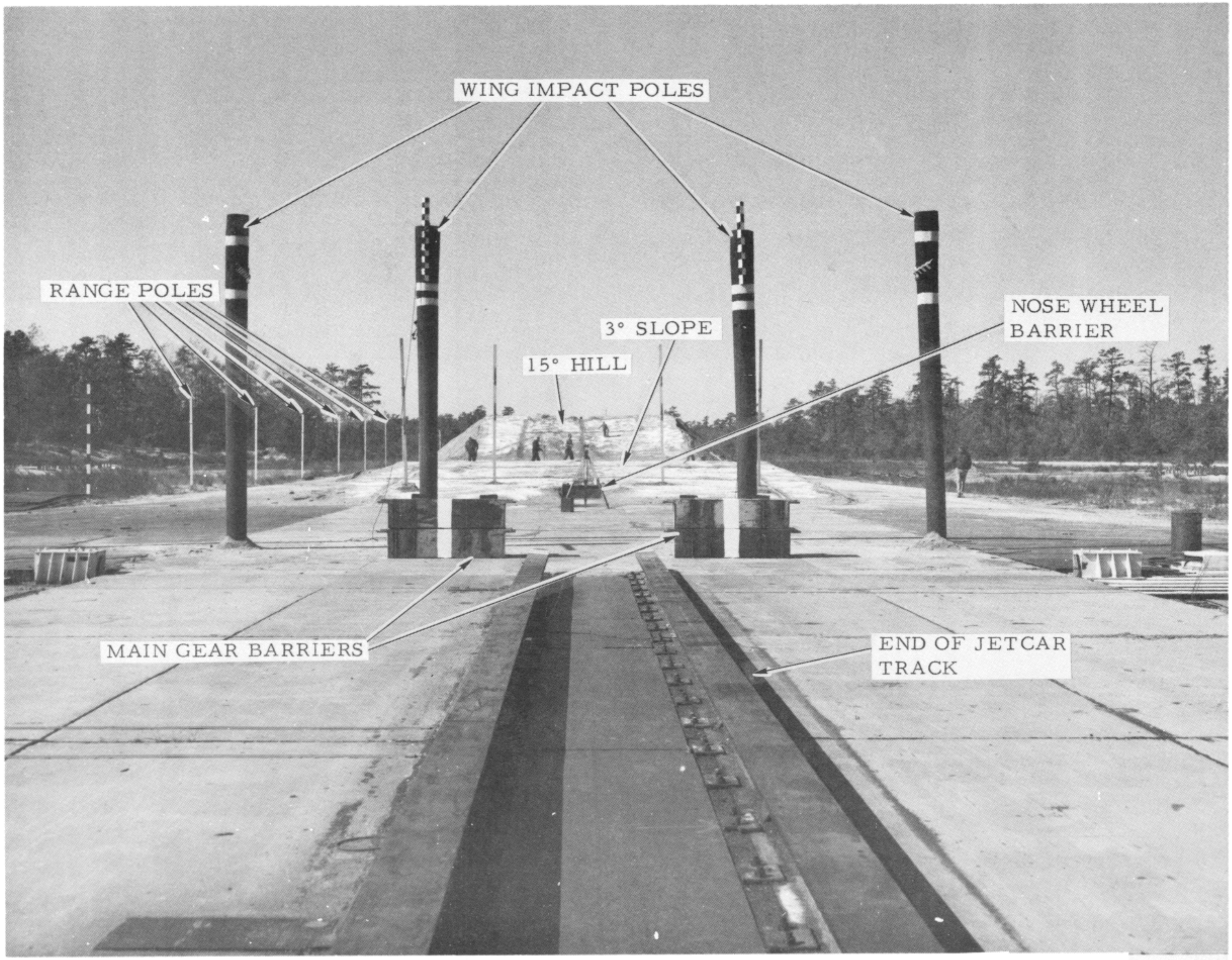
FIGURE 6. DOWNTRACK VIEW OF TEST SITE AREA FOR TEST NO. 1



74-38-9

FIGURE 7. TEST SITE AREA OF TESTS NO. 2 AND NO. 3

reached the first hill, the tendency for the nose to settle as noted in test No. 1 was eliminated, and the impact forces were distributed over most of the fuselage and engine nacelles rather than concentrated at the nose of the airplane. By relieving the stress concentration on the nose, the survivability aspects of the crash were enhanced.

Although the primary purpose of this second run with the A3-type airplane was to verify the mathematical predictions on the dynamic crash forces, it was decided to take advantage of this test run by filling the integral wing tanks of the A3 with modified fuel, rather than dyed water, as was done in the previous test, so that the fuel dispersal characteristics under impact and the misting tendencies under deceleration could be evaluated. The starboard (right) wing tank of the A3 was filled with 750 gallons of modified fuel, dyed green. The fuel additive was manufactured by the Imperial Chemical Industries of Great Britain and was designated as FM-4. A concentration of 0.3 percent by weight of this additive was mixed with JET A type fuel. The port (left) wing tank was filled with 750 gallons of modified fuel, dyed red. The fuel additive was manufactured by the Continental Oil Co. and was designated as CONOCO-AM-1. A concentration of 0.2 percent of this additive was mixed with JET A type fuel.

The airplane was propelled down the track by the four jet engines of the pusher car and reached an impact speed of 115 knots as shown in table 1. There were no ignition sources placed in the test site area, because it was not desired to test the flammability characteristics of the modified fuels in the wing tanks, but only their dispersal and misting properties under impact conditions. The airplane came to rest just over the crest of the 15° sloped hill, as shown in figures 9, 10, 11, and 12. This airplane was considered to have remained within the upper limits of survivability, even though it went over the crest of the 15° hill, because the fuselage remained intact in the crew's compartment and the only real break in the structure was in the aft section, as shown in the figures. On this test, the telemetering equipment remained operating for 2 seconds, and the deceleration forces experienced in various parts of the airplane during the test were recorded and are presented in table 2.

TEST NO. 3 - RB66 AIRPLANE.

The test site area configuration for this test was the same as that for test No. 2 (i.e., a 3° sloped bank followed by a 2-foot-high pillow of earth and a 15° sloped hill), except that in this test, there were positive ignition sources spaced as shown in figure 8. Figure 13 is a sketch of the flares used as ignition sources along with the temperatures measured in various parts of the flame area.

The aircraft's integral wing tanks were filled with 1,500 gallons of modified fuel (750 gallons on each side of wing). The additive used was manufactured by the Dow Chemical Company and was designated as XD8132. The additive was mixed with Jet A type fuel at a concentration of 0.7 percent.

TABLE 1. DATA SHEET

	Test Event No. 1	Test Event No. 2	Test Event No. 3	Test Event No. 4
Date	June 1, 1972	September 1, 1972	October 20, 1972	January 24, 1973
Test Vehicle	A3	A3	RB66	RB66
Test Vehicle Weight (lbs)	52,900	53,600	53,000	53,400
Type of Fuel	Water	Conoco - Port FM - 4 - Starboard	Dow 0.7 percent	Dow 0.5 percent
Fuel Temp (°F)	-	74	61	80-85
Ambient Temp (°F)	70	82	50	43
Aircraft Engine Setting				
Percent r/min				
Port	-	-	-	72
Starboard	-	-	-	72
Wind Velocity (knots)	0-10 W	5-20 NW	5-10 NW	10-20 W
Test Veh. Cutoff Speed (knots)	128.2	124.4	112.1	110.0
Impact Speed (knots)	119.8	115.2	104.6	102.4
Ignition Source	None	None	60 Flares, 8 Smudge pots	30 Flares 4 Propane Torches
Test Site Configuration (slope of mounds)	6° slope 2 ft pillow	3° slope 15° hill	3° slope 15° hill	3° slope 2-ft-high pillow 700 ft long
Remarks	1	2	3	4

NOTES:

1. Test vehicle was severely damaged when it encountered 6° slope. It was totally wrecked on the 2-foot-high pillow 300 feet down range from the 6° slope. It was considered not a survivable crash. The spray pattern of water from ruptured tanks was well defined.

2. Test vehicle came to rest in a nose-down attitude on the far side of the 15° hill. Fuselage broke in half at station 595. Modified fuel was dyed red in port wing tank and green in starboard tank. Spray pattern from ruptured tanks was not nearly as great as last shot.

3. Test vehicle came to rest on face of 15° hill only the fiberglass tailcone broke off fuselage. Spray pattern similar to last shot. No postcrash fire. Even attempts to light it after the test were not considered successful.

4. Test vehicle burst into flame on the 3° slope and came to rest 600 feet downtrack on the 2-foot pillow.

FIGURE 9. A3 AIRPLANE OF TEST NO. 2 (OVERHEAD RIGHT FRONT VIEW)

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FIGURE 10. A3 AIRPLANE OF TEST NO. 2 (OVERHEAD RIGHT REAR VIEW)

74-38-13

FIGURE 11. A3 AIRPLANE OF TEST NO. 2 (GROUND FRONTAL VIEW)

FIGURE 12. A3 AIRPLANE OF TEST NO. 2 (GROUND RIGHT FRONT VIEW)

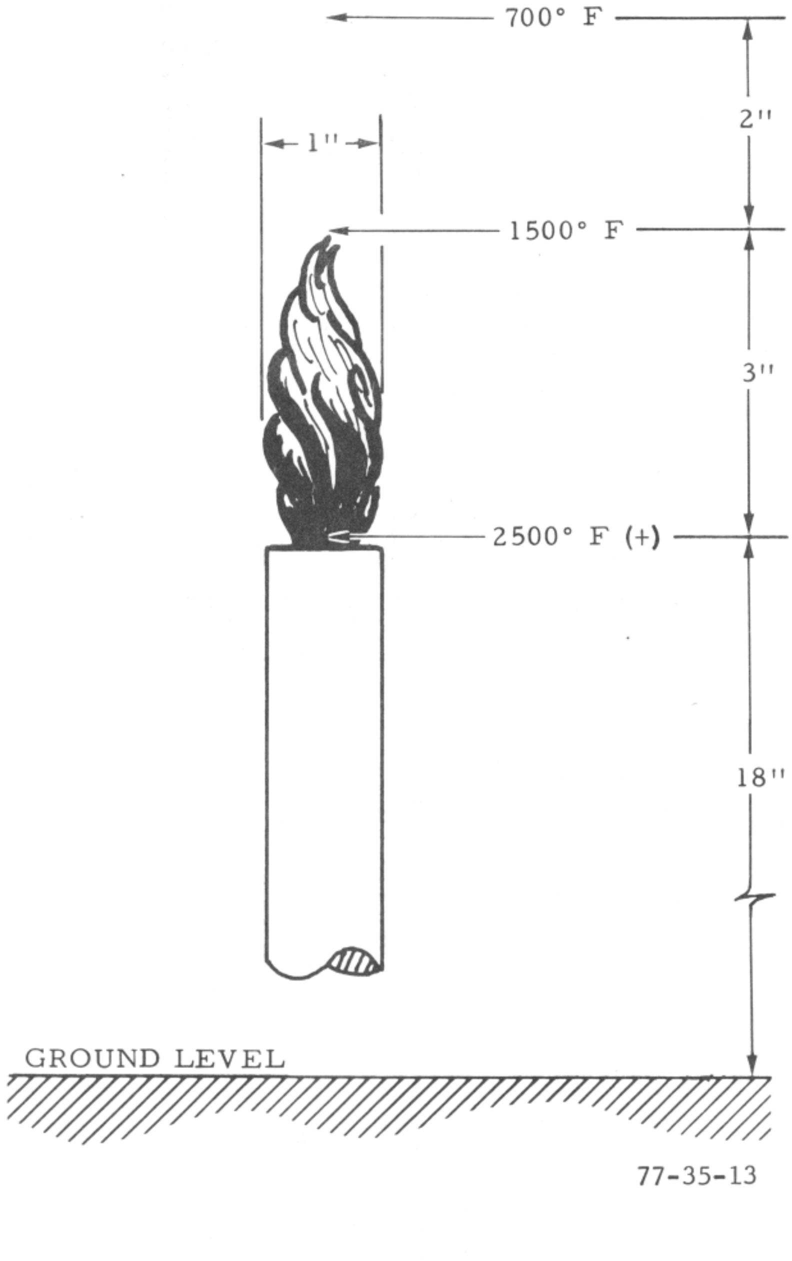

FIGURE 13. TEMPERATURE PROFILE OF FLARES USED IN TEST NO. 3

TABLE 2. TEST NO. 2 - A3 AIRPLANE

PEAK ACCELERATION VALUES*

Description	Station*	Impact With Telephone Poles		Impact With 3° Slope		Impact On 15° Slope		Remarks
		Long	Vert	Long	Vert	Long	Vert	
Accelerometers								
Pilot Compartment	134	+11	+ 8	+30	+59	-	-	Telemetry equipment stopped functioning
		- 7	- 7	-12	-45	-	-	before airplane reached 15° slope.
Wheel Well	531	+44	+17	+37	+68	-	-	
		-11	-40	- 9	-44	-	-	
Electronics Bay	770	+29	+ 8	+45	+15	-	-	
		- 9	- 8	- 9	-14	-	-	
Left Wing	WL - 110**	+80	+15	+80	+44	-	-	Values exceedingly high because accel-
		-79	- 4	-79	- 8	-	-	erometer was located on wing directly in line with pole impact.

NOTES:

* For station identification and acceleration sign interpretation see figure 32.

** Wing waterline (WL) is distance in inches outboard of airplane fuselage centerline.

The instrumentation on the airplane was the same as in test No. 2 except that an anthropomorphic dummy was placed in the pilot position and instrumented in the pelvic area with two zero-to-50-g accelerometers. The time-history traces of these accelerometers are shown in appendix B. Also, four linear displacement transducers were installed in the fuselage to gain experience in measuring and evaluating structural distortion during impact when compared with the deceleration forces measured by the installed accelerometers, the time-history traces of the linear displacement transducers and time-history traces of the installed accelerometers are also shown in appendix B.

The RB66 airplane was accelerated by the jet car to an impact speed of 105 knots. The integral wing tanks were ruptured by the telephone poles, and the modified fuel spilled out through the channels formed by the broken spar web. The airplane came to rest on the crest of the 15° sloped hill, but did not burn; nor were there any fires along the path of the airplane, even though fuel spilled out of the ruptured tanks, and the airplane ploughed through the ignition sources that were placed in the test site area. It came to rest on the crest of the 15° sloped hill as shown in figures 14, 15, and 16.

TEST NO. 4 - RB66 AIRPLANE.

Since there were no fires either on the ground or in the airplane on the last test, it was felt that the test was not severe enough to demonstrate satisfactorily the flame-resistant characteristics of modified fuel. It was also felt that the last test did not truly represent a full-scale crash, because the engines on the aircraft were not running, as would most likely be the case in an actual situation where the airplane was either taking off or landing. A further consideration was the desirability of having the airplane come to rest on level ground so that the fuel in the ruptured tanks could leak out and form pool fires that could be fought using new experimental fire-fighting techniques; thus getting additional valuable data from these tests.

Therefore, the 15° sloped hill was removed from the test site area, and the earth it contained was spread out downtrack to lengthen the runout area. The type of ignition source was changed from the last test. On the crest of the 3° sloped bank, four propane torches were spaced as shown in figure 17. Figure 18 is a sketch of the propane torches that were used. The torches were fed from four 6-cubic-feet propane tanks that were buried in the ground and heated electrically to maintain a pressure of 120 pounds per square inch (psi).

Beyond the propane torches were 30 ignition sources suspended from 1-inch-diameter steel poles, as shown in figure 19. Figure 17 shows the location of these ignition sources along the crash site area.

The obstacles for removing the landing gear were the same as in the last two tests, but the telephone poles that ruptured the leading edges of the wing were faced with fabricated steel angles placed at the point where the wing leading edge contacted the poles (figure 20).

FIGURE 14. RB66 AIRPLANE OF TEST NO. 3 (LEFT REAR VIEW)

74-38-17

FIGURE 15. RB66 AIRPLANE OF TEST NO. 3 (LEFT SIDE VIEW)

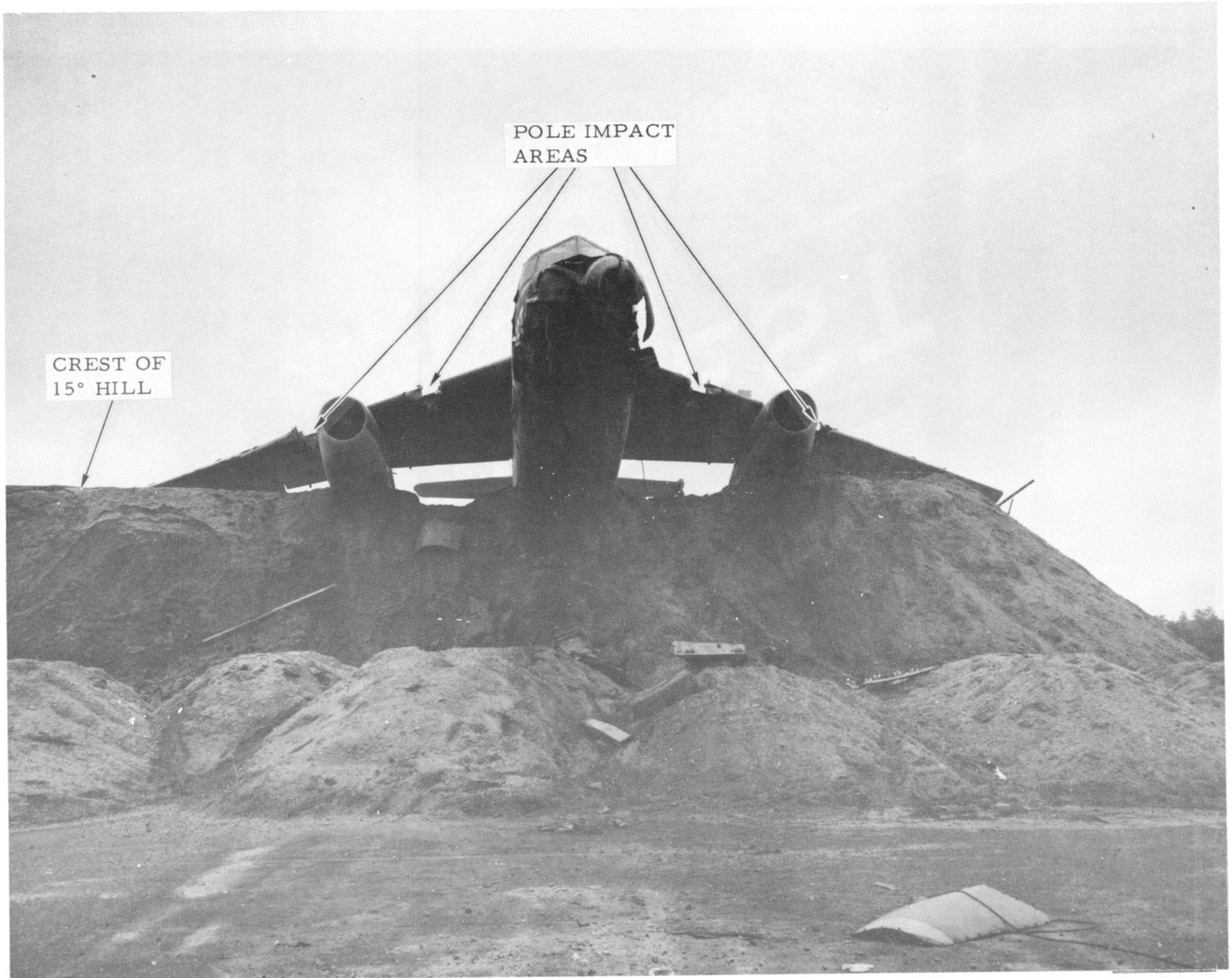

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FIGURE 16. RB66 AIRPLANE OF TEST NO. 3 (GROUND FRONTAL VIEW)

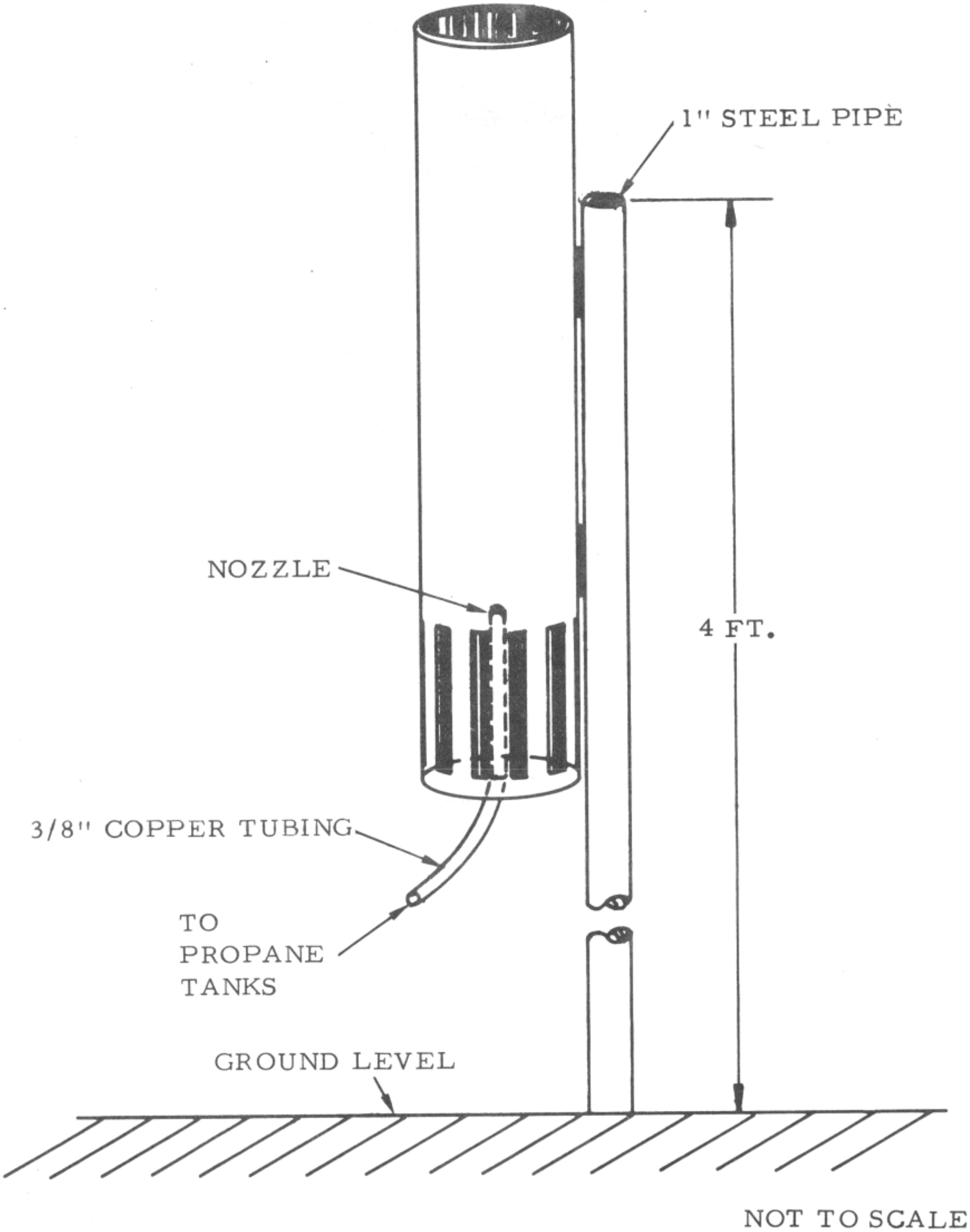

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FIGURE 18. SKETCH OF ESSENTIAL FEATURES OF THE PROPANE TORCHES USED IN TEST NO. 4

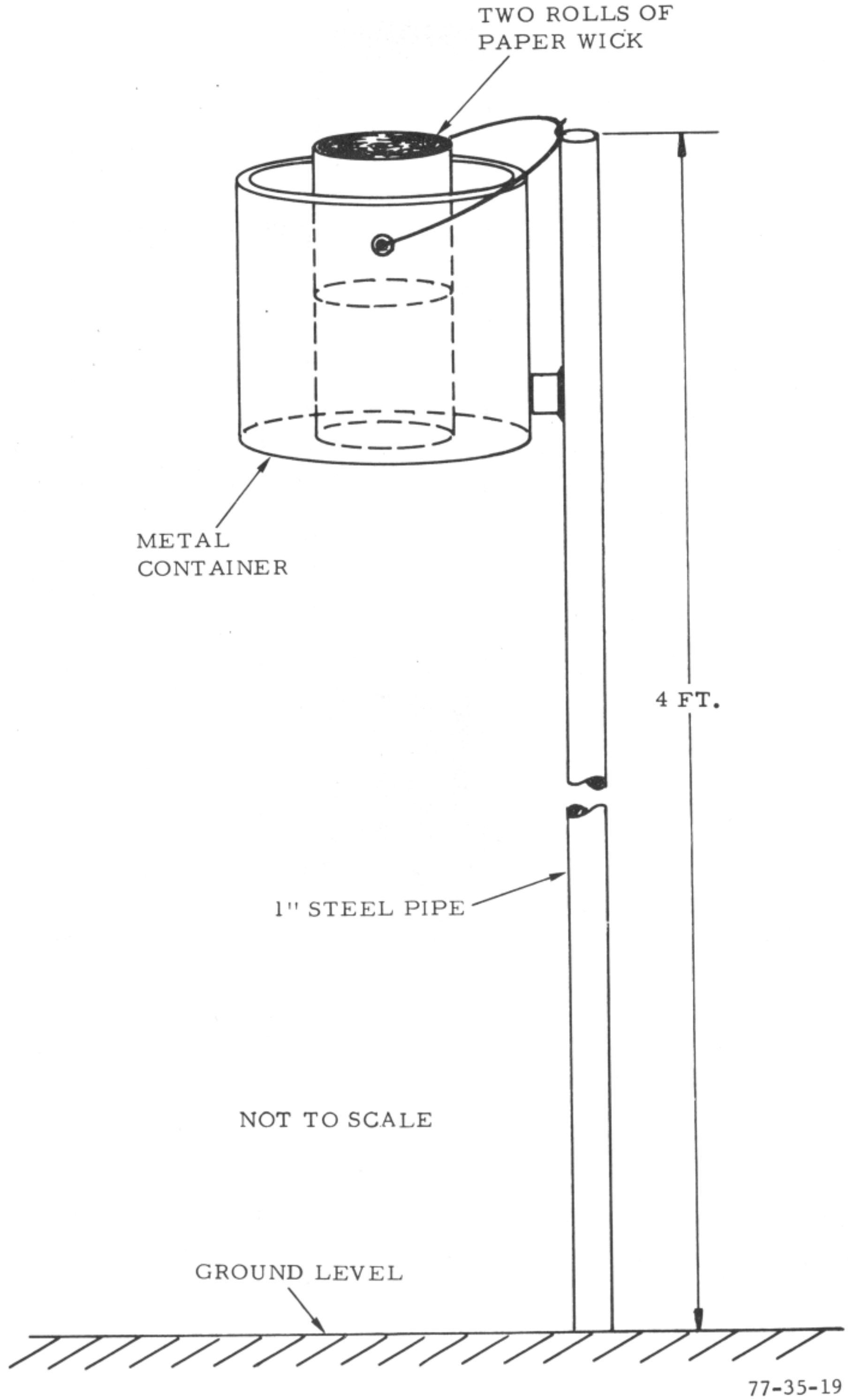

FIGURE 19. SKETCH OF THE FLARE POTS USED IN TEST NO. 4

74-38-22

FIGURE 20. OVERALL VIEW OF TEST SITE AREA OF TEST NO. 4

A special effort was made on this test to increase the quantity of fuel gushing from the wing tanks as the test airplane passed through the test site area. One scheme was to forcibly pull out two fuel tank inspection plates under each wing. Figure 21 shows the installation on the port wing. Heavy steel levers were welded to a modified inspection plate that had a weak membrane of thin aluminum, fabricated within its structure. The steel lever arm was connected to the main landing gear strut by 3/8-inch-diameter stainless steel cable, and during the actual test, when the landing gear was knocked off, the inspection plates were torn out of the wing tank skin allowing the fuel to escape. Figure 22 shows the underside of the wing after an inspection plate had been pulled out during the test.

From the results of the previous tests, it was determined that the wing spar caps were protecting the spar web from damage when the wing leading edge contacted the telephone poles during the test. The wing spar web formed the front bulkhead of the integral wing tank, and if this web did not suffer structural failure, the fuel inside the tank would not be released. To increase the probability of failing, the spar web devices as shown in figure 23 were fabricated from structural iron shapes and located in the four positions of the wing leading edge that would contact the upright telephone pole obstacles in the test site area. The fabricated structural shape would be the first thing to contact the telephone poles and therefore would be driven through the spar web, thus rupturing the wing fuel tank. Figure 24 shows the type of structural failure that the wing tanks underwent during this test.

The aircraft's integral wing tanks were filled with 1,500 gallons (estimated to be 1,250 gallons at impact) of modified fuel (750 gallons on each side of wing). The additive used was manufactured by the Dow Chemical Co., and was designated as XD8132. The additive was mixed with JET A fuel at a concentration of 0.5 percent.

In order to meet the requirement to have the engines running during this test, a special fuel system had to be devised. The normal fuel system installed in the RB66 requires that the engines pump fuel from the aft fuselage tank at all times, and that the fuel in the forward fuselage tank and the two wing tanks be transferred to the aft fuselage tank as needed. For the tests in this project, fuel was carried in the wing tanks only; the two fuselage tanks were empty, and therefore there was no direct feedline available to run the engines.

The modified fuel contained in the wing tanks was fed by means of a flexible hose leading from a modified wing tank inspection plate, as shown in figure 25, to a two-way valve, then on to the engine. A flexible hose leading from a supply of JP-4 fuel also connected to the two-way valve.

At this time, the fuel nozzles installed in the engines would not satisfactorily atomize modified fuels to the extent required for engine starts. Therefore, the engines had to be started on JP-4 type fuel, and after being brought up to 72-percent r/min, the fuel supply was switched from JP-4 to the modified fuel in the wing tanks by means of the two way valve shown in figure 25.

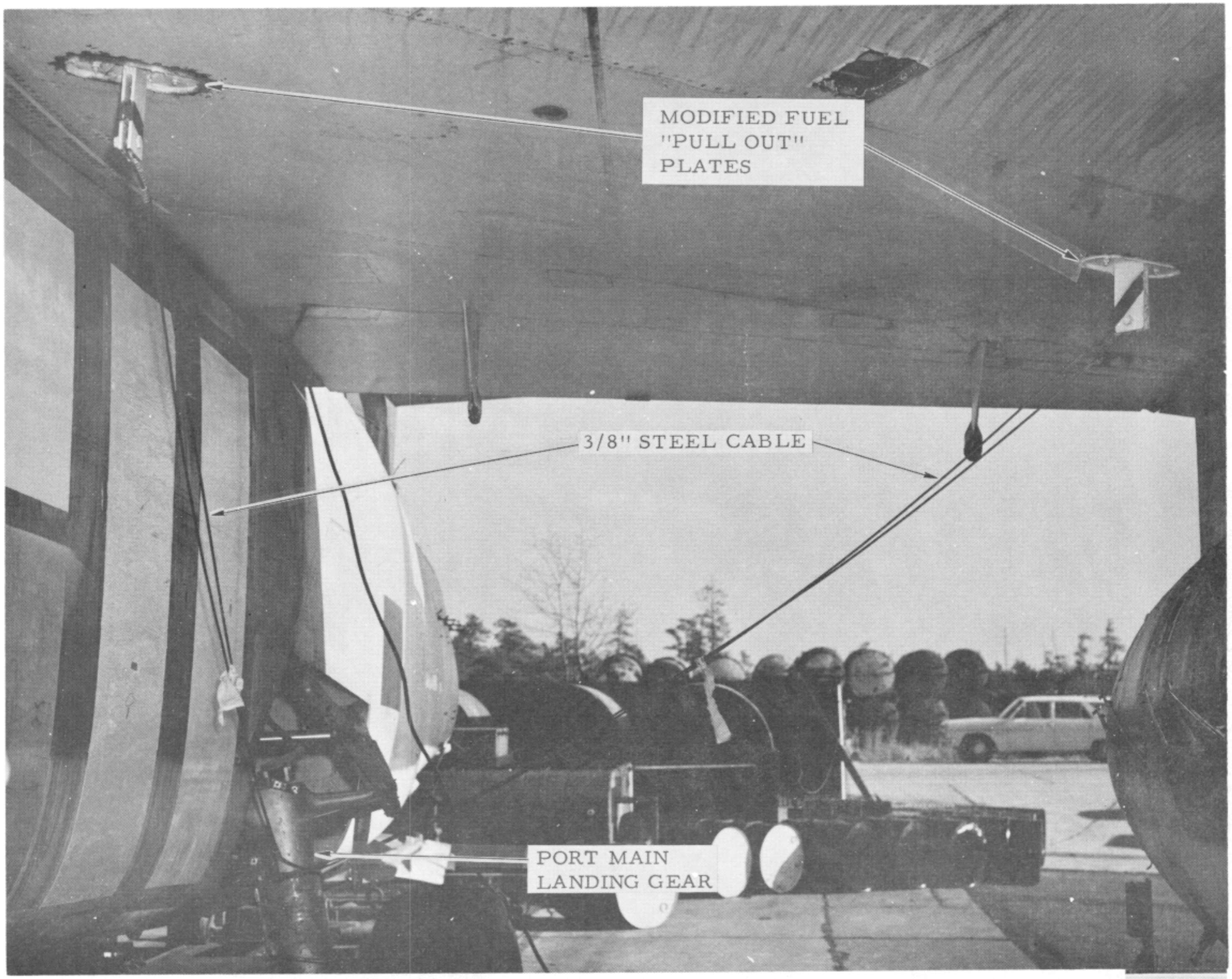

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FIGURE 21. FUEL TANK INSPECTION PLATE PULLOUT INSTALLATION

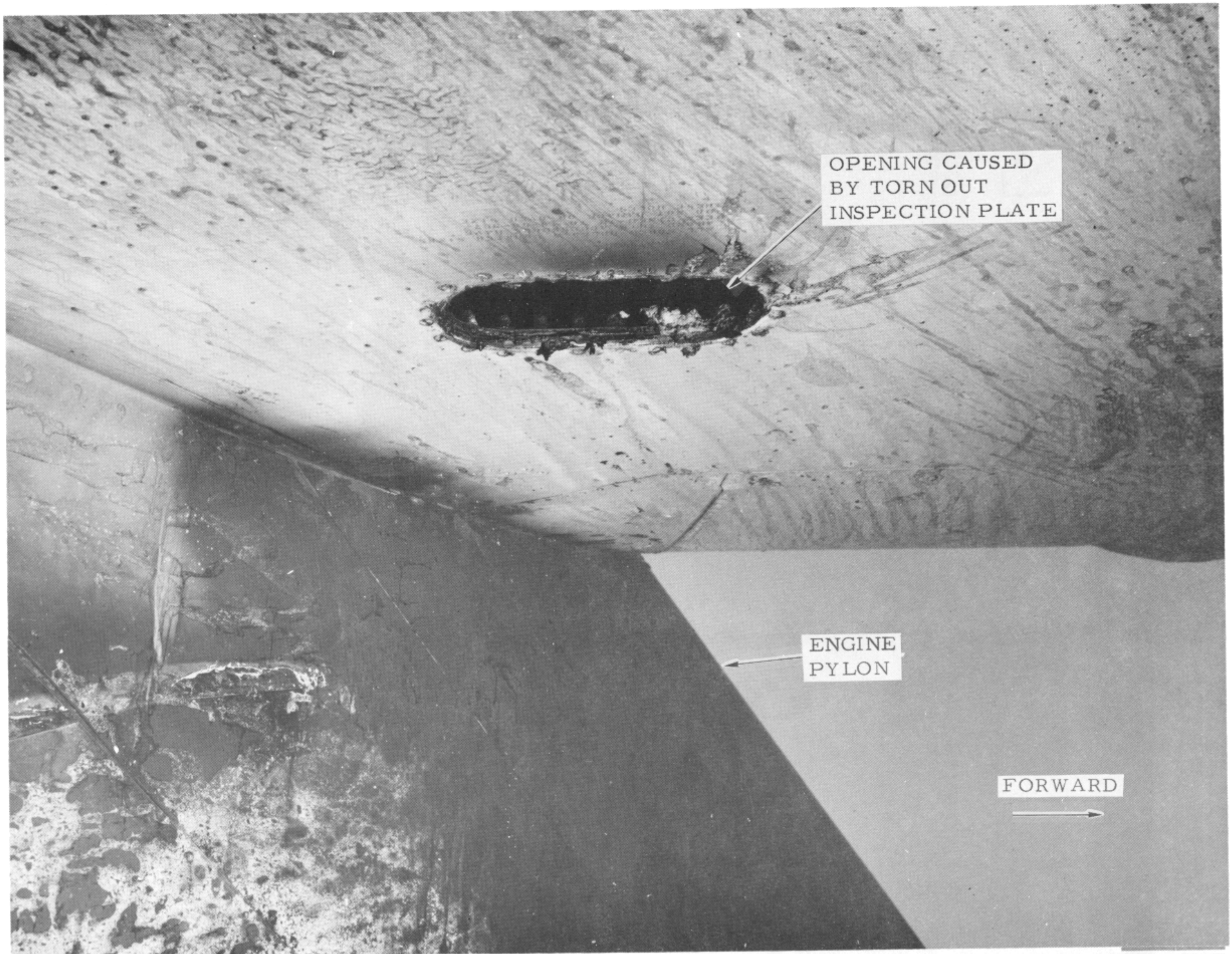

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FIGURE 22. EXISTING OPENING IN FUEL TANK AFTER INSPECTION PLATE HAD BEEN PULLED OUT

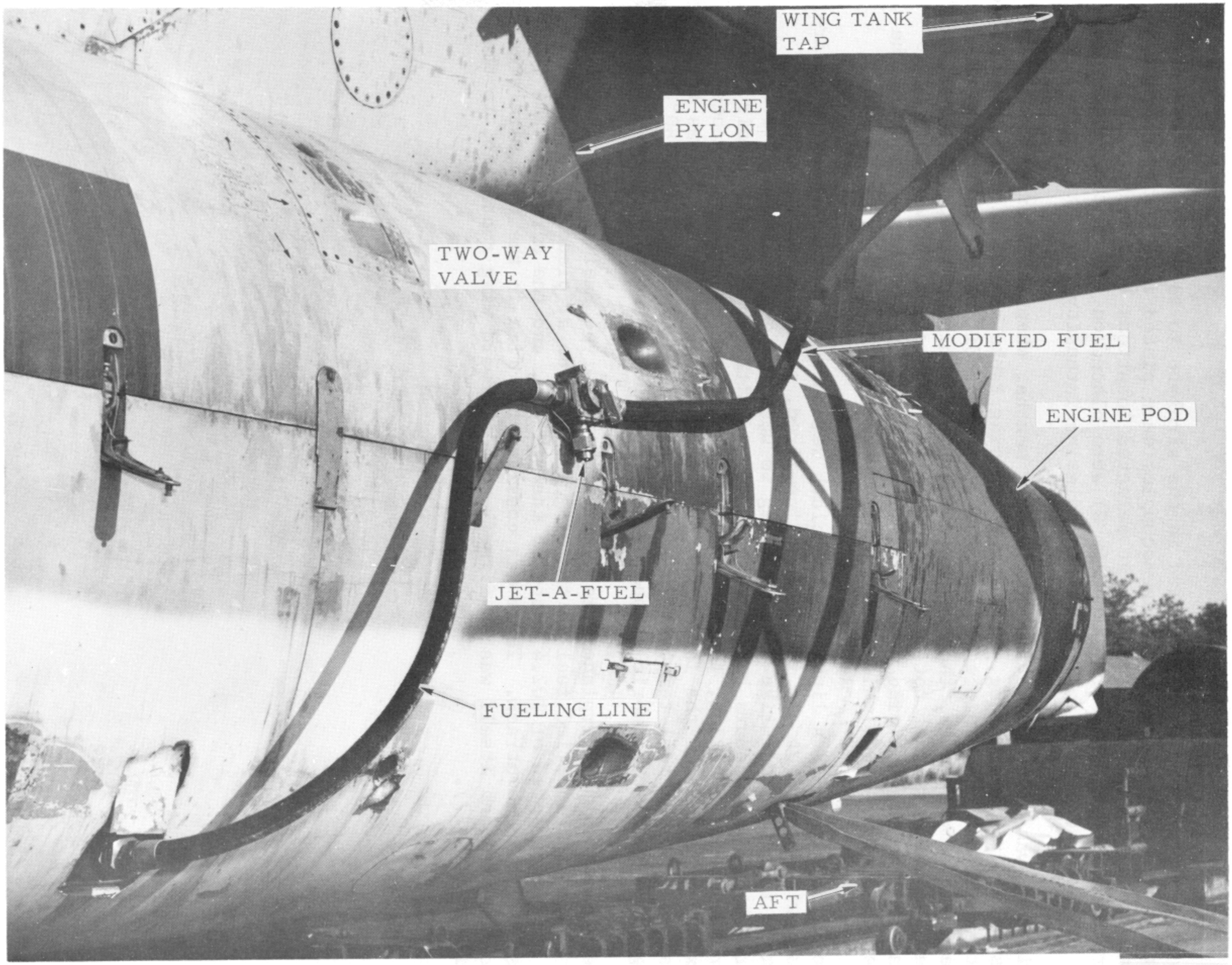

FIGURE 25. PLUMBING MODIFICATION TO ENABLE THE ENGINES TO RUN ON MODIFIED FUEL - TEST NO. 4

Four accelerometers were installed in the airplane to record crash forces during the test, and several thermocouples were placed in the airplane and against the skin of the fuselage at selected locations to measure skin temperature. Ten radiometers (appendix C) were placed on top of the range poles in the test site area to record the radiation intensity from the fire.

This test was conducted during the winter months, when the average temperature was below freezing during, and after, the wing tanks of the airplane were filled with modified fuel. Since the viscosity of the modified fuel is higher at the lower temperatures, the breakup of the fuel would be less, and it was felt that the results might not be representative. Therefore, it was decided to heat the fuel in the wing tanks and effectively increase the severity of the test conditions under which the fuel would have to perform.

Two shrouds made of a light plastic cloth were draped over the port and starboard wings, and hot air from a gasoline-fired, portable heater cart was pumped into the tent-like wing covering to heat the fuel. The heater cart was fired up about 10 hours before test time, and although the ambient air temperature was below freezing, the air temperature within the shrouds reached a maximum of 80° F for the last 3 hours of the heating process. It was therefore assumed that the fuel temperature reached a maximum temperature of 80° F. Figure 26 shows the fabric shroud draped over the starboard wing of the RB66.

The engines on the airplane were started on JP-4-type fuel and switched over to run on the modified fuel in the wing tanks. The airplane was accelerated to an impact speed of 102.4 knots by the jet car, and the airplane came to rest 600 feet downtrack. There was considerable fire in the test site area, and the airplane itself was on fire from the forward wing roots aft, to the tail (figure 27).

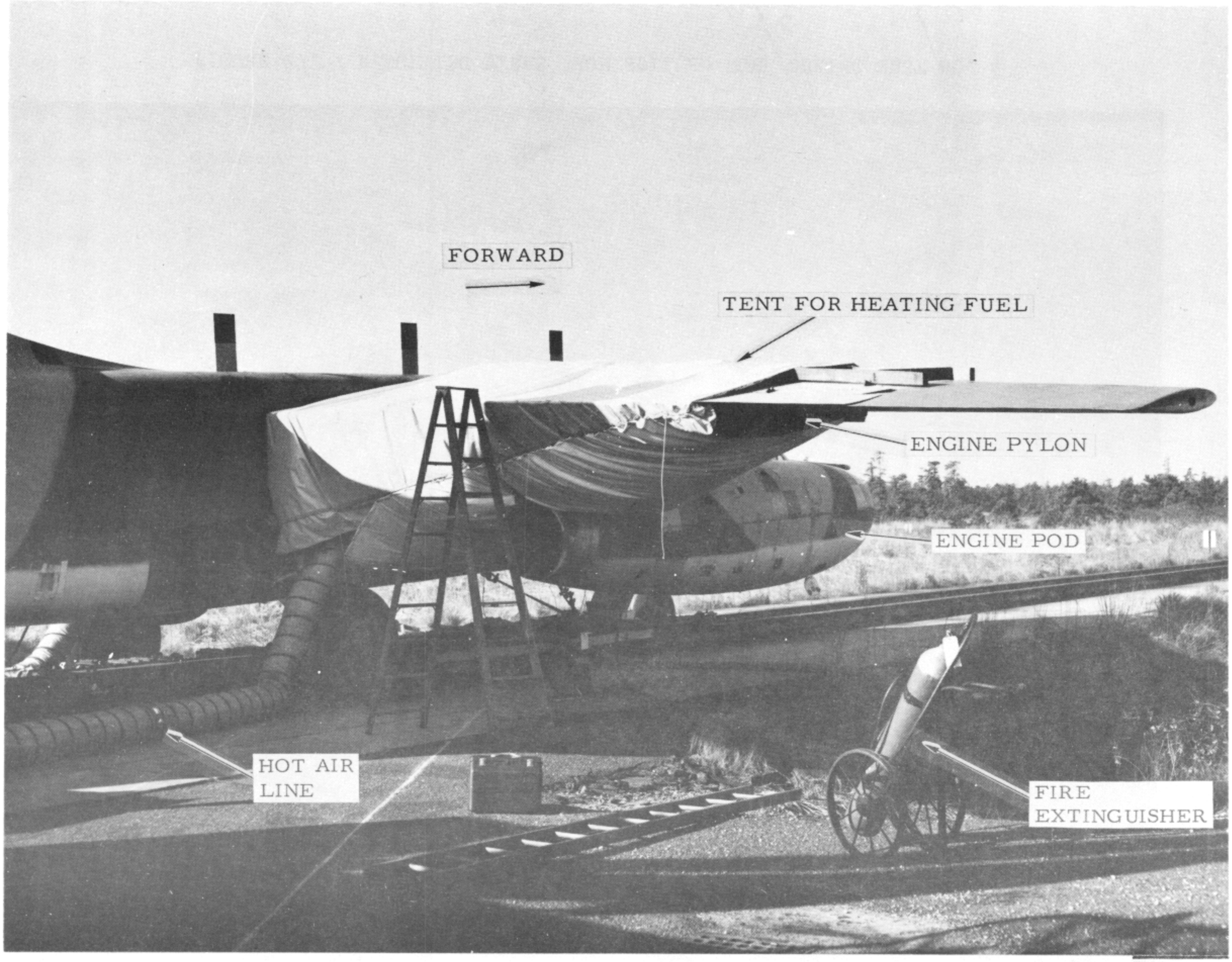

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FIGURE 26. METHOD USED TO HEAT MODIFIED FUEL BEFORE LAUNCHING AIRPLANE

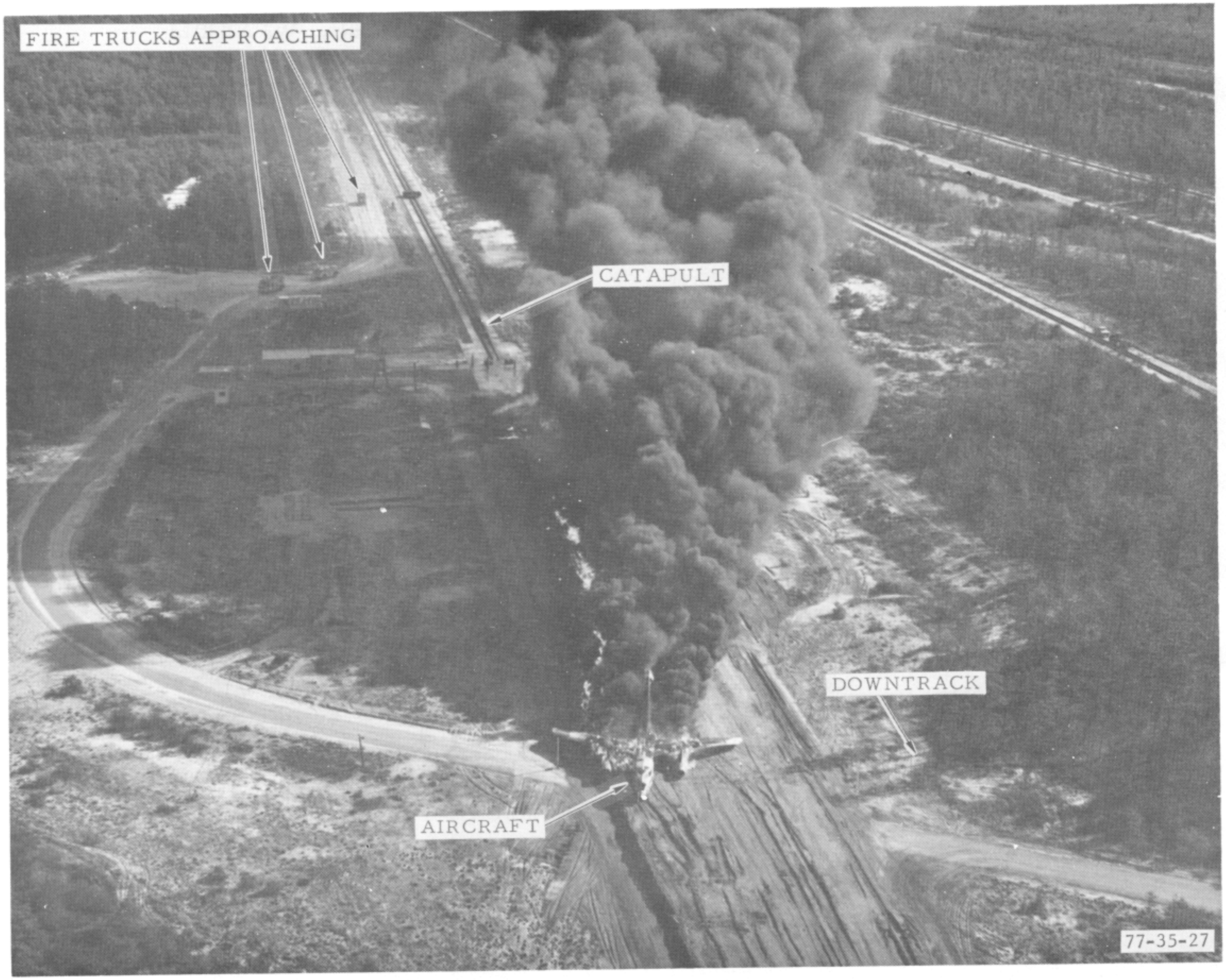

FIGURE 27. RESULTING FIRES FROM SPILLED FUEL DURING TEST NO. 4

RESULTS

TEST NO. 1.

This test was performed to determine the dynamic characteristics of the airplane in the test site area. The path of the airplane and its velocity profile were predicted by the calculations shown in appendix A, but certain assumptions had to be made relating to the dissipation of energy as the airplane structure broke up and retarding forces developed as the fuselage and nacelles ploughed through the loose earth in the test site area.

Accelerometers were installed at three locations in the fuselage and one location on the wing. Both longitudinal and vertical accelerations were to be measured, but the instrumentation system failed immediately after the airplane encountered the landing gear obstacles that forcibly tore off the two landing wheels, and the four telephone poles that ruptured the wing tank leading edges. All of the telemetering-transmitting equipment and the wet cell batteries necessary to power the equipment were installed in the after portion of the airplane tail, and it was determined that the high deceleration forces encountered as the airplane struck the obstacles caused internal damage to the wet cell batteries, thus all power was lost, and the telemetry equipment could not transmit the data.

After encountering the landing gear obstacles and the four telephone poles, the A3 airplane settled onto the 6° sloped hill in a nose-down attitude. The vertical descent of the airplane was abruptly changed as it proceeded forward, and the nose section and cockpit area of the fuselage ploughed into the 6° slope. Both engines contacted the 6° slope as the forward part of the fuselage failed, and the port engine was torn from its mount. The airplane became airborne for approximately 190 feet after it left the crest of the 6° slope, and as it settled back toward the ground, the port wing contacted the edge of the 2-foot-high earthen pillow that was constructed 200 feet downtrack from the 6° slope. The whole airplane folded upon itself and came to rest in a large compact pile of wreckage, as shown in figures 28 and 29.

Since the airplane structural integrity had been completely destroyed by the time it came to rest, this test could not be considered to be within the limits of survivability. From the high-speed motion picture coverage of the test, it was determined that the impact forces encountered by the airplane as it settled onto the 6° sloped hill were so severe that they caused vital structures, such as the wing roots and engine-pod attachment points, to fail. As the airplane left the 6° slope, it appeared to be structurally sound, but more than likely, it was not. The movement of all the parts, wings, fuselage, and tail, through the air at the same velocity made it appear intact, but when the airplane settled back to ground level, there was not enough structural strength left to hold the various parts in their relative positions, and the airplane came to rest in a heap.

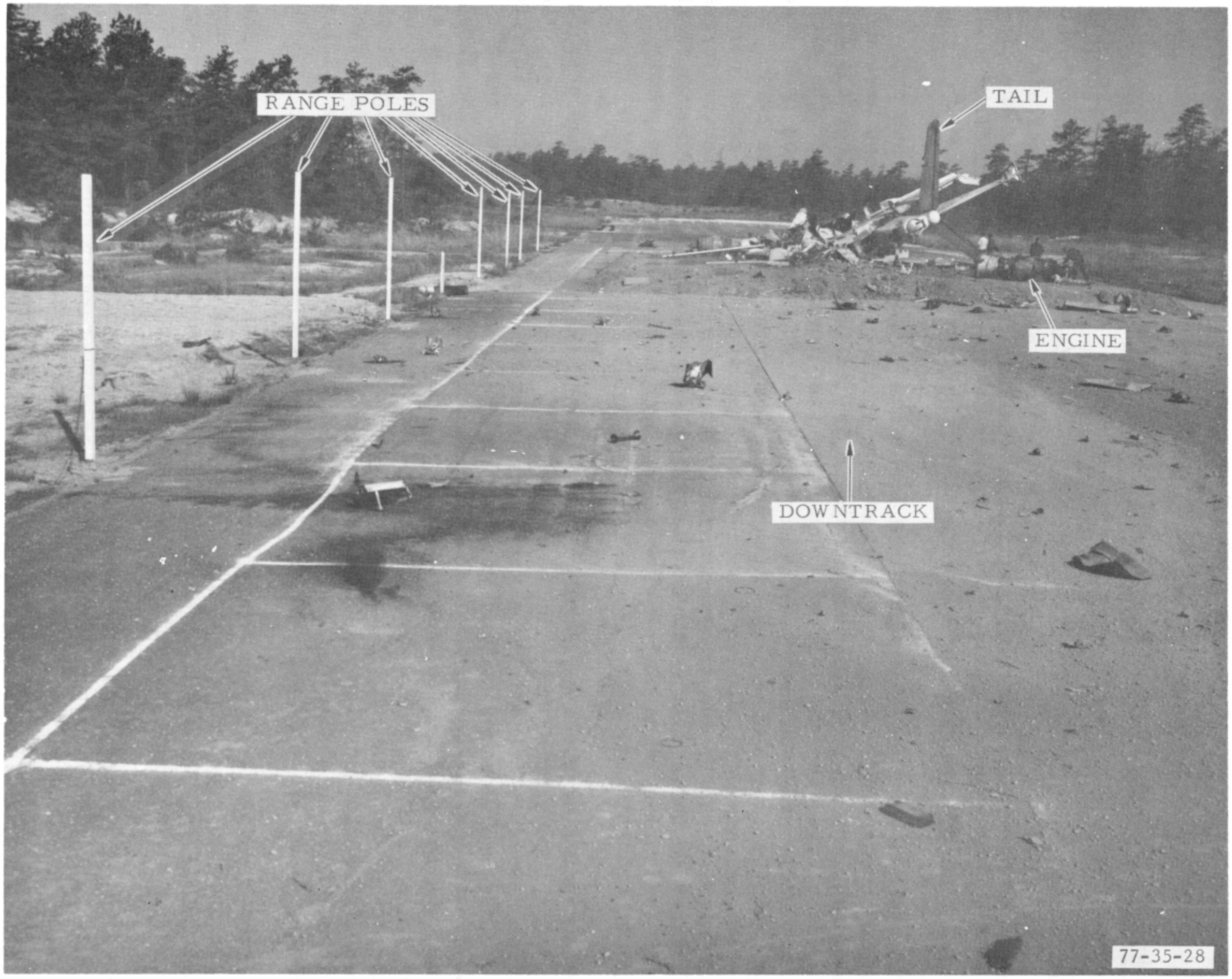

FIGURE 28. WRECKAGE OF A3 AIRPLANE OF TEST NO. 1 (LEFT REAR VIEW)

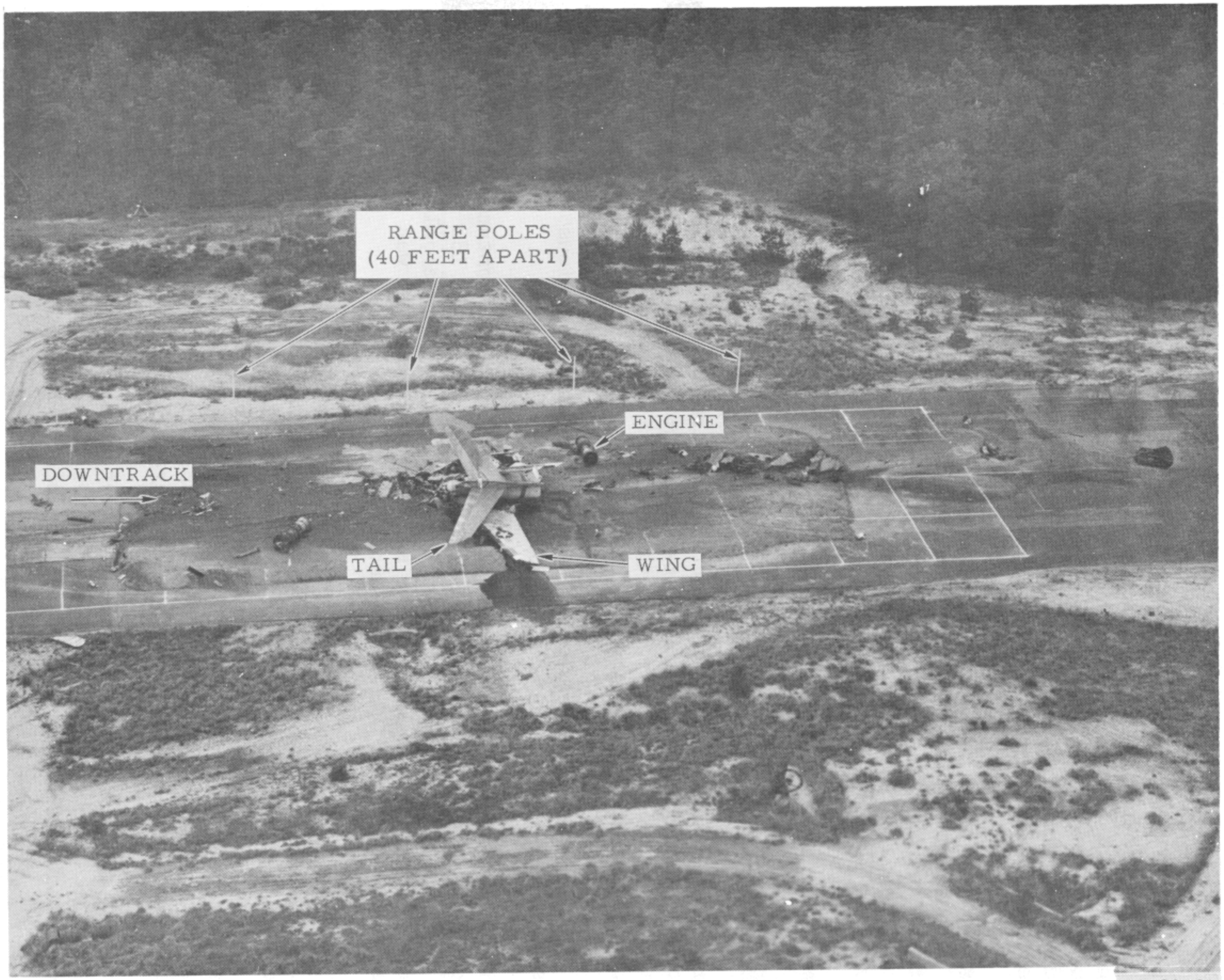

FIGURE 29. WRECKAGE OF A3 AIRPLANE OF TEST NO. 1 (OVERHEAD RIGHT SIDE VIEW)

The spray pattern of the water was very profuse from the four ruptured parts of the wing tank leading edge as the airplane came off the crest of the 6° sloped hill. The deceleration forces, that were so effective in destroying the structural integrity of the airplane itself, also caused the water inside the wing tanks to impinge up against the wing tank leading edge where the tanks were ruptured by the four telephone pole obstacles. As a result, the hydraulic force created by the situation forced the liquid out of the ruptured tanks and a very satisfactory simulated fuel mist pattern was created.

Although no data were obtained from the accelerometers installed in this airplane, because of power failure to the transmitting equipment, it was determined that the forces encountered by the airplane as it contacted the 6° slope were too great and caused unwanted structural failure. From the results of this test, it was decided to reduce the slope of the first hill and to construct the base of a 15° sloped hill (figure 8) at the point where the wreckage came to rest on this test.

TEST NO. 2.

The airplane was instrumented with accelerometers at various locations, and the data were transmitted by telemetry equipment mounted in the tail section of the airplane. The entire travel of the airplane through the test site area was recorded on high-speed motion picture film. This film was analyzed to obtain the velocity profile of the airplane as it traversed the test site area. Figure 30 is a plot of the velocity of the airplane at various distances along the test site.

The information in figure 30 starts just as the airplane contacts the four telephone-pole wing obstacles. The airplane was moving at 183 feet per second (ft/s) (108 knots), and after contacting the four telephone poles and the main wheel obstacles that sheared off the main landing gear, it was slowed down 13 ft/s (8 knots) to 170 ft/s (100 knots).

After leaving the 3° slope, the airplane was traveling at approximately 125 ft/s (74 knots) when it contacted the 15° slope, where it came to rest just over the crest of the hill, as shown in figures 11, 12, 13, and 14. From figure 30, it can be seen that most of the energy was dissipated at the 15° slope where the airplane went from 125 ft/s to zero; whereas, it went from 183 ft/s to 125 ft/s for a total loss of 57 ft/s (34 knots) for the entire length of the test site area traversed before it encountered the 15° slope.

Figure 31 is a plot of the distance the test aircraft traveled through the test site area versus the time elapsed after the airplane encountered the landing gear obstacles and the telephone poles. From figure 31, it can be seen that the airplane had encountered the landing gear obstacles, sheared off the four telephone poles, slid up the 3° hill, and was airborne beyond the crest of the hill in a time span of 0.8 seconds. By the end of 4.3 seconds, the airplane had slid up the 15° slope and had come to rest on the crest.

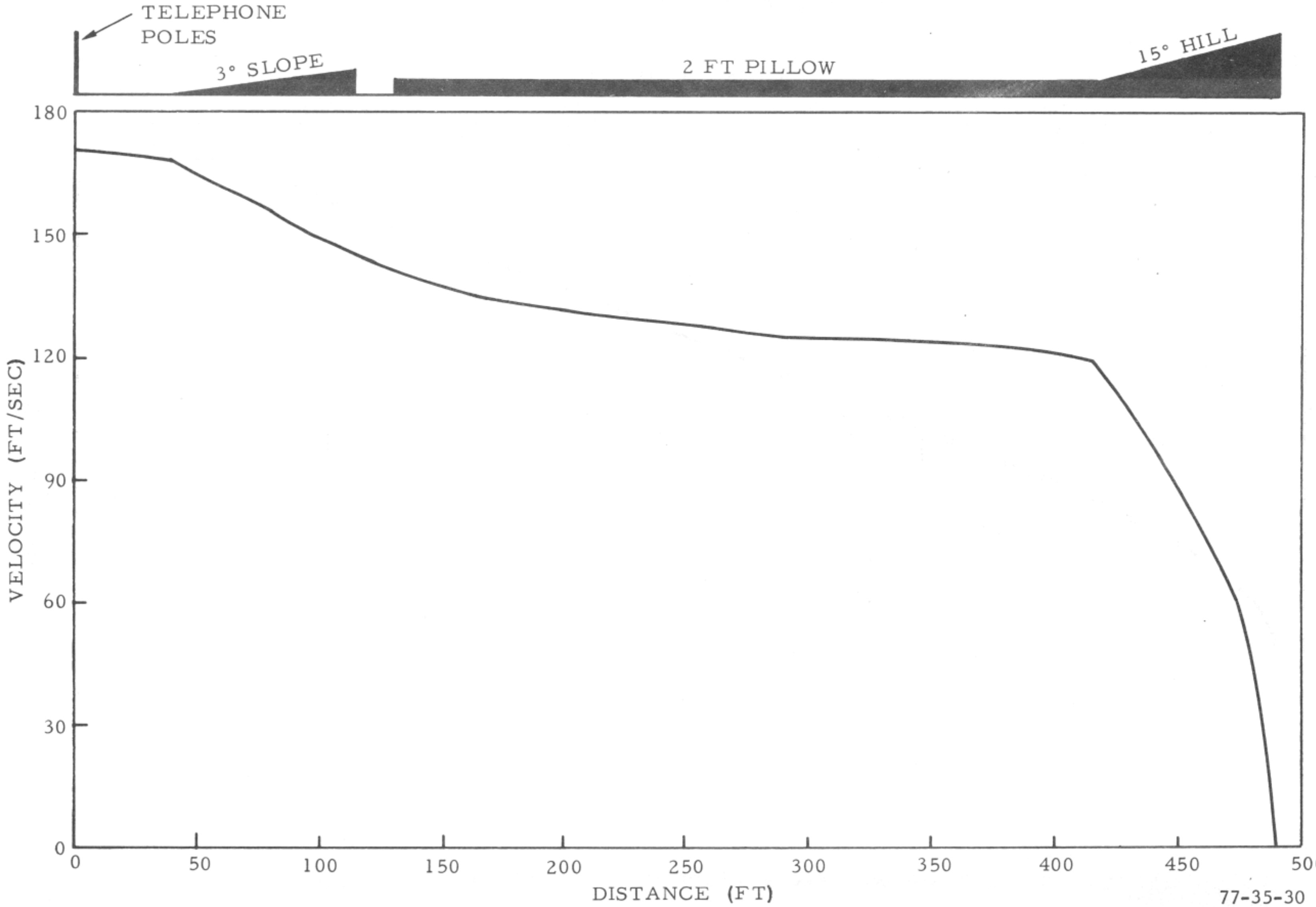

FIGURE 30. VELOCITY-VERSUS-DISTANCE PROFILE FOR TEST NO.2

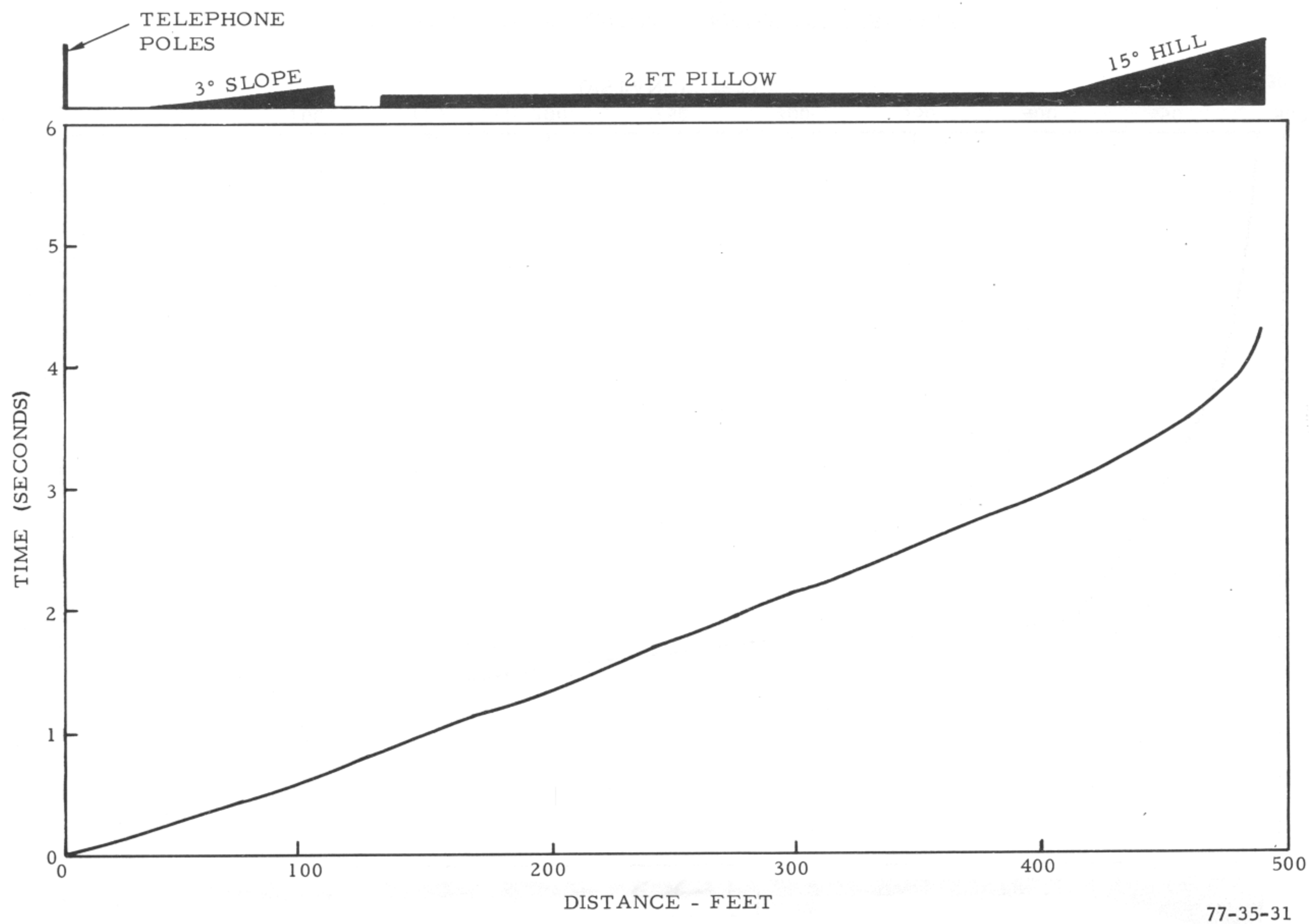

FIGURE 31. TIME-VERSUS-DISTANCE PROFILE FOR TEST NO. 2

For this test, the airplane was equipped with accelerometers having a range of zero to 100 g's, sensing both the longitudinal and vertical accelerations at three locations in the fuselage and one location on the left wing. Table 2 is a summary of the maximum extremes of accelerations recorded on the time-history plots contained in appendix B. The station identification and acceleration sign interpretation are shown in figure 32.

The telemetry equipment was installed in the tail of the airplane and when the airplane encountered the 15° slope the tail broke away, and all data transmission ceased. Therefore, no data were obtained on the deceleration forces the airplane encountered as it contacted the 15° slope.

From the high-speed motion picture coverage of the test, it was determined that the spray was coarse and that the misting tendencies and the spray patterns of these modified fuels were markedly reduced from those in the first test where plain water was used in the wing tanks. The fuel temperature for this test was 74°.

The airplane came to rest in a nose-down attitude passing over the crest of the 15° slope, and the modified fuel that remained in the wing tanks drained out. From the spillage on the ground after the airplane came to rest, it was estimated that about half of the 1,500 gallons of fuel that was aboard the airplane was released along the test site area, and the remaining half of the fuel was spilled out of the tanks when the airplane came to rest.

TEST NO. 3.

This was the first test of this series to have ignition sources placed along the intended route of the test airplane. The ignition sources used consisted of forty (40) highway safety flares with a burning time of 45 minutes, and eight (8) smudge pots with flame bases flush with the test site surface to provide ignition sources for any pools of fuel that might form during the test. The flame profile of the safety flares is shown in figure 13. The ignition sources were spaced along the test site as shown in figure 8.

The test airplane was accelerated to a speed of 190 ft/s (112.1 knots) by the four jet engines on the NATF pusher car. The engines on the test airplane were not running for this test. After release by the pusher car, the test airplane had slowed down to 177 ft/s (104.6 knots) as it entered the test site area and encountered the four telephone poles and the main landing gear obstacles.

The airplane passed over the crest of the 3° slope at a speed of 138 ft/s (82 knots) and settled onto the 2-foot-high earthen pillow, where it encountered the burning safety flares and the kerosene smudge pots.

The velocity of the airplane at other points along the test area can be ascertained from figure 33. It was traveling at a speed of 70 ft/s (41.5 knots) as it reached the base of the 15° sloped hill and then came to a complete stop at the crest.

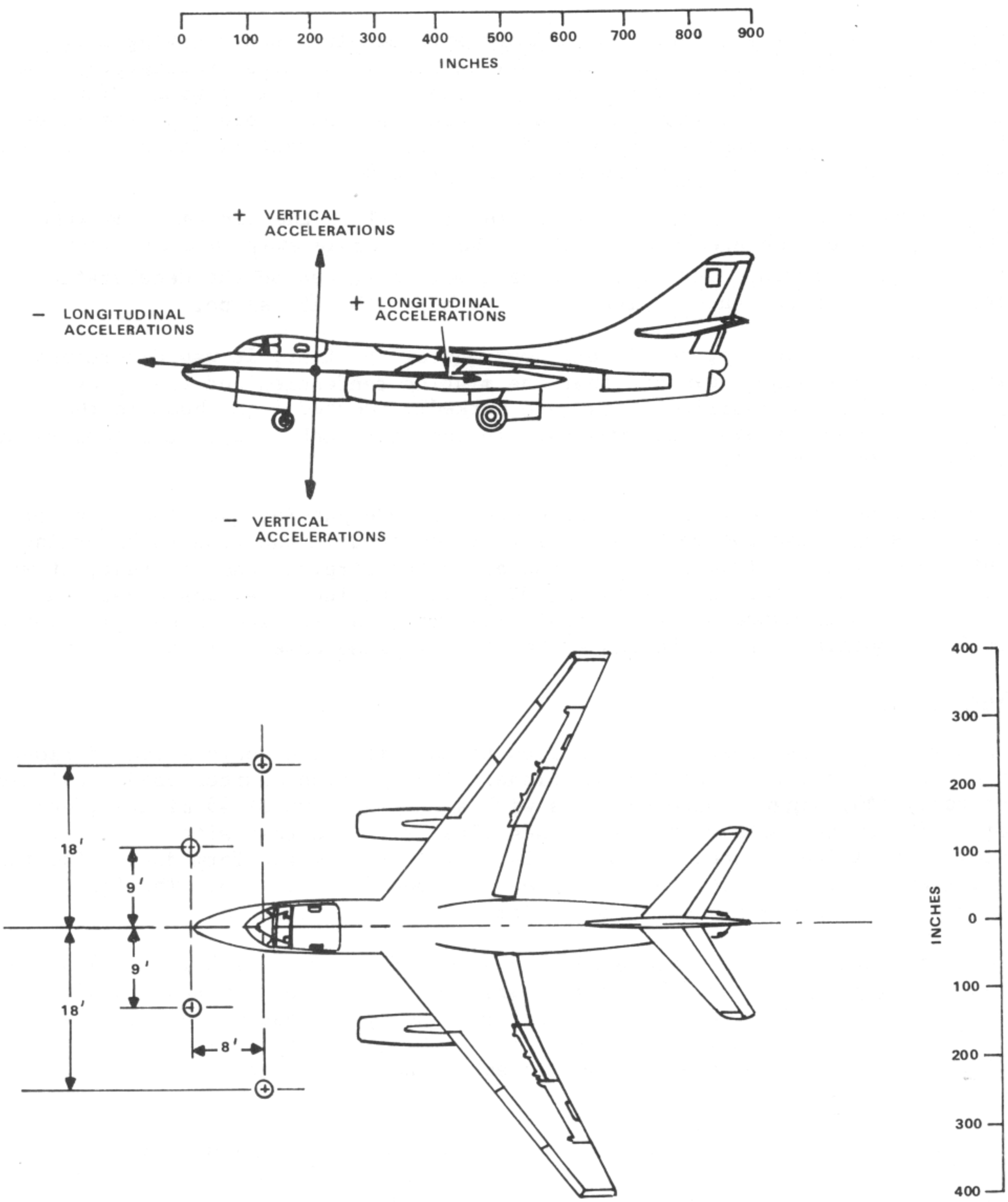

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FIGURE 32. ACCELERATION AND STATION LEGEND FOR THE A3 AND RB66 AIRPLANES

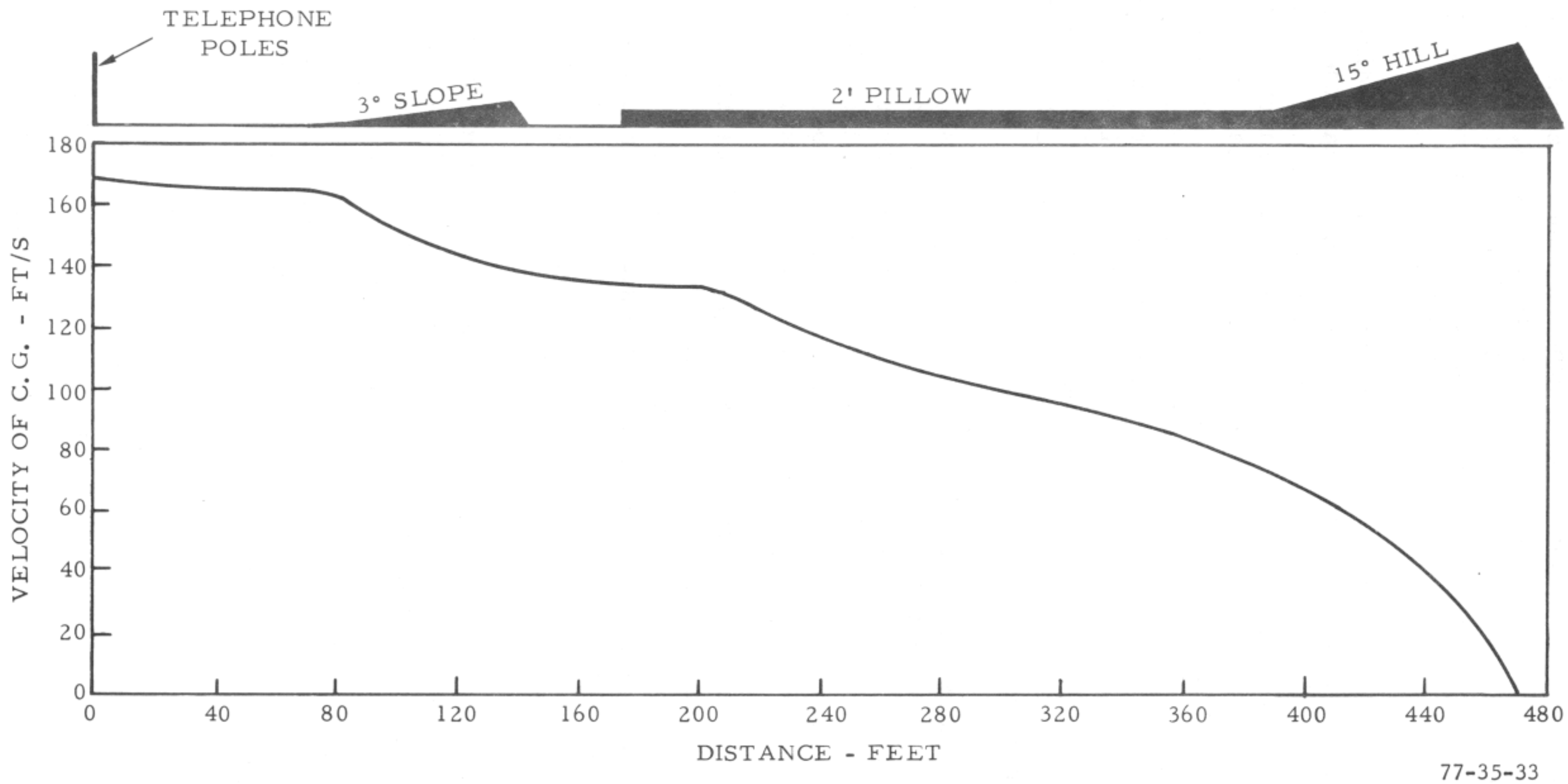

FIGURE 33. VELOCITY-VERSUS-DISTANCE PROFILE FOR TEST NO. 3

Figure 34 is a plot of the distance the test aircraft traveled through the test site area versus the time elapsed after the airplane encountered the landing gear obstacles and the telephone poles. From figure 34, it can be seen that the airplane had slid over the crest of the 3° sloped hill within 0.8 seconds after encountering the landing gear obstacles and came to a complete halt on the crest of the 15° sloped hill in 6.4 seconds.

The modified fuel was flowing from the holes put in the leading edge of the wings by the four telephone poles, but it was judged that the rate of flow was not very great, and the misting of the fuel was very small compared to the first test in this series, where plain water was used in the integral wing tanks of the A3 test airplane. The reduced misting tendencies in this test are attributed to the dilatant characteristics of the modified fuel, which reduces the misting tendency of the fuel by increasing viscosity as flow rates increase, and also to the reduced deceleration forces the test airplane experienced as it contacted the first hill. In the test with the A3 airplane, the hill had a 6° slope and imparted more restraining force on the airplane, thus causing the hydraulic head of the liquid to be much higher where the tanks were ruptured, causing a higher rate of flow through the openings.

From the high-speed motion pictures taken during the test, it was determined that the spilled fuel encountered the burning flares as the airplane skidded along the test site, but no fires developed. Even after the airplane came to rest on the crest of the 15° sloped hill, fuel continued to flow from the ruptured tanks, and it was estimated that less than 40 percent of the fuel had spilled out of the tanks before the airplane came to rest.

An attempt was made to ignite the standing pools of fuel that were being formed by the leaking fuel tanks, but the pools proved extremely difficult to light and after several minutes of this effort only small fires had been started. These burned for a short time and then extinguished themselves. An attempt was made to ignite the fuel as it poured directly from the ruptured tank. Here the fuel did ignite, but the flame was not considered vigorous and was easily extinguished by the fireman.

The acceleration values developed during this test are given in table 3. The forces were very high in the cockpit floor area, ranging upwards to 66 g's, but the transducer had been torn from its mountings during the test, and these values could be abnormally high. The maximum value recorded at the pilot's seat was 22 g's, and the acceleration force on the dummy itself, strapped in the pilot seat, was 11 g's. These values can be considered to be well within the range of survivability. The telemetering equipment stopped functioning before the airplane encountered the 15° slope.

TEST NO. 4.

Figure 35 is a plot of the speed of the test aircraft at various positions on the test site area, and by using figure 36, a correlation between the position of the airplane, its speed at that instant, and the time that had elapsed before the airplane got to that position, can be determined. For instance, from

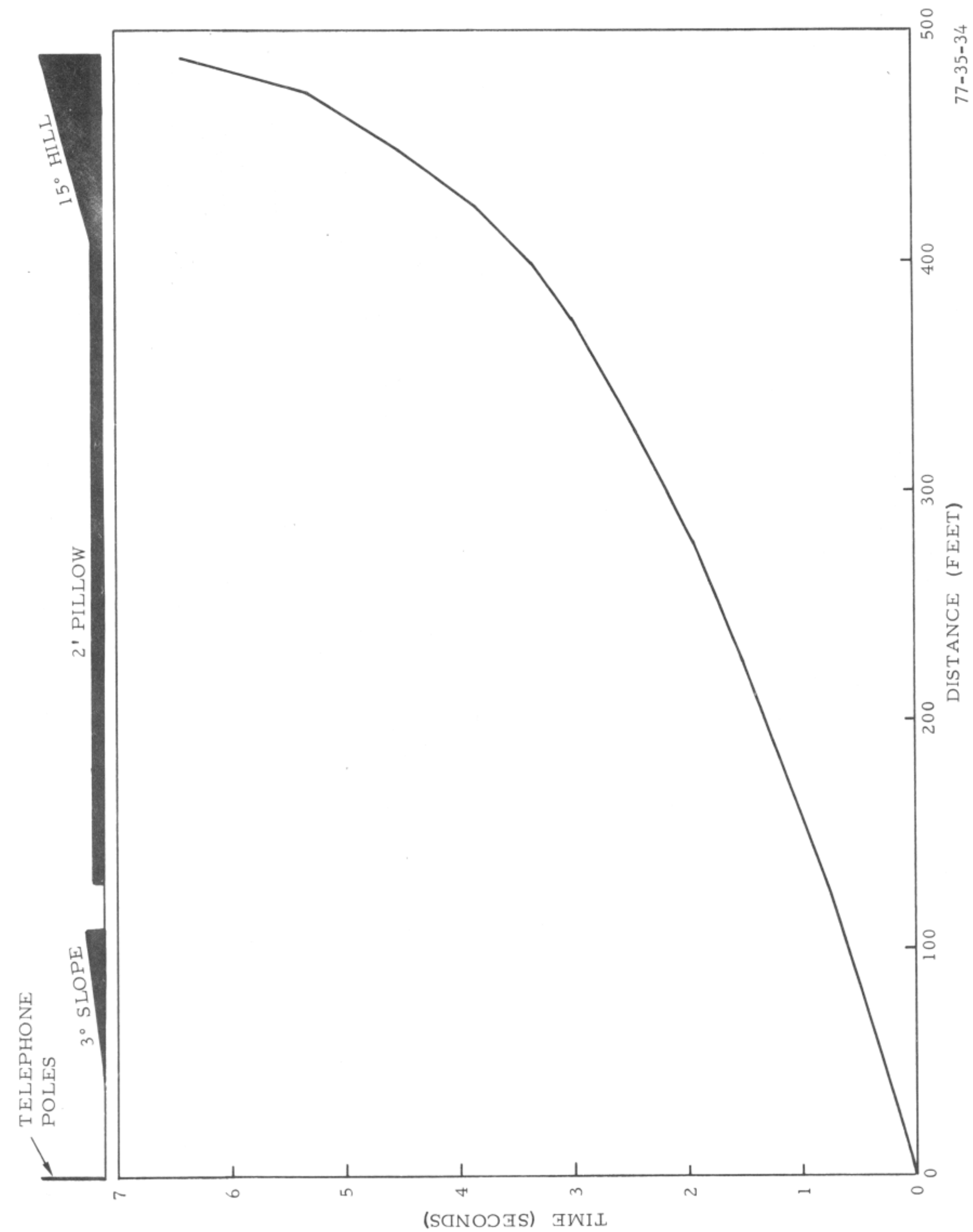

FIGURE 34. TIME-VERSUS-DISTANCE PROFILE FOR TEST NO. 3

TABLE 3. TEST NO. 3 - RB66 AIRPLANE

PEAK ACCELERATION VALUES** - G's

Description	Station*	Impact With Telephone Poles		Impact With 3° Slope		Impact With 15° Slope		Remarks
		Long	Vert	Long	Vert	Long	Vert	
Cockpit Floor	134	+24 - 8	+ 6 -37	+42 -14	+56 -66	- -	- -	Accelerometer was torn from its mountings at first impact point.
Tunnel Roof	245	+ 7 - 2	+16 -10	+58 -12	+42 -45	- -	- -	Signals failed after 1.4 seconds on long. axis and 2.4 seconds on vert. axis.
Aft Bomb Bay	450	+24 - 2	+13 -16	+12 - 7	+10 - 3	- -	- -	
Electronics Bay	770	+13 -11	+ 6 - 6	+40 -10	+48 -12	- -	- -	
Left Wing	WL-165	+15 - 2	+20 -18	+12 - 1	+13 - 6	- -	- -	
Pilot Seat	114	+14 - 5	+ 6 - 6	+22 -11	+11 -22	- -	- -	
Dummy Pilot	114	+ 3 - 3	- -	+ 9 -11	- -	- -	- -	Vertical channel signals failed to record.

DEFORMATION IN INCHES

Description	Station	Impact With Poles		Impact With 3° Slope		Airborne		Impact With Ground		Remarks
		Crushing (in.)	Extension (in.)	Crushing (in.)	Extension (in.)	Crushing (in.)	Extension (in.)	Crushing (in.)	Extension (in.)	
Transducer No. 1	177	-	-	0.4	-	-	7.2	-	-	Telemetry ends after 0.9 seconds.
Transducer No. 2	346	-	-	1.6	2.4	-	6.2	-	7.2	Telemetry ends after 2.2 seconds.
Transducer No. 3	410	-	-	-	-	3.5	-	10.4	-	Telemetry ends after 3.5 seconds.
Transducer No. 4	519	10.8	-	-	1.8	6.3	-	14	-	Telemetry ends after 1.8 seconds.

NOTES:

* See figure 32 for general location on airplane.

** See appendix C.

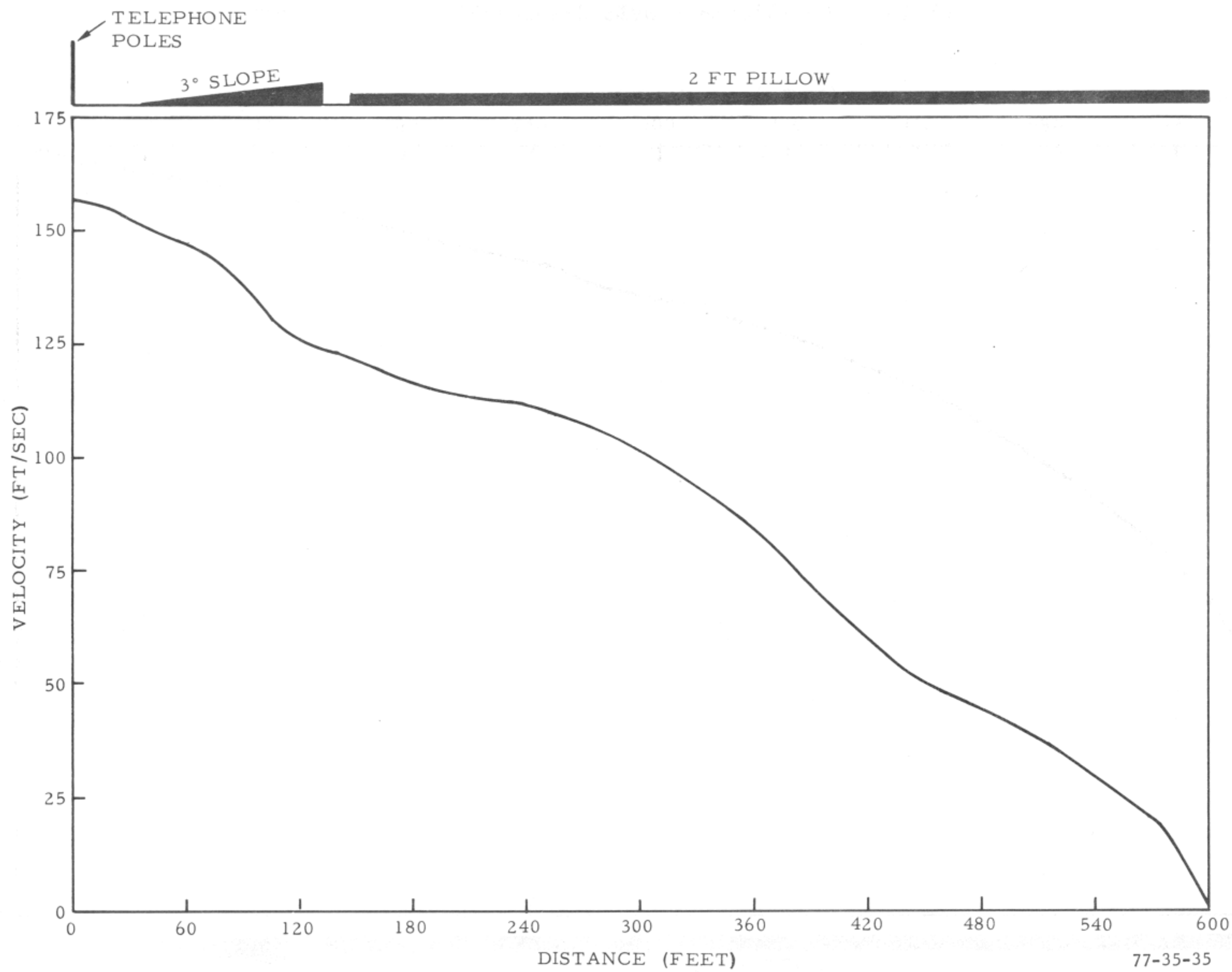

FIGURE 35. VELOCITY-VERSUS-DISTANCE PROFILE FOR TEST NO. 4

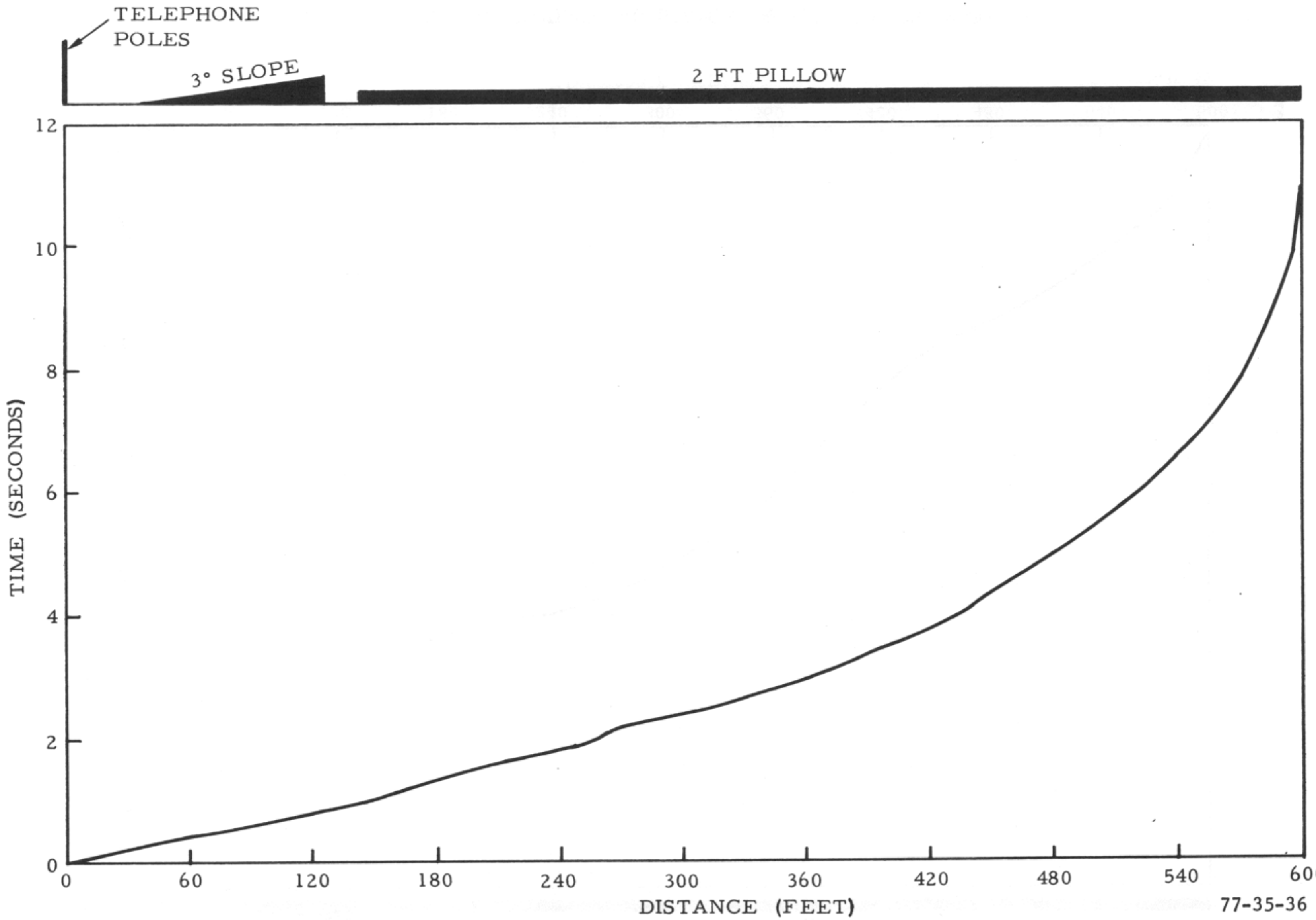

FIGURE 36. TIME-VERSUS-DISTANCE PROFILE FOR TEST NO. 4

figure 35, it can be ascertained that the speed of the aircraft as it left the crest of the 3° sloped hill was 125 ft/s, and from figure 36, it can be determined that it took the airplane 0.95 seconds to get there.

The four inspection plate plugs that were attached to the landing gear by stainless steel cable lanyards were pulled out successfully and produced openings in the bottom of the wing tanks, as shown in figure 22. The four telephone poles caused large gaping openings in the leading edge of the wings, as shown in figure 24. The combination of these openings allowed copious quantities of fuel to pour out of the wing tanks as the airplane ploughed into the 3° sloped hill (figure 37). From the high-speed motion pictures taken during the test, it was determined that the fuel spewing from the ruptured tanks was ignited between the fuselage and the starboard engine which was running at the time. A fire developed at ground level in line with and aft of the starboard engine as the airplane slid along the 3° sloped hill and reached the crest of the hill. The propane torches burning at the top of the 3° sloped hill ignited the spilling fuel on the left side of the airplane. From that point on, the airplane was enveloped in flame as it passed down the test site area, and it continued to burn after it came to rest until the flames were extingushed by the fire fighters on the scene.

The airplane traversed through 90 percent of the test site area in 6.6 seconds and came to a complete stop 10.8 seconds after it had encountered the telephone pole barrier that had ruptured the wing tanks. Yet from the data presented in table 4, the temperature probes positioned inside the fuselage at various locations did not sense maximum temperature rises until 60 to 90 seconds after impact. Temperatures in the aft bomb bay and electronics compartment areas started to rise 5 seconds after impact, but had only reached 420° to 440° F when the airplane came to rest 5.8 seconds later. Thirty seconds after impact, the temperatures at these two locations were down to 260° F before rising again to higher values, indicating that the fire surrounding the aircraft had not penetrated the fuselage skin until a considerable time after the airplane came to rest.

The fire had to be extinguished by the fire-fighting crew that was standing by at the crash site area. Figures 38 and 39 show the hulk of the airplane after the fires were extinguished. The fires were confined to that part of the airplane aft of the wing leading edge mainly because the forward velocity of the airplane caused the spilled fuel and flames to propagate aft, and also because the airplane came to rest with its nose pointed into the prevailing wind. Therefore, the cockpit area of the fuselage was relatively free of fire and heat.

The greatest acceleration loads occurred in the cockpit area at station 134 when the airplane encountered the 3° sloped hill. At this time, the main wheels had been sheared off by the landing gear obstacles, and the nosewheel gear had been sheared off at the base of the 3° sloped hill, so that the nose of the airplane was the first to contact the hill, which caused the high loads to be developed in the cockpit area.

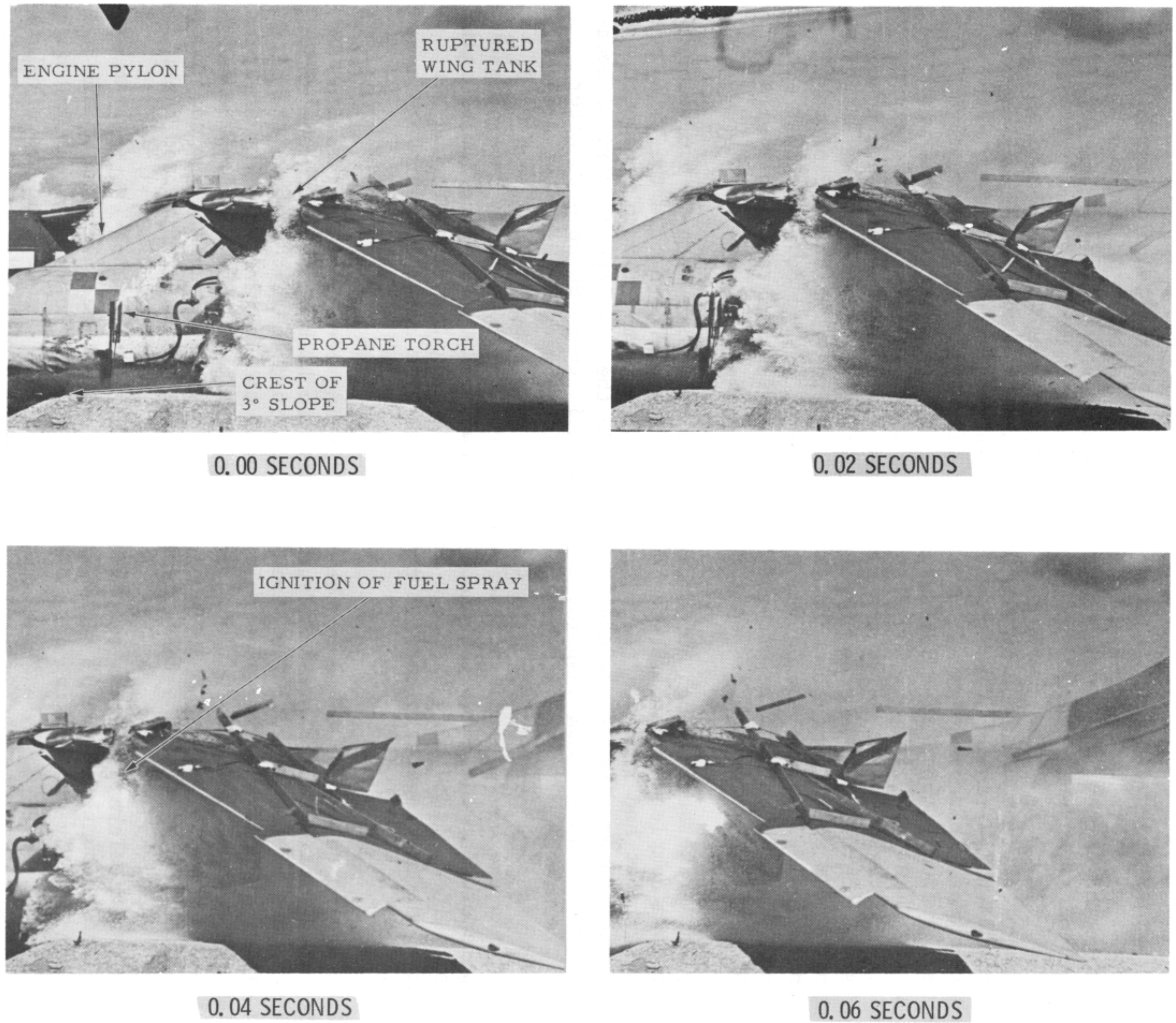

FIGURE 37. FUEL RELEASED DURING FULL-SCALE CRASH TEST

TABLE 4. TEST NO. 4 - RB66 AIRPLANE

PEAK ACCELERATION VALUES**

Description	Station*	Impact With Telephone Poles Long g's	Impact With Telephone Poles Vert g's	Impact With 3° Slope Long g's	Impact With 3° Slope Vert g's	Sliding On Earthen Pillow Long g's	Sliding On Earthen Pillow Vert g's	Remarks
Cockpit	134	+19 - 6	+ 6 - 2	+36 - 5	- -	- -	- -	Telemetric discontinuity occurred on impact with 3° slope.
Aft Fuselage Area	770	+28 - 6	+ 6 -10	+13 - 7	+14 - 8	+ 7 - 6	+ 3 -10	
Left Wing	W.L. -165	+18 - 7	+15 -28	+13 - 3	+13 -20	+ 5 - 1	+ 7 - 2	

SURFACE TEMPERATURES

Temperature Rise After Impact (° F)

		5 Seconds	6.6 Seconds	10.8 Seconds	15 Seconds	30 Seconds	60 Seconds	90 Seconds
Thermocouple No. 2	326 (Starboard)	-	-	-	-	20	40	60
Thermocouple No. 3	342 (Starboard)	-	-	-	-	-	60	60
Thermocouple No. 4	363 (Starboard)	-	-	-	-	-	20	20
Thermocouple No. 8	321 (Port)	-	-	-	20	80	360	570
Thermocouple No. 9	341 (Port)	-	-	-	20	180	1320	1380
Thermocouple No. 10	357 (Port)	-	-	20	60	220	400	540

COMPARTMENT AIR TEMPERATURES

Temperature Rise After Impact (° F)

		5 Seconds	6.6 Seconds	10.8 Seconds	15 Seconds	30 Seconds	60 Seconds	90 Seconds
Pilot Compartment	123	-	-	-	-	20	20	20
Fwd. Bomb Bay	300	-	-	90	120	340	1360	1400
Aft Bomb Bay	450	20	165	420	400	260	460	560
Electronics Compartment	600	20	360	440	400	260	1160	1200

NOTES:

* See figure 32.
** See appendix C.

74-38-37

FIGURE 38. LEFT-SIDE VIEW OF RB66 AIRPLANE USED IN TEST NO. 4

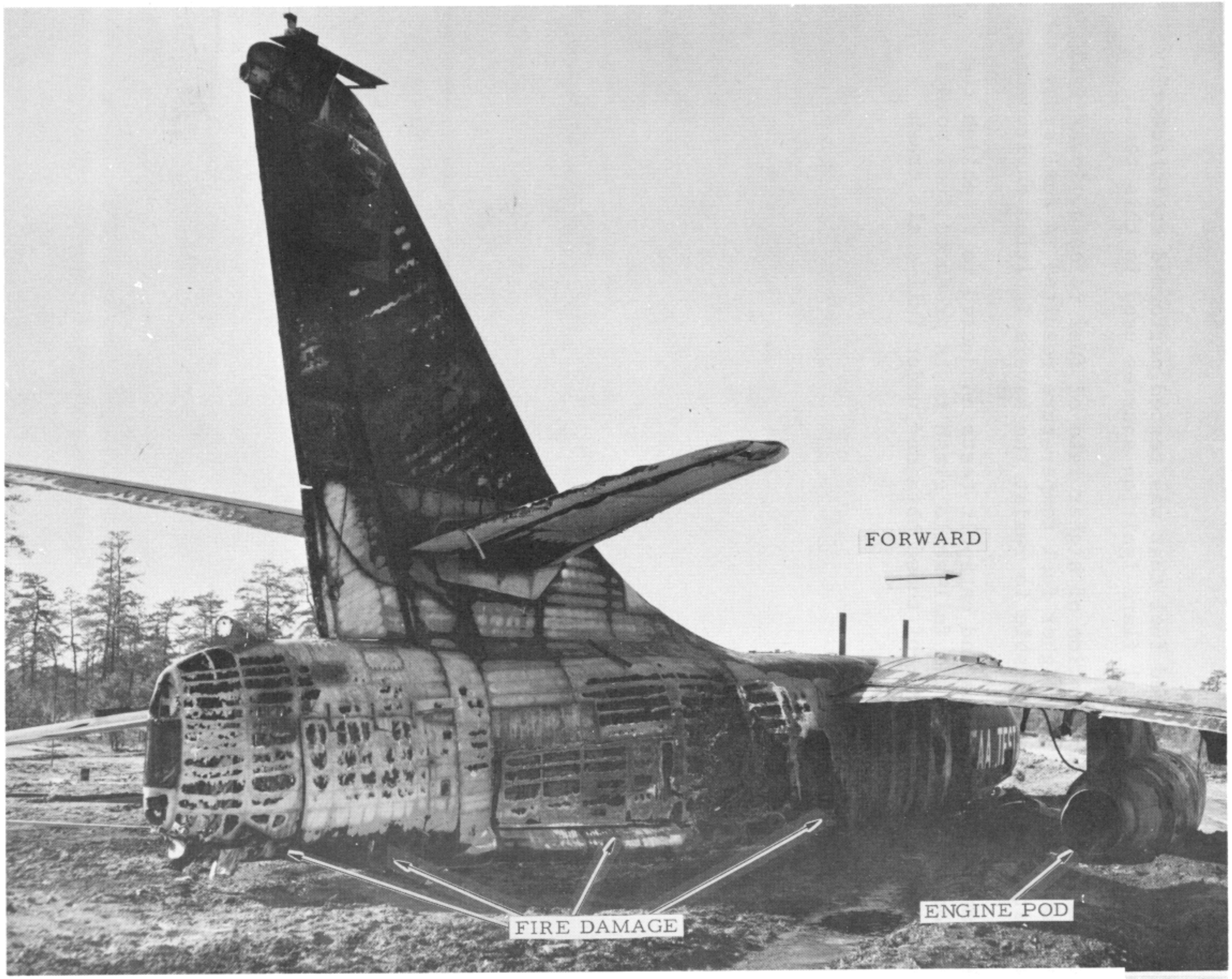

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FIGURE 39. RIGHT-SIDE VIEW OF RB66 AIRPLANE USED IN TEST NO. 4

CONCLUSIONS

1. Test No. 3: The modified fuel that was tested performed satisfactorily in the presence of the signal flare ignition sources used in this test.

2. Test No. 4: The combination of higher rate of fuel release, lower fuel additive concentration and higher fuel temperature resulted in fuel spray characteristics which were unable to resist the severe ignition environment.

3. The small-scale air gun and catapult tests conducted to establish the additive concentration for the full-scale crash tests produced fuel sprays which were not representative of those obtained under full-scale crash conditions.

APPENDIX A

CONSIDERATIONS LEADING UP TO THE SELECTION OF THE TEST SITE AREA CONFIGURATION

Since a large part of the cost of the project will be spent in construction of the crash site, any scheme to reduce the cubic yardage of fill to be moved will be advantageous, as long as it gives the decelerating forces required. Therefore, it appears that the initial idea of using one hill should be abandoned and a crash site of two separate hills should be planned (see figure A-1).

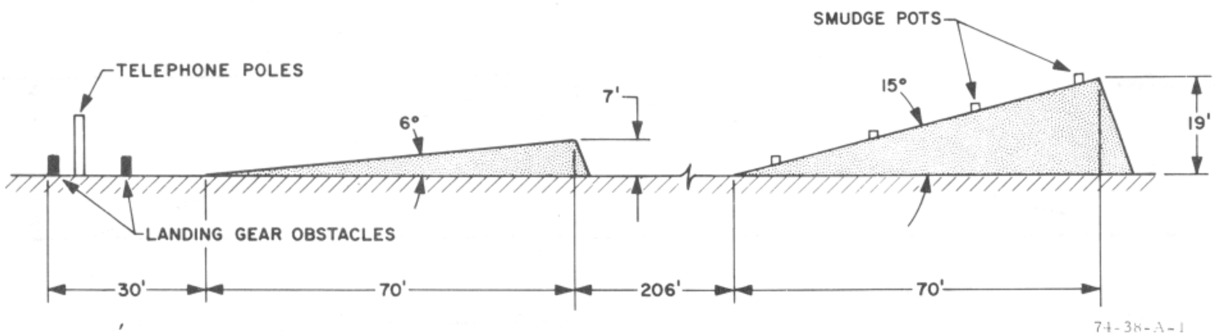


FIGURE A-1. PROPOSED TEST SITE

The first test site proposed was a 20° slope, 20 feet high, which contained 2,200 cubic yards of fill, as shown in figure A-2. After a series of conferences, it has been decided that the 20° sloped hill is too severe, especially when considered from the view that the aircraft will probably not stop on the hill, but will pass over and drop off from the 20-foot-high crest of the hill to the far side.

Based on the Armour Research Report, wherein a survivable crash is defined as one with an impact angle of not more than 15°, it was next decided to consider a crash site hill with a 15° slope on the impact side, and a gradual slope on the downtrack side, to minimize the drop off, since the aircraft would not be expected to stop on the hill. Such a hill might have the configuration of figure A-3. The amount of fill required would be 5,400 cubic yards, based on a hill width of 70 feet.

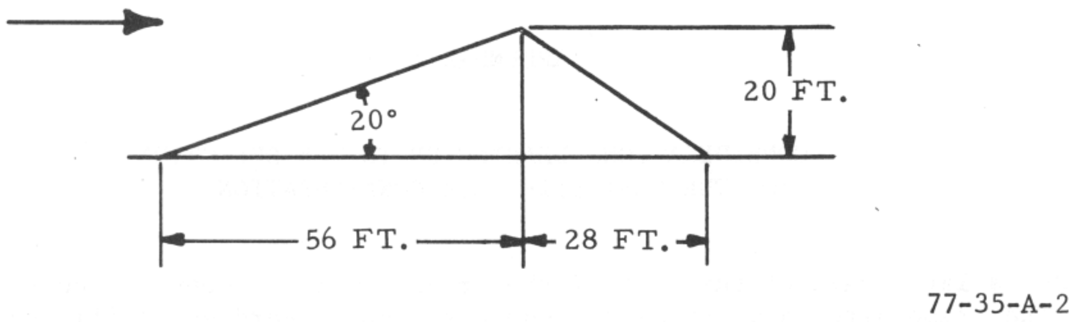


FIGURE A-2. CONSTRUCTION DETAILS OF A 20° SLOPE

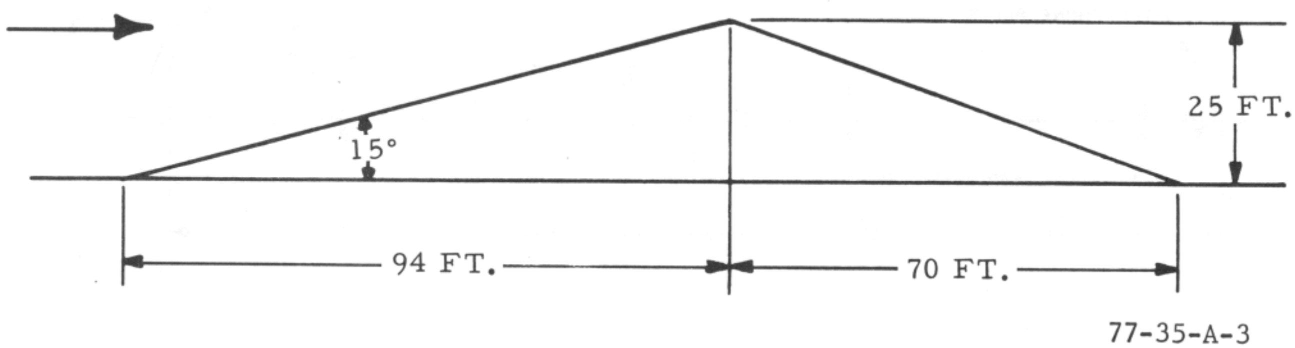


FIGURE A-3. CONSTRUCTION DETAILS OF A 15° SLOPE

Both of these hills are afflicted with the fact that the aircraft probably will not lose all of its momentum on the hill and will continue over the top scattering burning fuel and wreckage over a wide area. Such an area would be difficult to cover with the instrumentation needed to obtain meaningful results.

A look at the Phoenix crash test on the L1649 reveals some interesting facts that may be of value for this test. That crash site consisted of a 6° slope followed by a 20° slope, and the aircraft did not go over the 20° slope. The 6° slope caused the aircraft to slow down from 180 feet per second to 115 feet per second in about one aircraft length (116 feet), and the aircraft came to a complete stop on the 20° slope, which was, again, about one aircraft length long.

Therefore, it is proposed to construct a crash site along the lines of that shown in figure A-1. First, the aircraft would encounter a 6° slope, which would be one aircraft length long, and then it would encounter a 15° slope at approximately 200 feet down track, which would stop the aircraft. Such a crash site, using two hills would require 2,800 cubic yards of fill, which is a slightly greater amount than the 20° sloped hill originally proposed and only about half the fill required for the 15° sloped hill.

The aircraft will hit the 6° slope at approximately 115 knots (194 feet per second) after passing over the landing gear obstacles and the upright poles that will rupture the wing fuel tanks. Test results at Phoenix show that there is no significant reduction in speed when the aircraft encounters these two obstacles. When the aircraft leaves the 6° slope, it will have a forward velocity of 71 knots (120 feet per second), from which we can determine the upward velocity:

$$120 \text{ feet per second} \times \sin 6° = 12 \text{ feet per second}$$

The g force encountered on the 6^{o} slope will be:

$$g = \frac{1}{2} \quad \frac{V_1^2 - V_2^2}{32.2\ (S)} \quad = \frac{(194)^2 - (120)^2}{64.4\ (70)} = 5.15$$

where S = one airplane length (70 feet)

The upward velocity of 12 feet per second will cause the aircraft to rise 2.24 feet above the physical height of the slope, which is 7 feet.

$$v = at \qquad s = 1/2\ at^2$$

$$t = \frac{v}{a} \qquad = 1/2\ a\ \frac{v^2}{a^2}$$

$$t^2 = \frac{v^2}{a^2} \qquad = 1/2\ \frac{v^2}{a}$$

Substituting the above values in $S = 1/2\ \frac{v^2}{a}$

$$= 1/2\ \frac{(12)^2}{32.2}$$

$$= 2.24 \text{ feet}$$

Therefore, the total height the aircraft will attain will be 7+2.24=9.24 feet. The time it takes to attain this height after it leaves the 6° slope will be:

$$t = \frac{v}{a} = \frac{12}{32.2} = .373 \text{ s}$$

and with a forward velocity of 120 feet per second, the distance out from the end of the hill will be:

$$d = .373\ (120) = 45 \text{ feet.}$$

To drop to ground level, the time will be:

$$S = 1/2\ at^2 \qquad \text{or} \qquad t = \sqrt{\frac{2\ S}{a}}$$

$$= \sqrt{2\ \frac{(9.24)}{32.2}}$$

$$= .759 \text{ s}$$

and at a forward velocity of 120 feet per second, this would be a distance for .759 (120) = 91 feet. Therefore, the total distance the aircraft will be airborne will be 45+91 feet, not counting on any lift from the wings. There is no way of predicting just how much lift the wings will contribute, but to take into account that they will contribute some, we will add one aircraft length (70 feet) to what will be the airborne phase of the aircraft's trajectory. This will give a total distance of 136+70=206 feet. At this point we will construct the second hill, which has a 15° slope.

At this stage the aircraft will be at ground level going 71 knots (120 feet per second). If the wings contribute more lift than we assumed, the aircraft may not be at ground level, but it certainly will be descending, and will hit the 15° slope somewhere close to its base. The time required for the aircraft to get to this point will be:

a. From landing gear obstacles and poles to base of 6° slope:

$$v = 115 \text{ knots}$$
$$d = 30 \text{ feet}$$
$$t = \frac{d}{v} = \frac{30}{115 \times 1.69} = .154 \text{ s}$$

b. From end of 6° slope to base of the 15° slope:

$$\text{average } v = \frac{115 + 71}{2} = 95 \text{ knots} - 157 \text{ feet per second}$$

$$d = 70 \text{ feet}$$

$$t = \frac{d}{v} = \frac{70}{157} = .445 \text{ s}$$

c. From end of 6° slope to base of the 15° slope:

$$v = 71 \text{ knots (120 feet per second)}$$
$$d = 206 \text{ feet}$$
$$t = \frac{d}{v} = \frac{206}{120} = 1.72 \text{ s}$$

Adding these times a total time is derived of .154+.445+1.72=2.319 seconds, which is the total elapsed time from the time the aircraft encounters the landing gear obstacles and poles until it contacts the base of the 15° slope.

Assuming the aircraft stops at the top of the 15° slope, the g forces will be:

$$g = \frac{V_1^2 - V_2^2}{64.4(S)}$$

$$= \frac{(120)^2 - (0)^2}{64.4(70)}$$

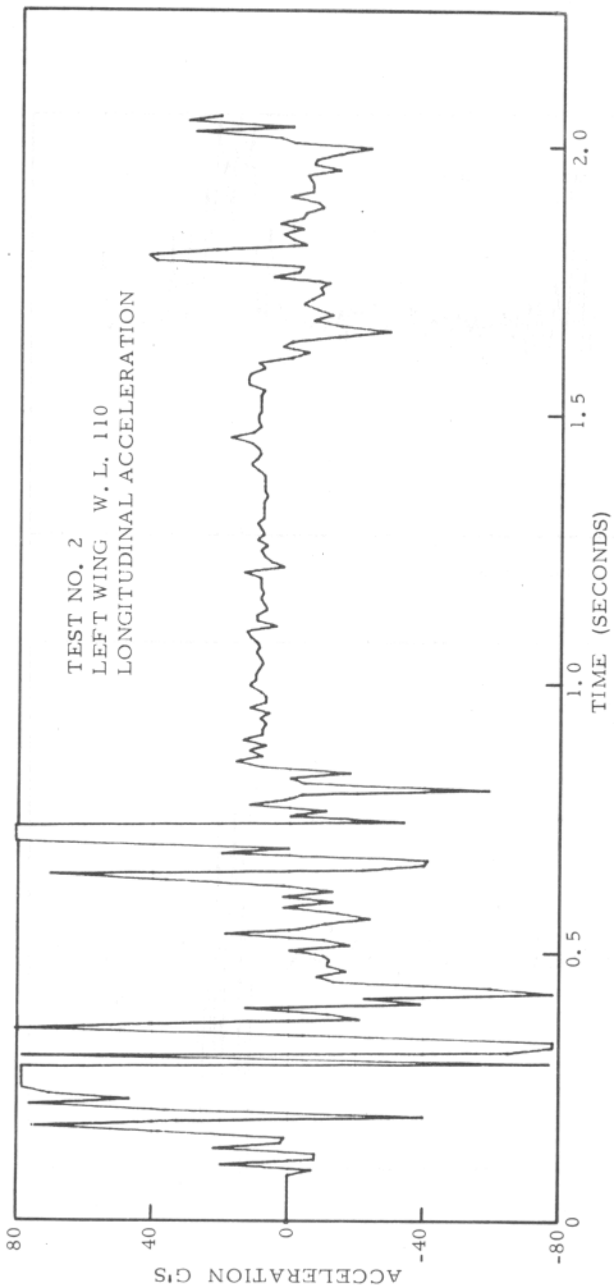
$$= 3.2$$

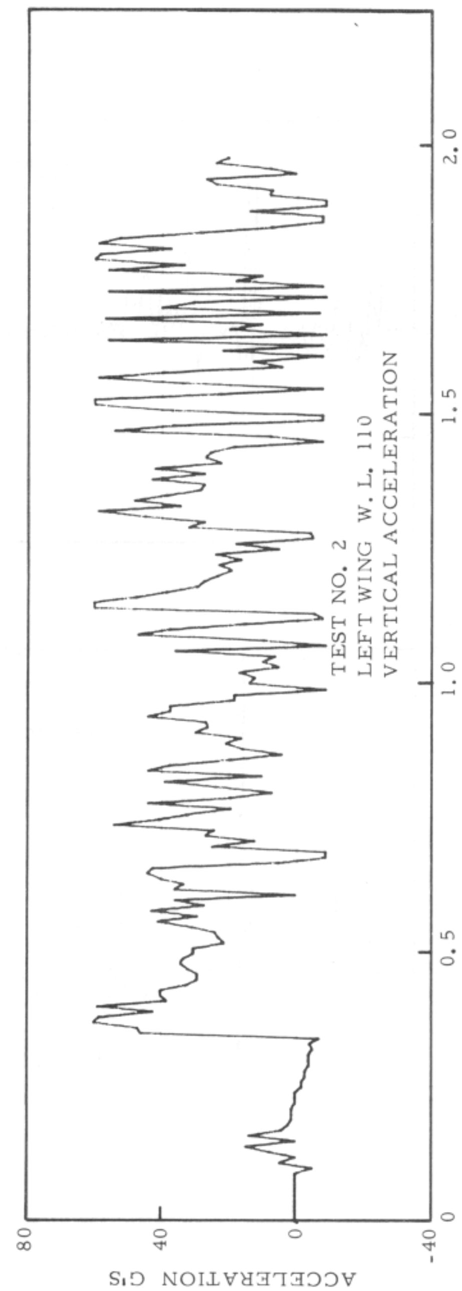
which is well within the survivable range.

There will be no artificial ignition sources, such as smudge pots, along the test site, until the base of the 15° slope. Any fuel spill or misting that occurs before the 15° slope is reached will be subjected only to hot surfaces from the idling jet engines or frictional sparks when the aircraft ceases to be airborne. However, at the 15° slope, smudge pots will be burning to insure ignition of the fuel, which should be spilling profusely and misting at this stage, so that a scientific comparison of the behavior of modified fuel over the JET A fuel may be evaluated. It is at the 15° slope that the instrumentation (radiometers, high-speed cameras) will be most heavily concentrated to assess the performance of the two fuels.

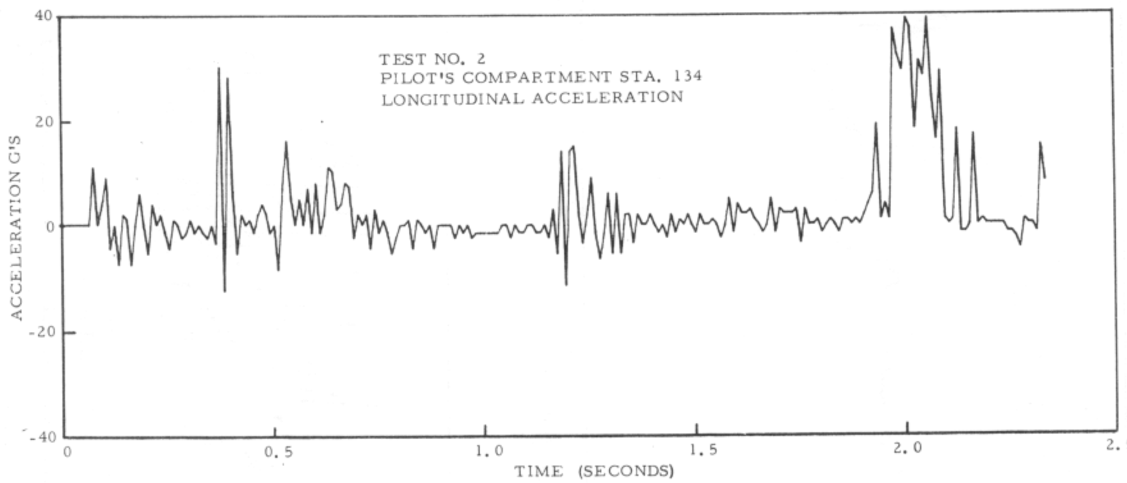
APPENDIX B

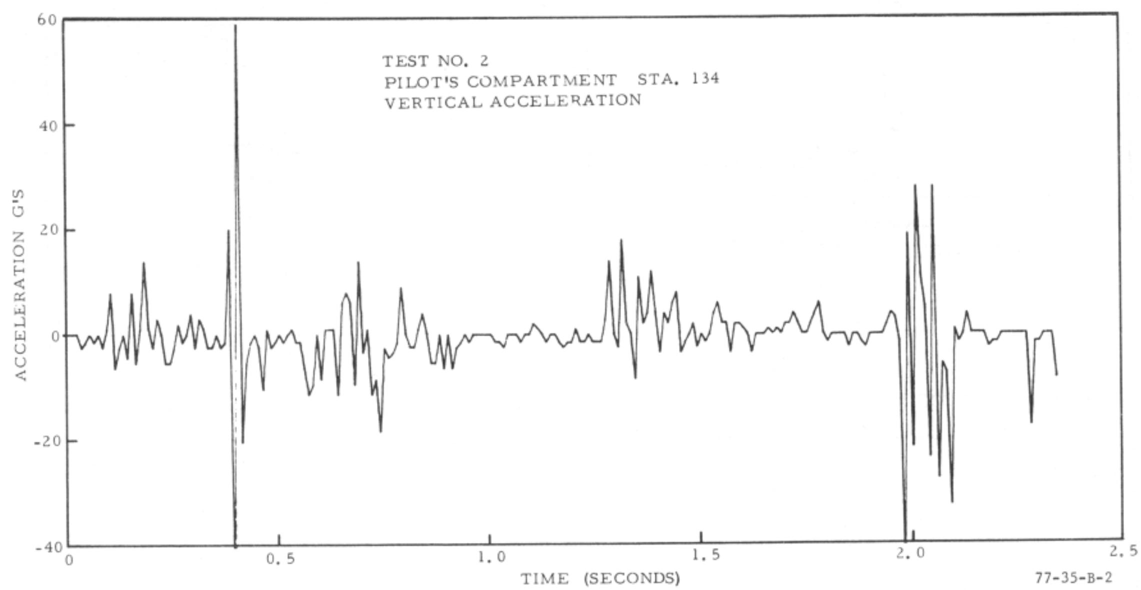
ACCELEROMETER PLOTS
DISPLACEMENT PLOTS
TEMPERATURE PLOTS


TEST NO. 2
LEFT WING W. L. 110
LONGITUDINAL ACCELERATION
ACCELERATION G'S
80
40
0
-40
-80
TIME (SECONDS)
0
0.5
1.0
1.5
2.0

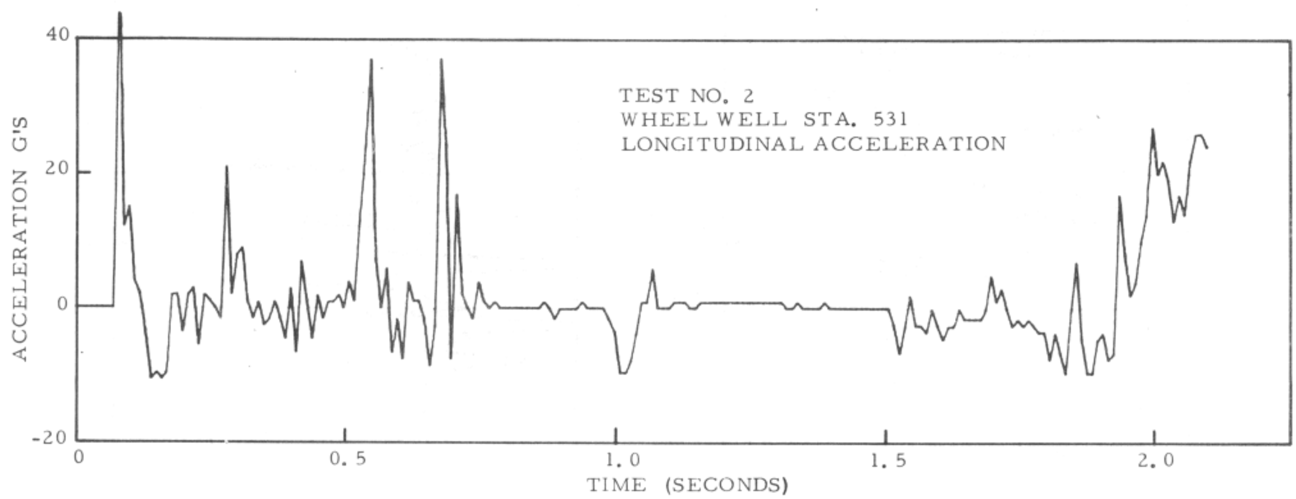

TEST NO. 2
LEFT WING W. L. 110
VERTICAL ACCELERATION
ACCELERATION G'S
80
40
0
-40
TIME (SECONDS)
0
0.5
1.0
1.5
2.0

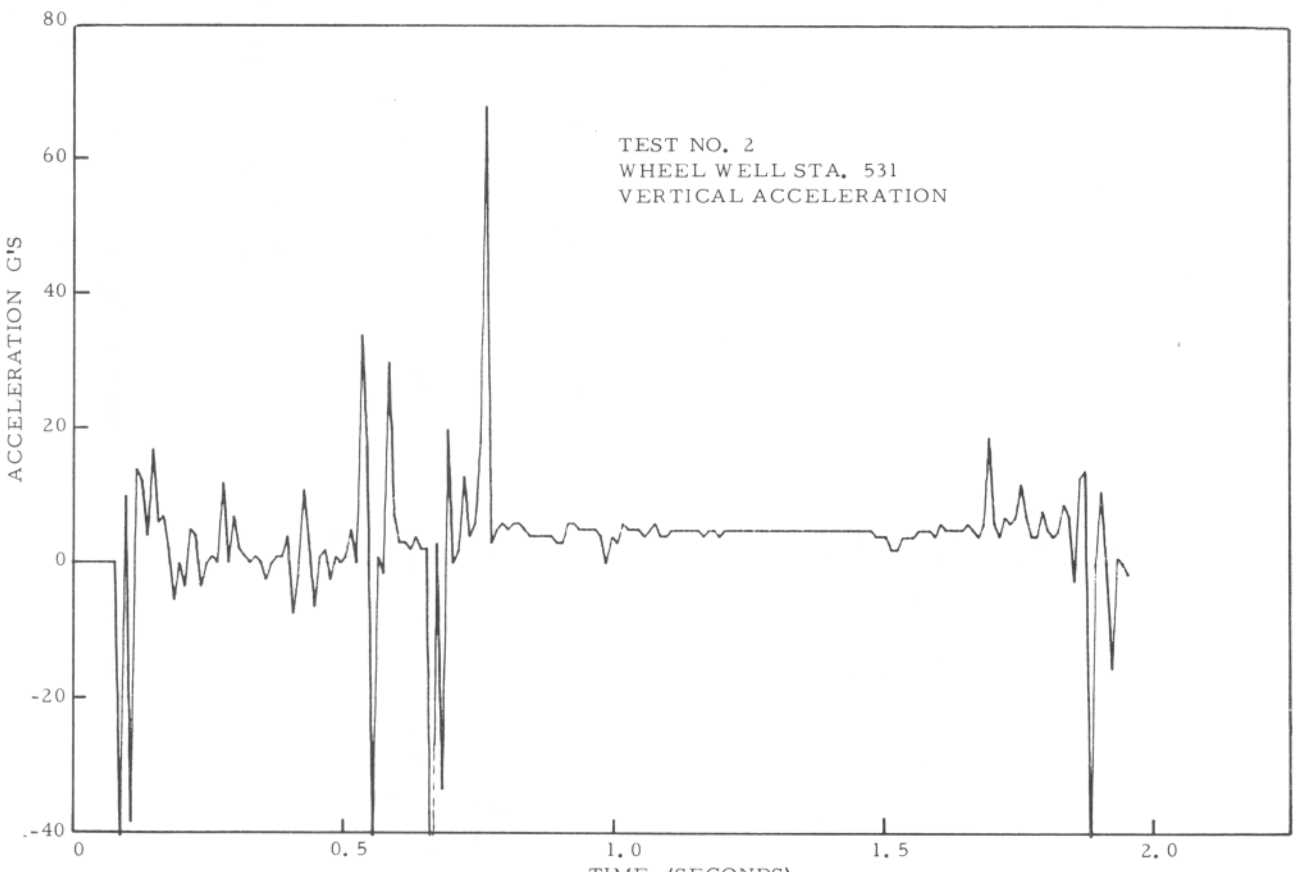
77-35-B-1


TEST NO. 2
PILOT'S COMPARTMENT STA. 134
LONGITUDINAL ACCELERATION
ACCELERATION G'S
40
20
0
-20
-40
0
0.5
1.0
1.5
2.0
2.5
TIME (SECONDS)

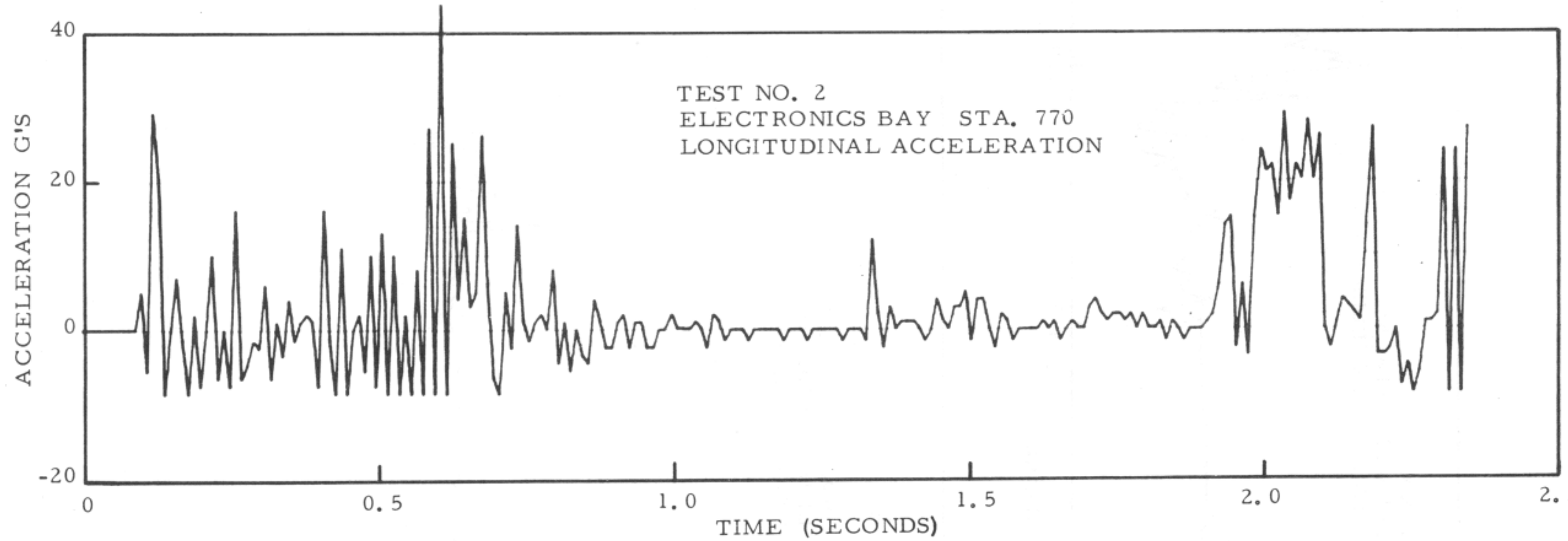

TEST NO. 2
PILOT'S COMPARTMENT STA. 134
VERTICAL ACCELERATION
ACCELERATION G'S
60
40
20
0
-20
-40
0
0.5
1.0
1.5
2.0
2.5
TIME (SECONDS)

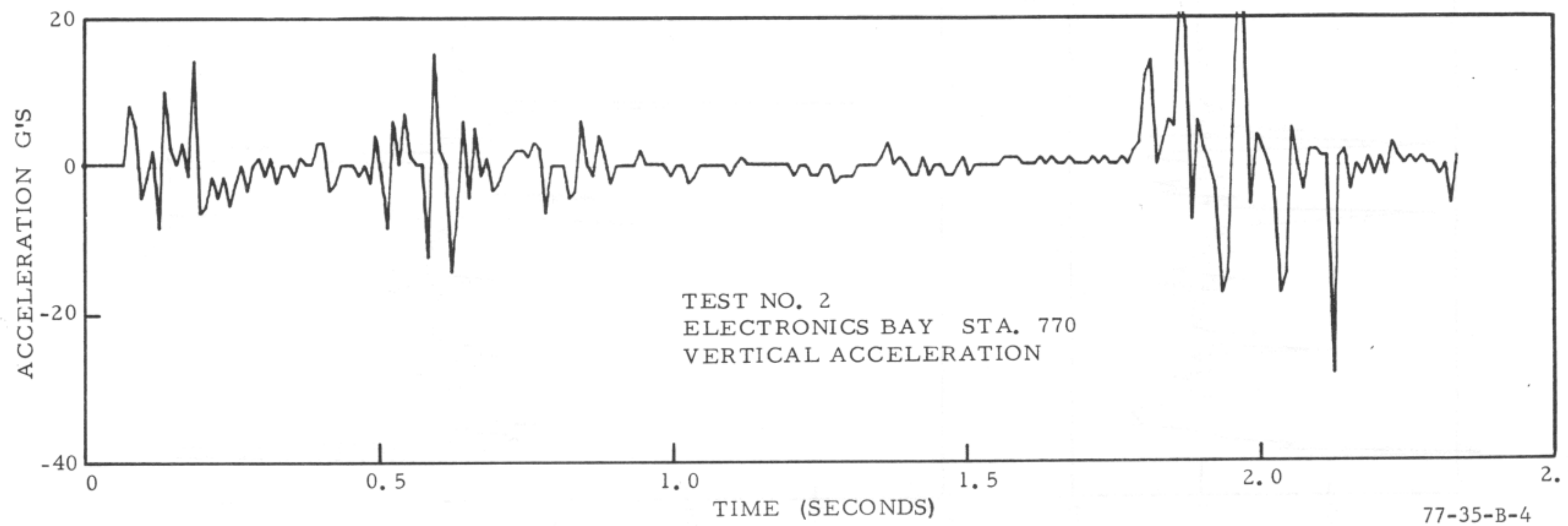
77-35-B-2


TEST NO. 2
WHEEL WELL STA. 531
LONGITUDINAL ACCELERATION
ACCELERATION G'S
40
20
0
-20
0
0.5
1.0
1.5
2.0
TIME (SECONDS)

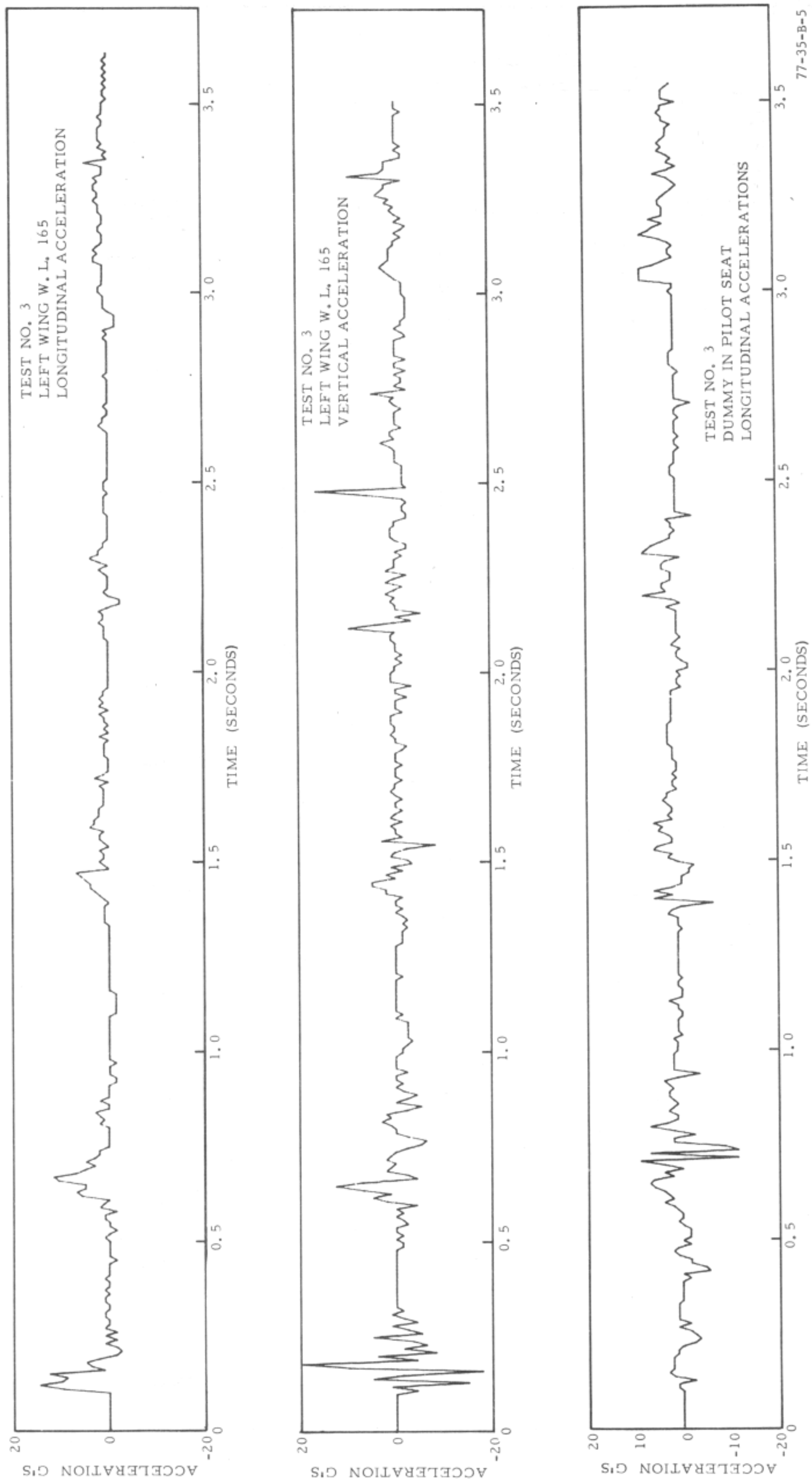

TEST NO. 2
WHEEL WELL STA. 531
VERTICAL ACCELERATION
ACCELERATION G'S
80
60
40
20
0
-20
-40
0
0.5
1.0
1.5
2.0
TIME (SECONDS)

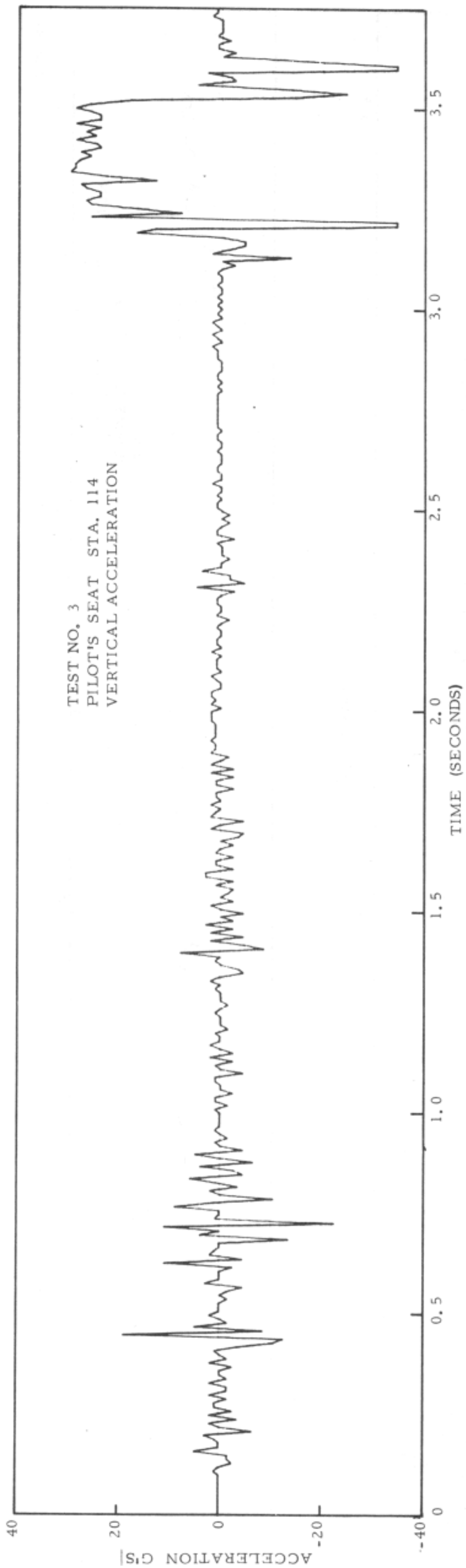
77-35-B-3

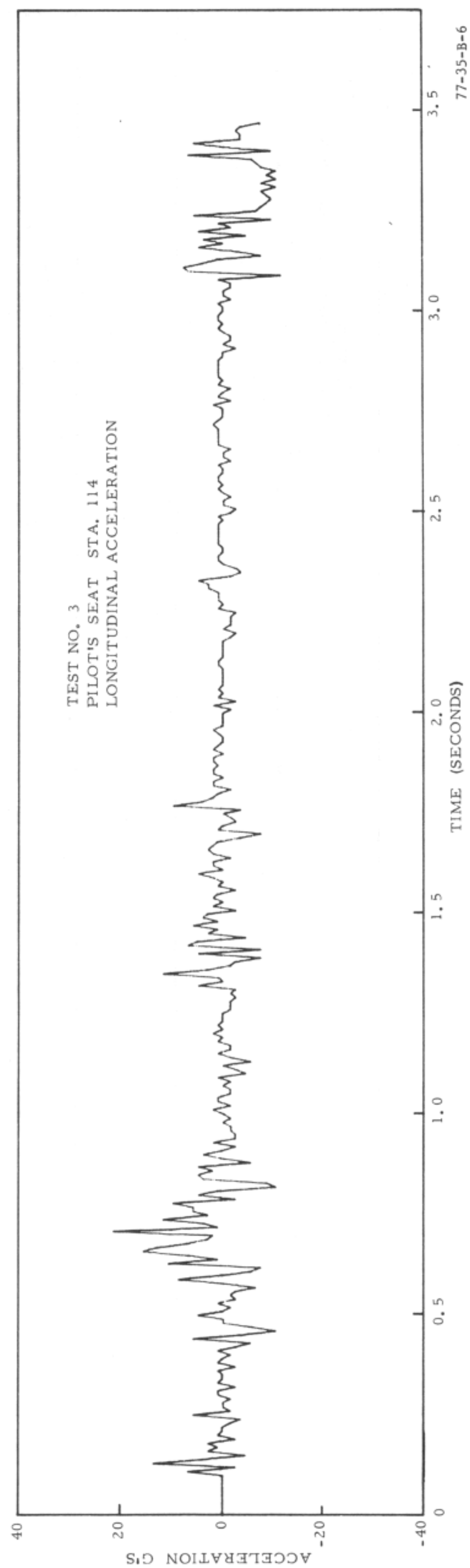

TEST NO. 2
ELECTRONICS BAY STA. 770
LONGITUDINAL ACCELERATION
ACCELERATION G'S
40
20
0
-20
0
0.5
1.0
1.5
2.0
2.5
TIME (SECONDS)

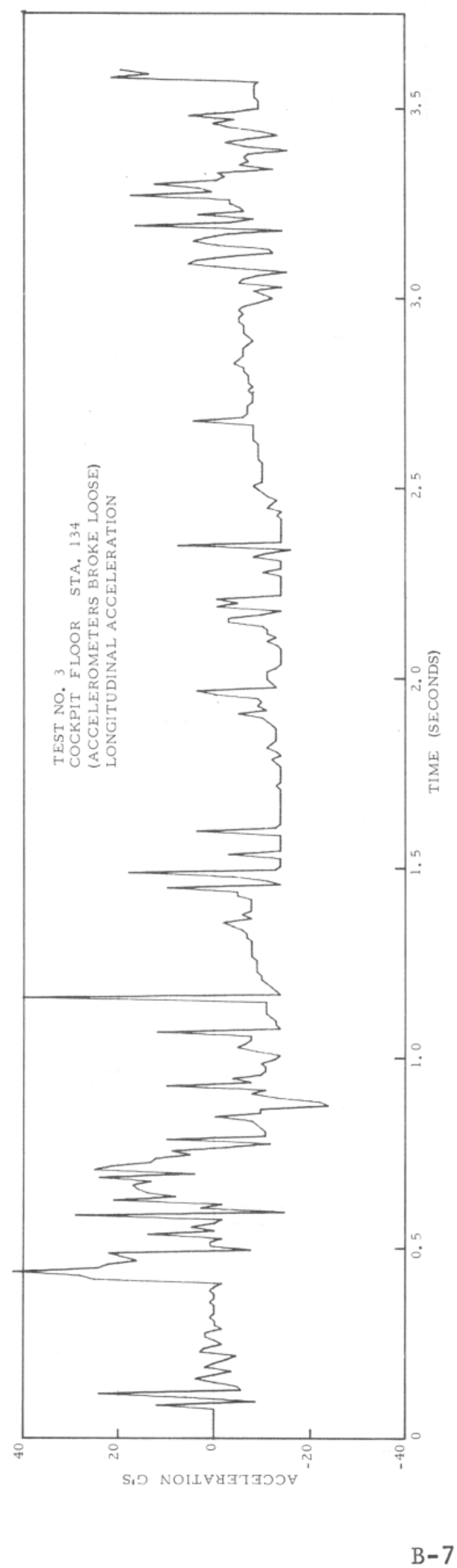

TEST NO. 2
ELECTRONICS BAY STA. 770
VERTICAL ACCELERATION
ACCELERATION G'S
20
0
-20
-40
0
0.5
1.0
1.5
2.0
2.5
TIME (SECONDS)

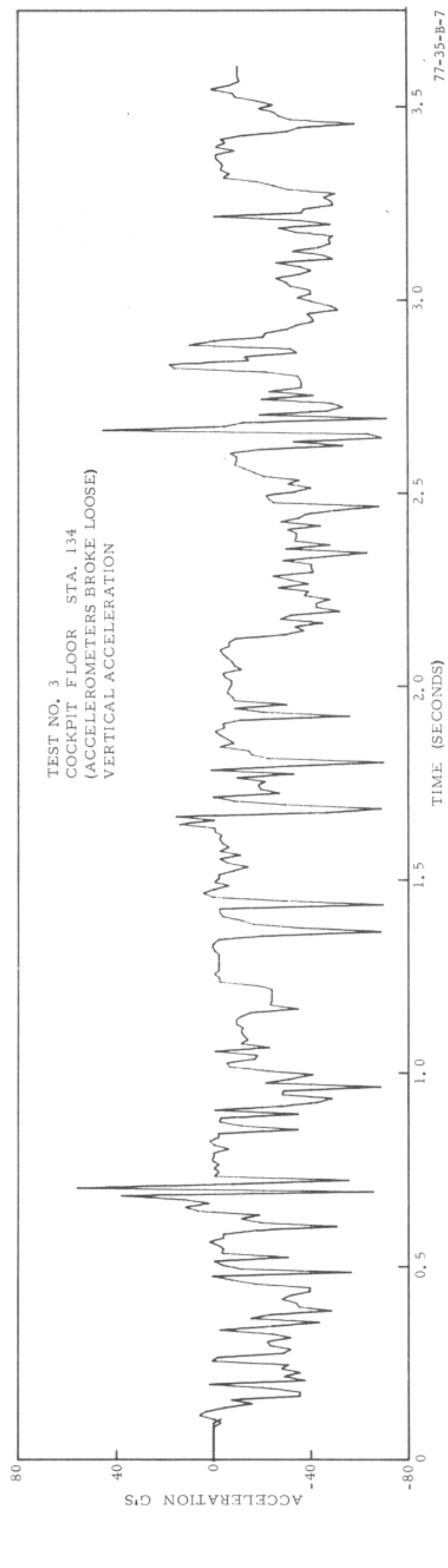
77-35-B-4


TEST NO. 3
LEFT WING W.L. 165
LONGITUDINAL ACCELERATION
TEST NO. 3
LEFT WING W.L. 165
VERTICAL ACCELERATION
TEST NO. 3
DUMMY IN PILOT SEAT
LONGITUDINAL ACCELERATIONS
ACCELERATION G's
TIME (SECONDS)
20
10
0
-10
-20
0.5
1.0
1.5
2.0
2.5
3.0
3.5
77-35-B-5

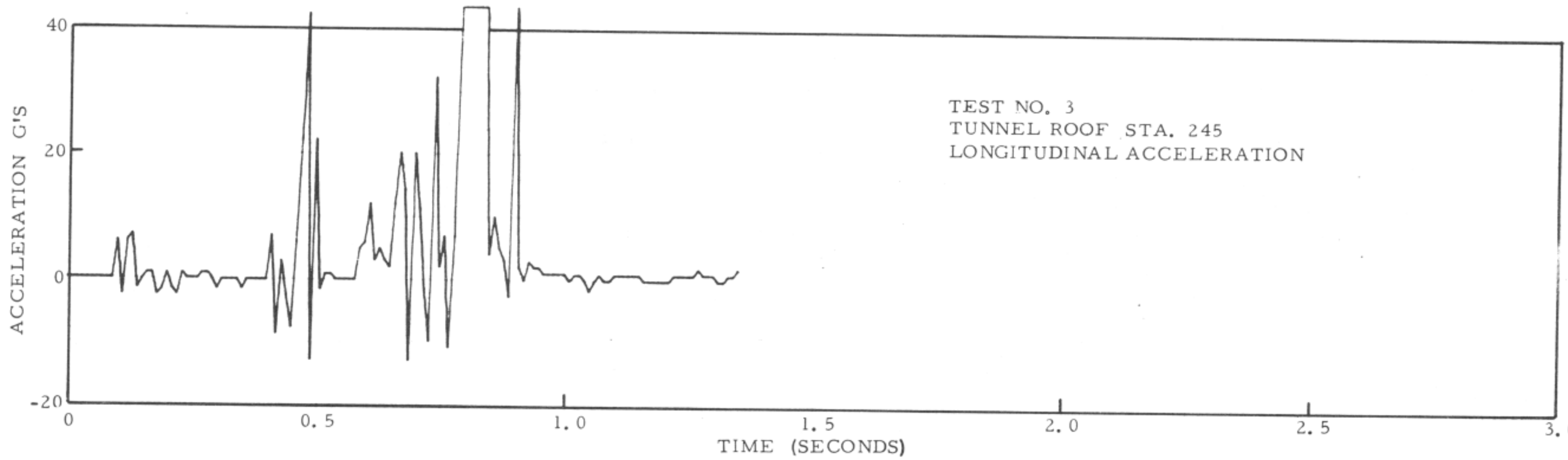

TEST NO. 3
PILOT'S SEAT STA. 114
VERTICAL ACCELERATION
ACCELERATION G'S
TIME (SECONDS)
40
20
0
-20
-40
0
0.5
1.0
1.5
2.0
2.5
3.0
3.5

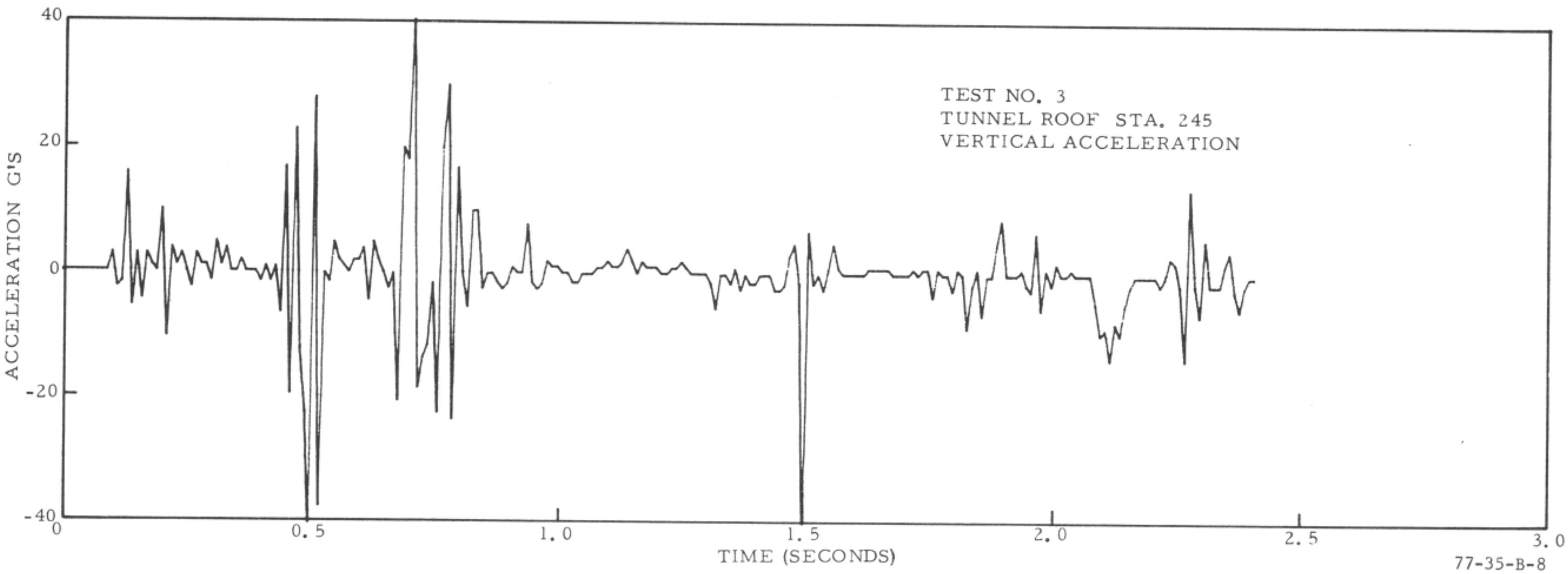

TEST NO. 3
PILOT'S SEAT STA. 114
LONGITUDINAL ACCELERATION
ACCELERATION G'S
TIME (SECONDS)
40
20
0
-20
-40
0
0.5
1.0
1.5
2.0
2.5
3.0
3.5
77-35-B-6


TEST NO. 3
COCKPIT FLOOR STA. 134
(ACCELEROMETERS BROKE LOOSE)
LONGITUDINAL ACCELERATION
ACCELERATION G'S
40
20
0
-20
-40
0
0.5
1.0
1.5
2.0
2.5
3.0
3.5
TIME (SECONDS)

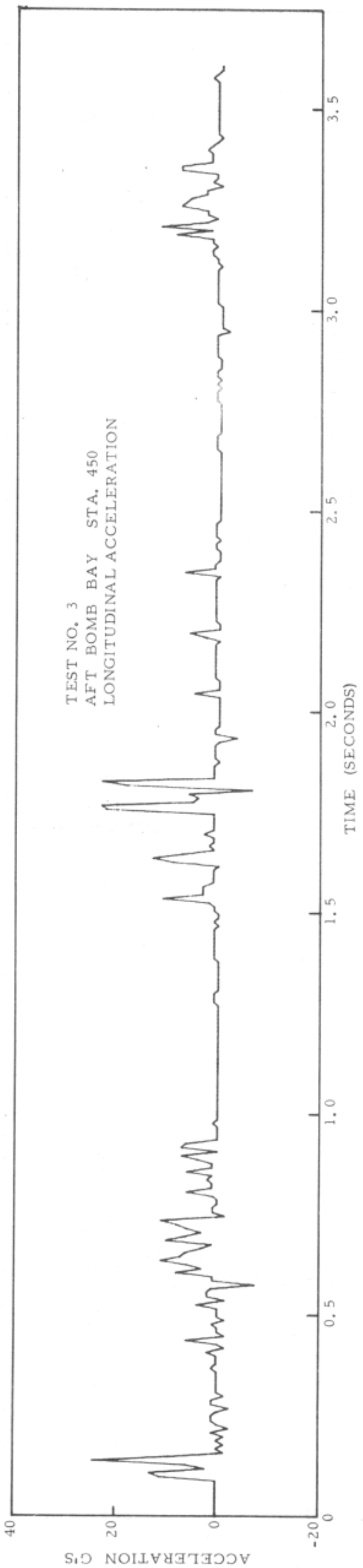

TEST NO. 3
COCKPIT FLOOR STA. 134
(ACCELEROMETERS BROKE LOOSE)
VERTICAL ACCELERATION
ACCELERATION G'S
80
40
0
-40
-80
0
0.5
1.0
1.5
2.0
2.5
3.0
3.5
TIME (SECONDS)

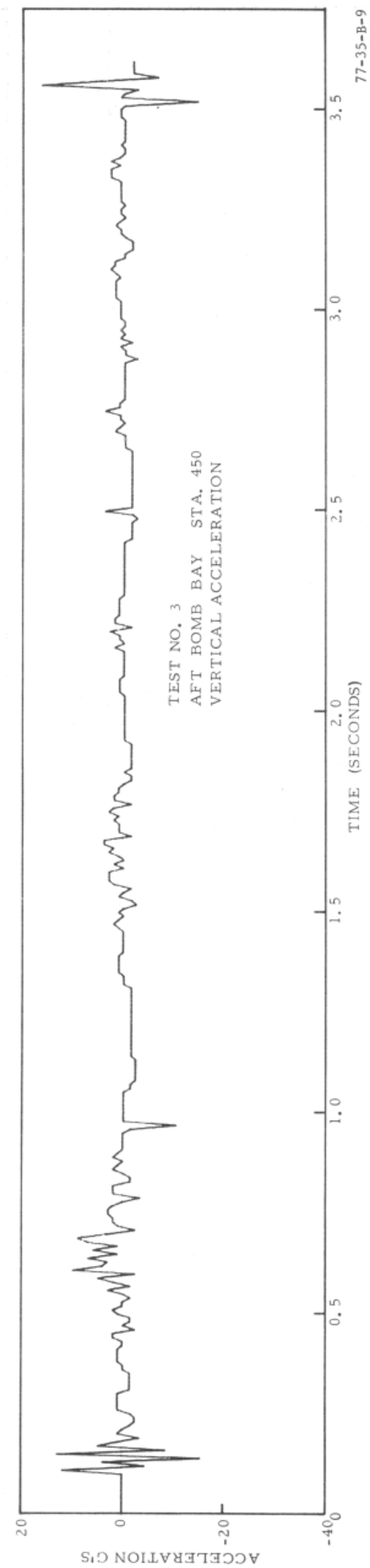
77-35-B-7


TEST NO. 3
TUNNEL ROOF STA. 245
LONGITUDINAL ACCELERATION
ACCELERATION G'S
40
20
0
-20
0
0.5
1.0
1.5
2.0
2.5
3.0
TIME (SECONDS)

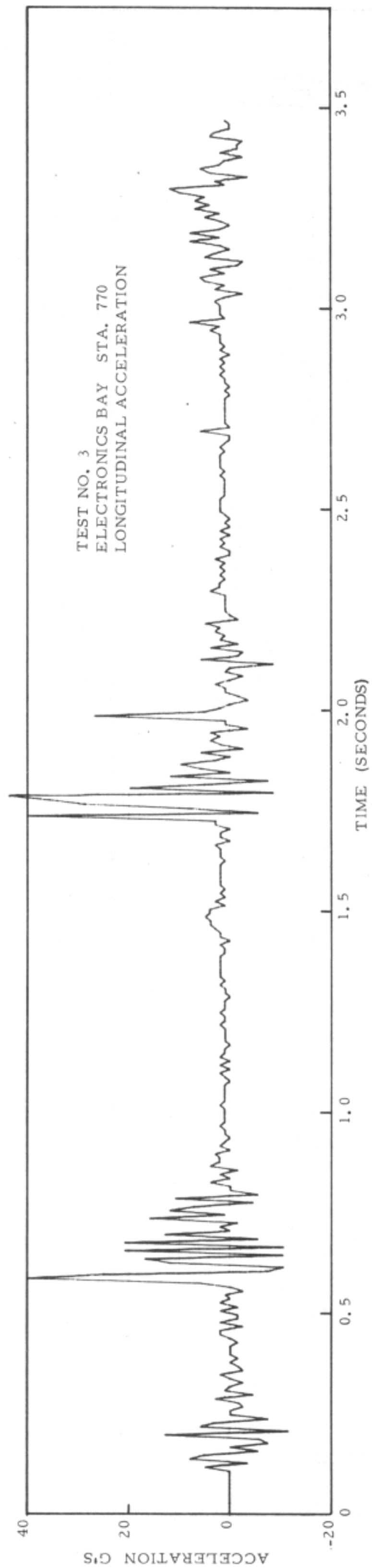

TEST NO. 3
TUNNEL ROOF STA. 245
VERTICAL ACCELERATION
ACCELERATION G'S
40
20
0
-20
-40
0
0.5
1.0
1.5
2.0
2.5
3.0
TIME (SECONDS)

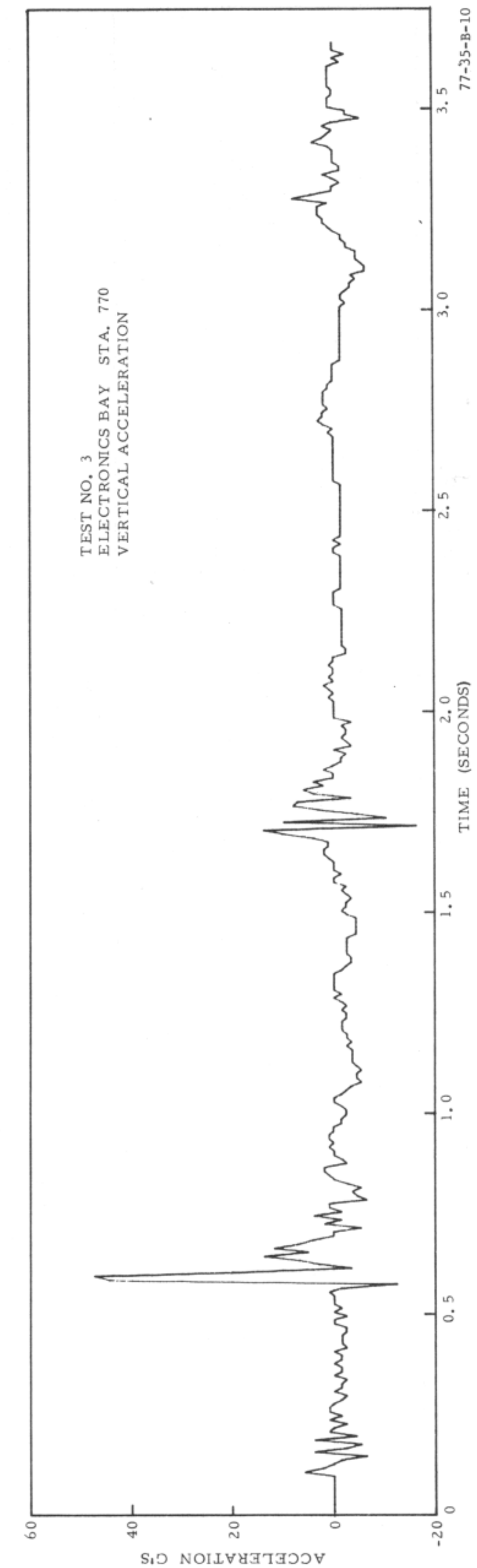
77-35-B-8

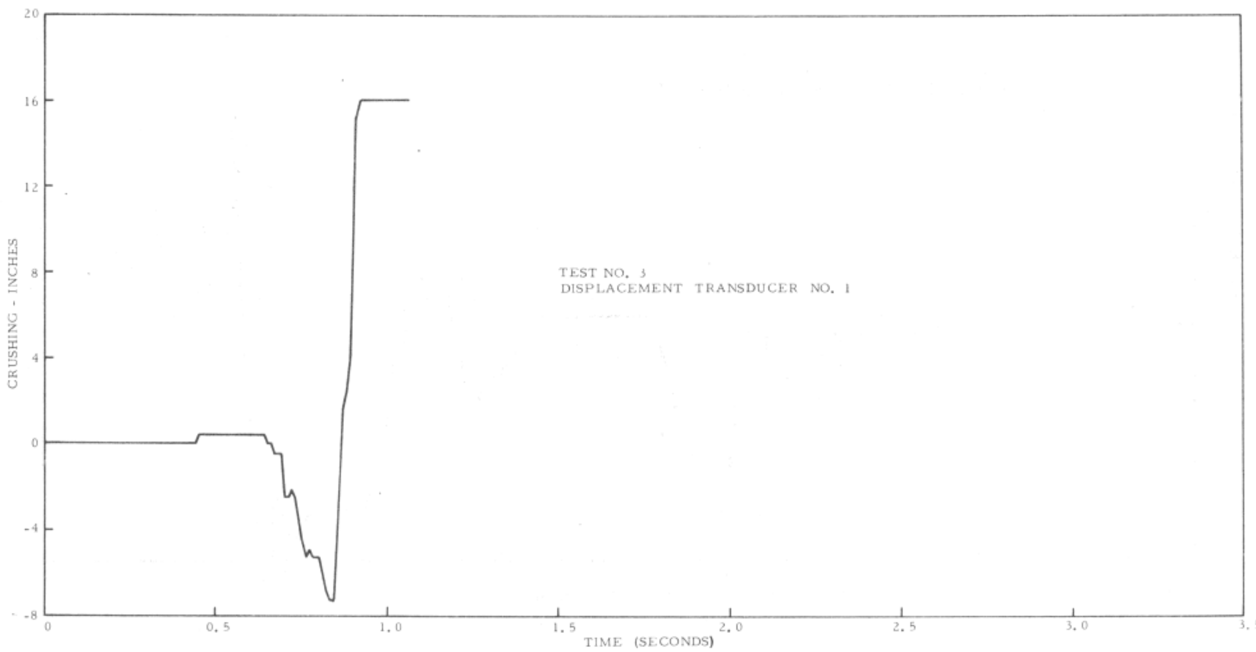

TEST NO. 3
AFT BOMB BAY STA. 450
LONGITUDINAL ACCELERATION
ACCELERATION G'S
40
20
0
-20
0
0.5
1.0
1.5
2.0
2.5
3.0
3.5
TIME (SECONDS)


TEST NO. 3
AFT BOMB BAY STA. 450
VERTICAL ACCELERATION
ACCELERATION G'S
20
0
-20
-40
0
0.5
1.0
1.5
2.0
2.5
3.0
3.5
TIME (SECONDS)

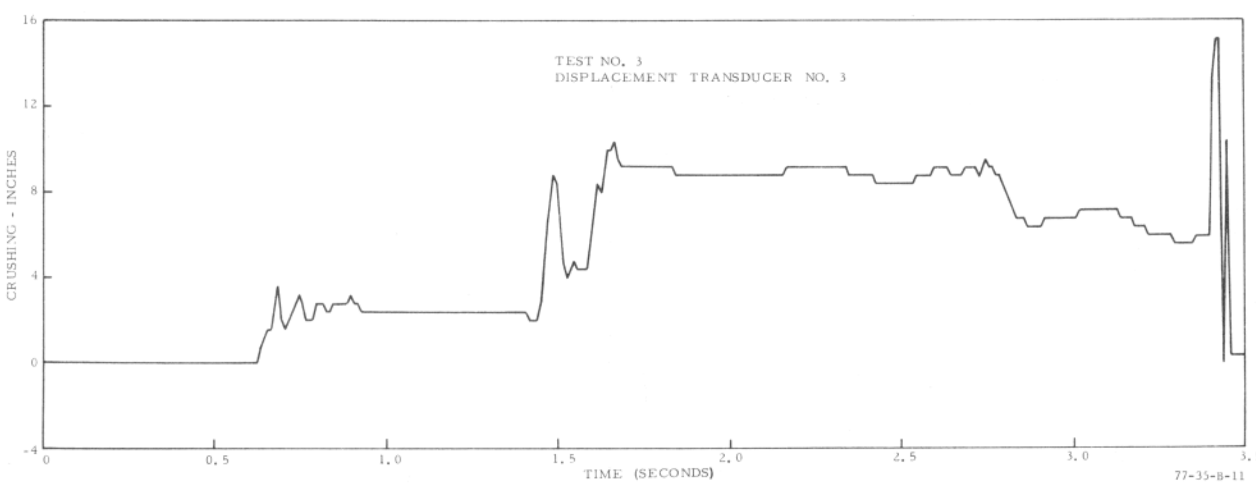
77-35-B-9


TEST NO. 3
ELECTRONICS BAY STA. 770
LONGITUDINAL ACCELERATION
ACCELERATION G'S
40
20
0
-20
0
0.5
1.0
1.5
2.0
2.5
3.0
3.5
TIME (SECONDS)

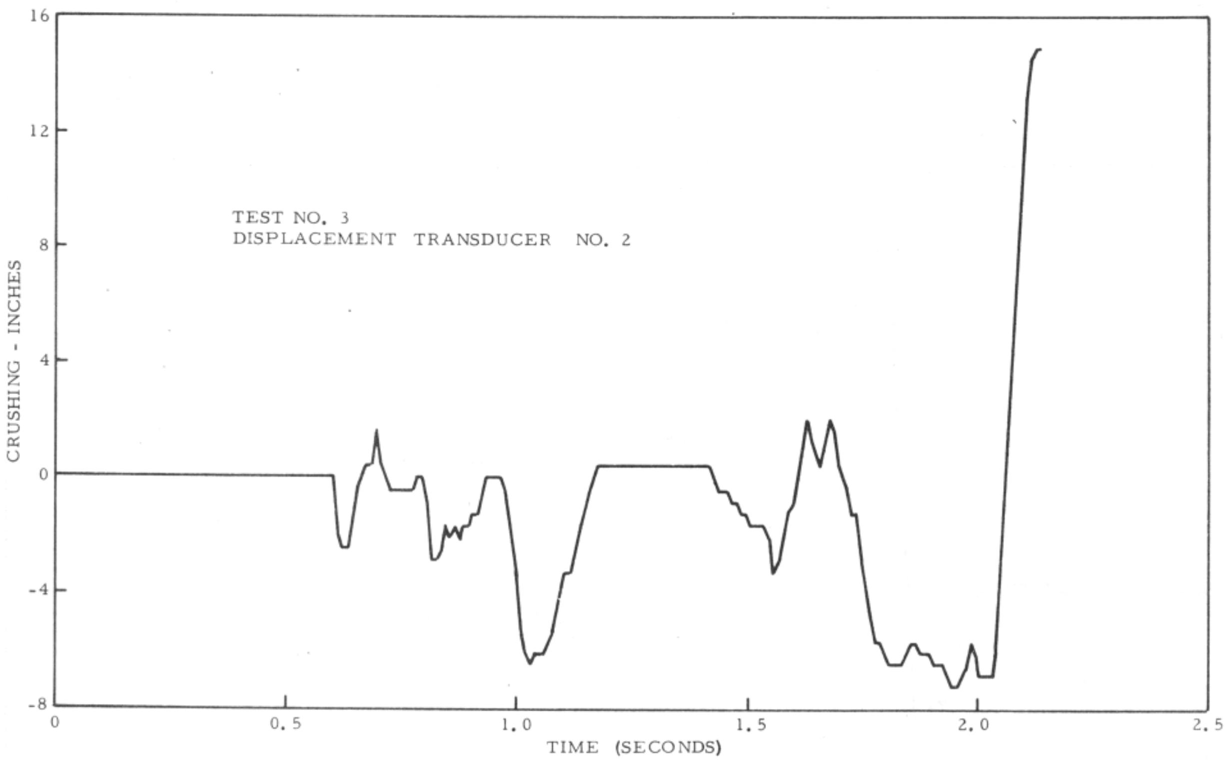

TEST NO. 3
ELECTRONICS BAY STA. 770
VERTICAL ACCELERATION
ACCELERATION G'S
60
40
20
0
-20
0
0.5
1.0
1.5
2.0
2.5
3.0
3.5
TIME (SECONDS)
77-35-B-10

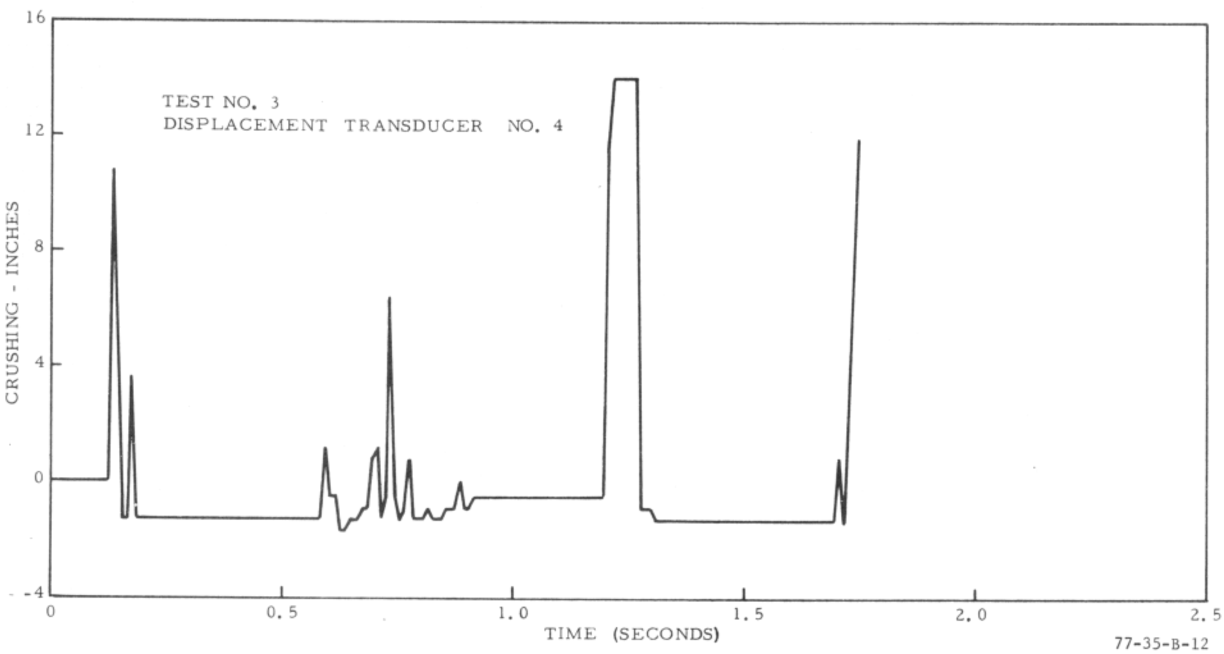


20
16
12
8
4
0
-4
-8
CRUSHING - INCHES
TEST NO. 3
DISPLACEMENT TRANSDUCER NO. 1
0
0.5
1.0
1.5
2.0
2.5
3.0
3.5
TIME (SECONDS)

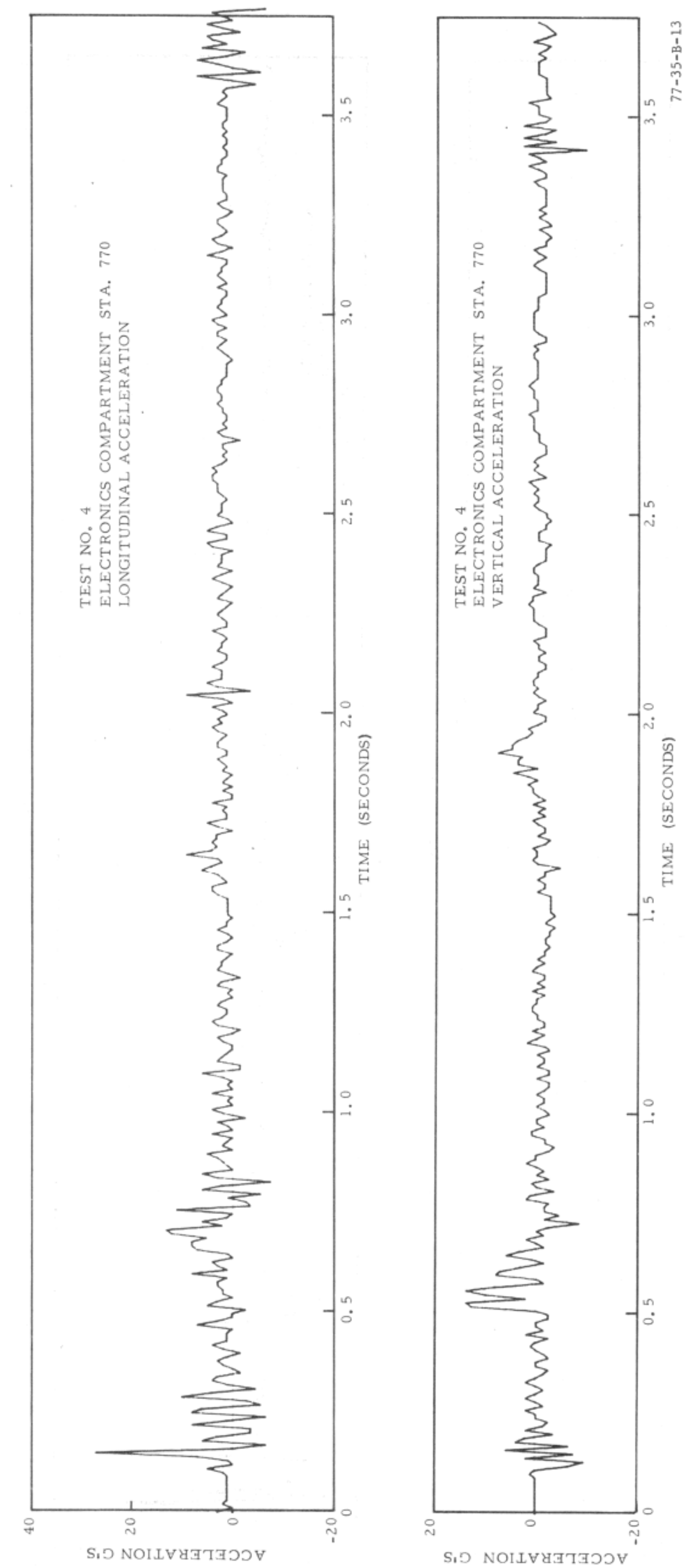


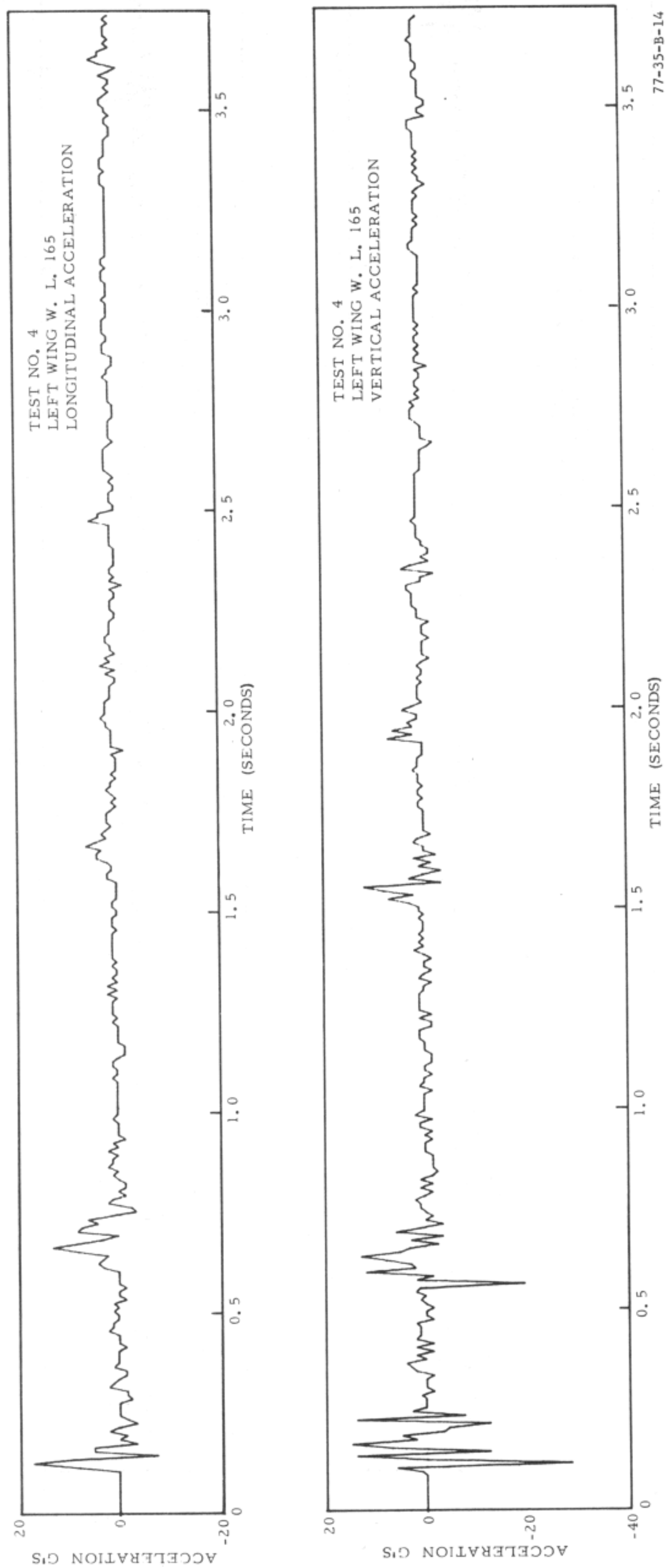
16
12
8
4
0
-4
CRUSHING - INCHES
TEST NO. 3
DISPLACEMENT TRANSDUCER NO. 3
0
0.5
1.0
1.5
2.0
2.5
3.0
3.5
TIME (SECONDS)
77-35-B-11


16
12
8
4
0
-4
-8
CRUSHING - INCHES
TEST NO. 3
DISPLACEMENT TRANSDUCER NO. 2
0
0.5
1.0
1.5
2.0
2.5
TIME (SECONDS)

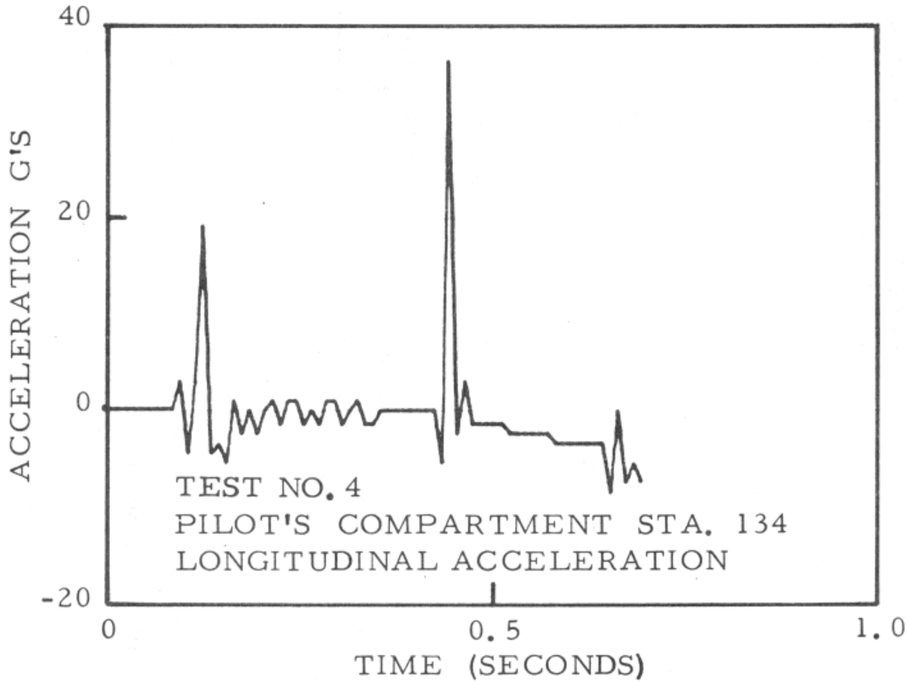

16
12
8
4
0
-4
CRUSHING - INCHES
TEST NO. 3
DISPLACEMENT TRANSDUCER NO. 4
0
0.5
1.0
1.5
2.0
2.5
TIME (SECONDS)

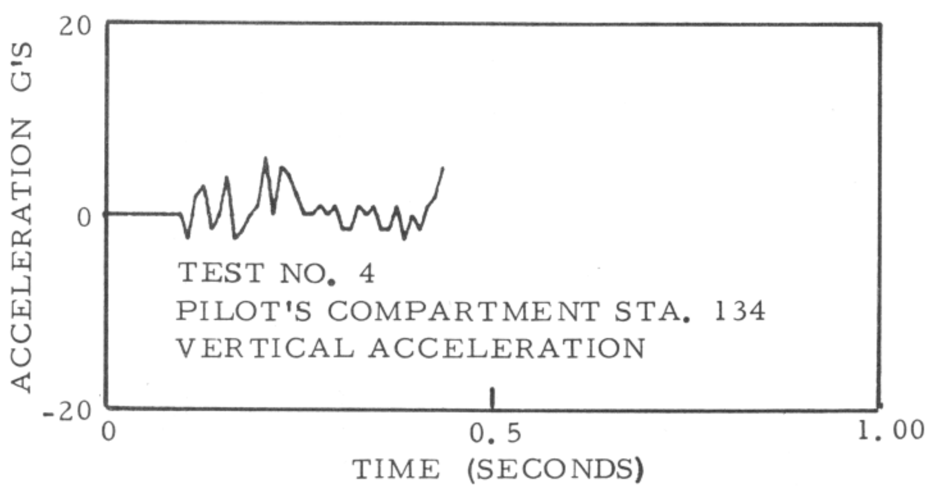
77-35-B-12


TEST NO. 4
ELECTRONICS COMPARTMENT STA. 770
LONGITUDINAL ACCELERATION
TIME (SECONDS)
ACCELERATION G'S
TEST NO. 4
ELECTRONICS COMPARTMENT STA. 770
VERTICAL ACCELERATION
TIME (SECONDS)
ACCELERATION G'S
77-35-B-13

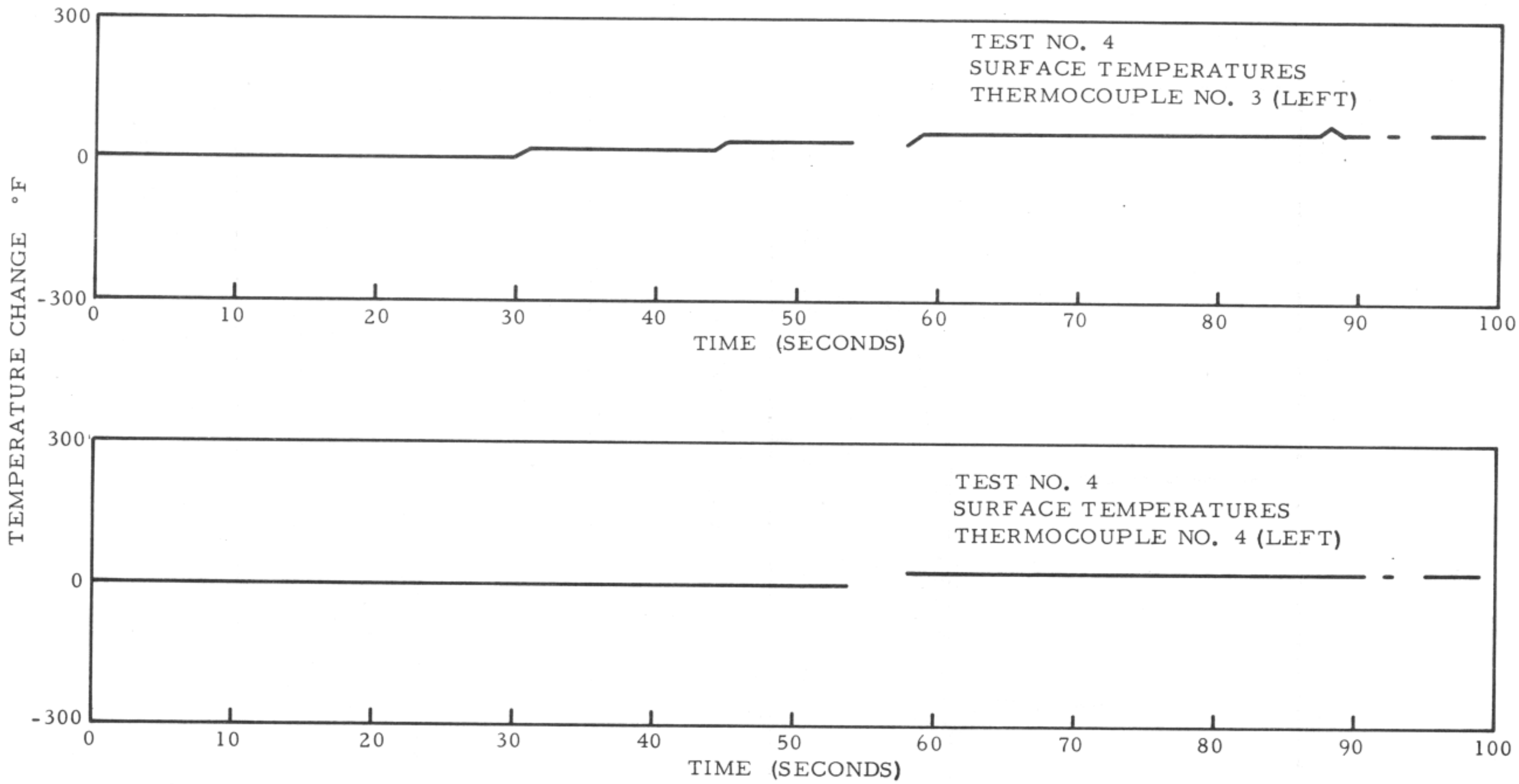

TEST NO. 4
LEFT WING W. L. 165
LONGITUDINAL ACCELERATION
ACCELERATION G'S
20
0
-20
TEST NO. 4
LEFT WING W. L. 165
VERTICAL ACCELERATION
ACCELERATION G'S
20
0
-20
-40
0
0.5
1.0
1.5
2.0
2.5
3.0
3.5
TIME (SECONDS)

77-35-B-14

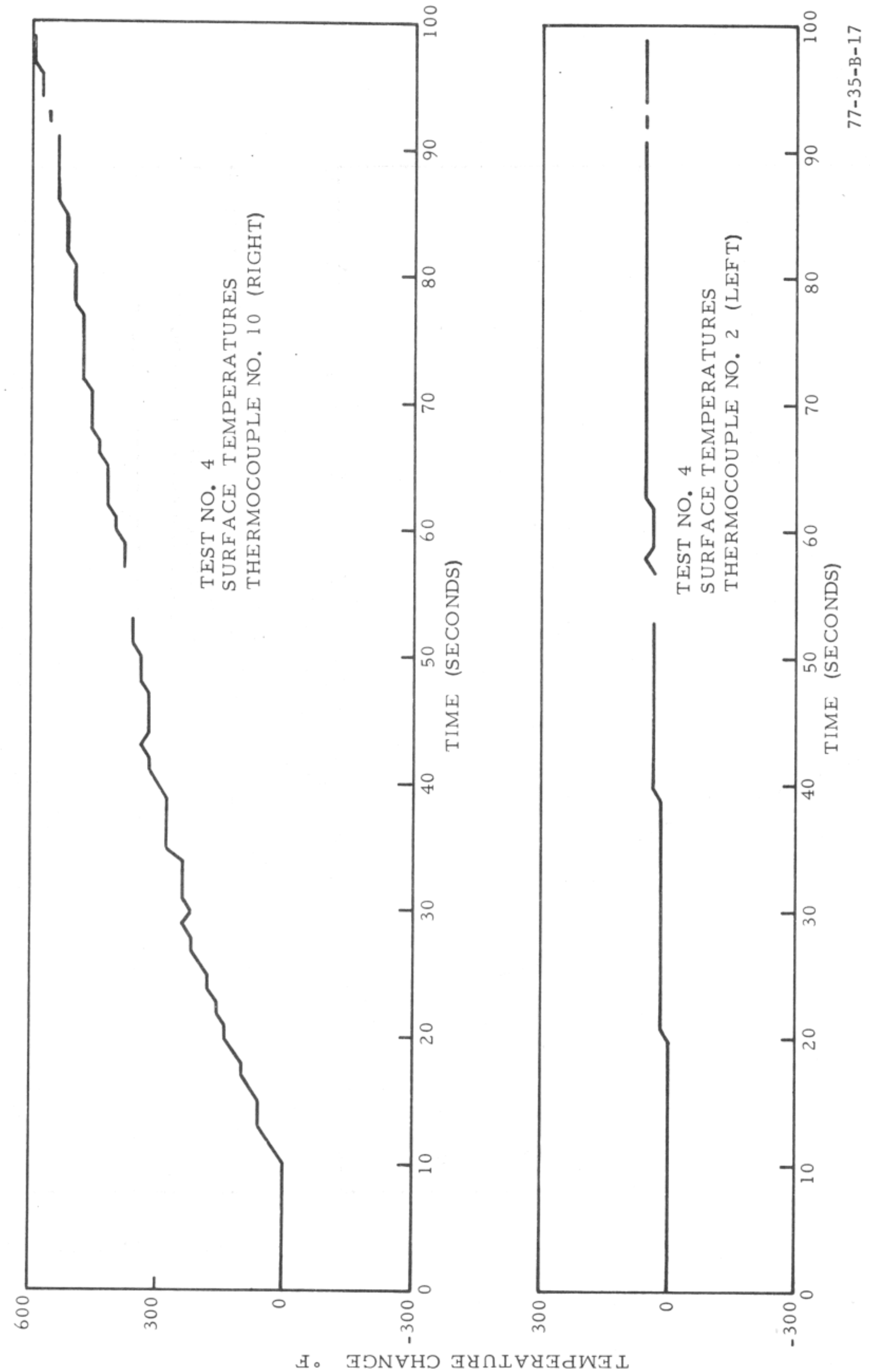

40
20
0
-20
ACCELERATION G'S
TEST NO. 4
PILOT'S COMPARTMENT STA. 134
LONGITUDINAL ACCELERATION
0
0.5
1.0
TIME (SECONDS)

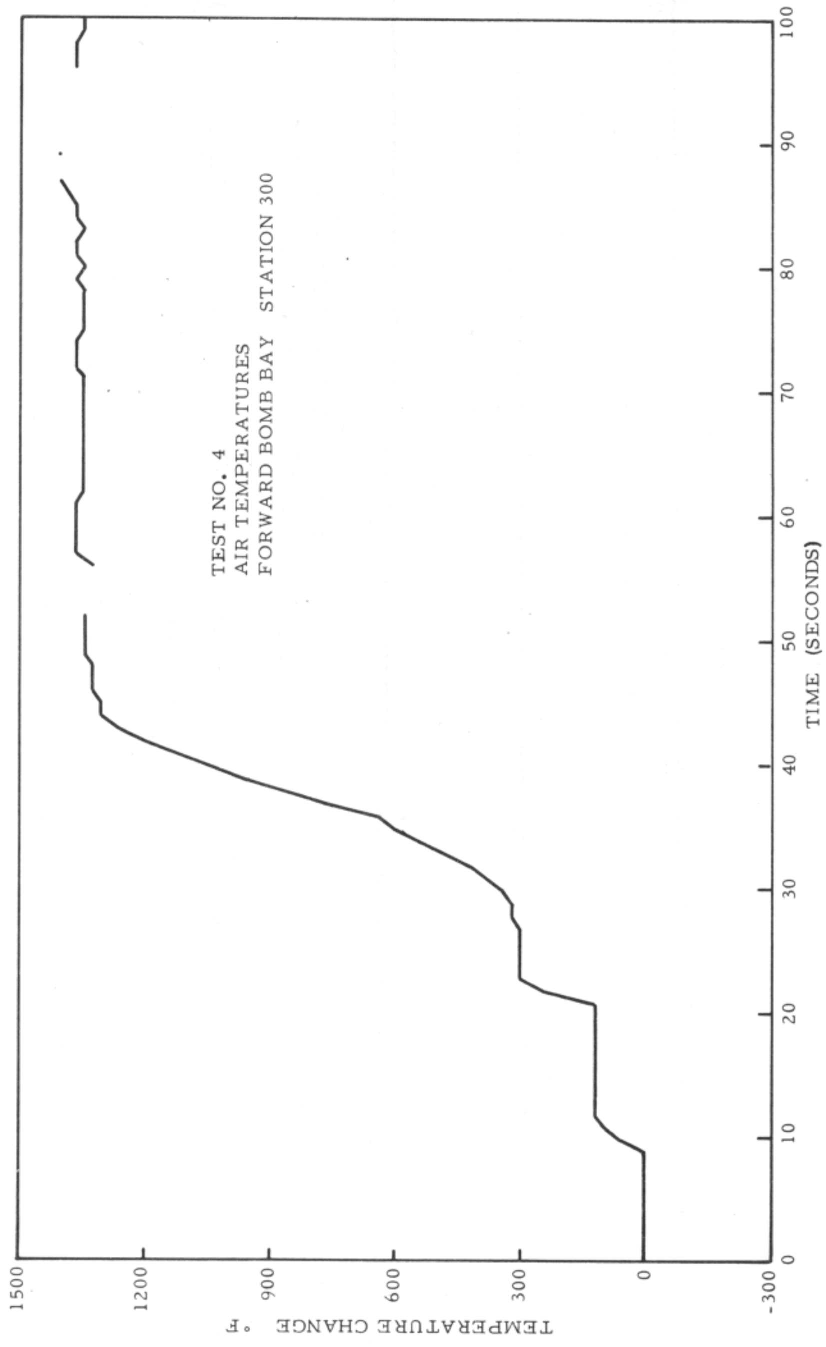

20
0
-20
ACCELERATION G'S
TEST NO. 4
PILOT'S COMPARTMENT STA. 134
VERTICAL ACCELERATION
0
0.5
1.00
TIME (SECONDS)

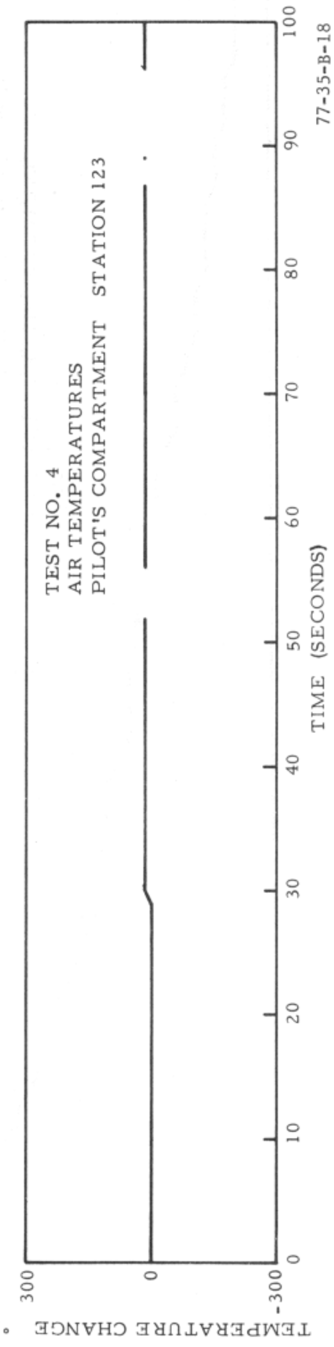
77-35-B-15

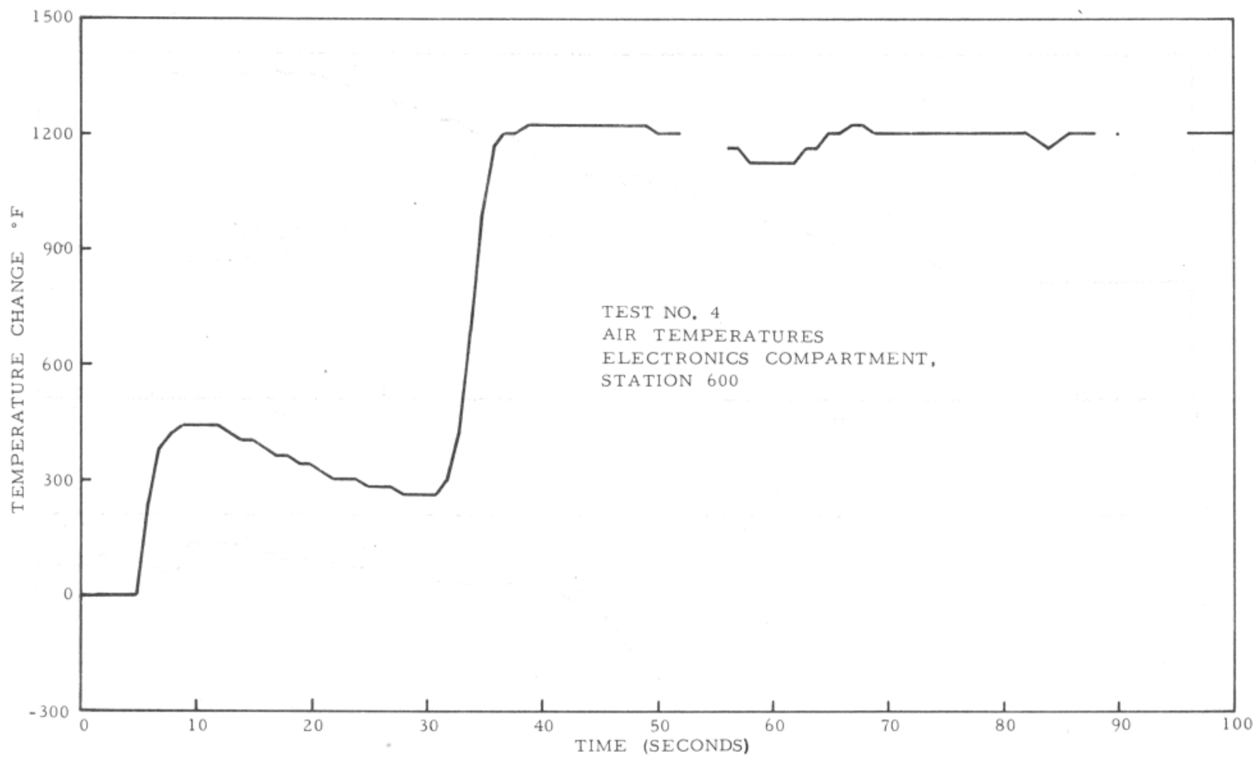

TEST NO. 4
SURFACE TEMPERATURES
THERMOCOUPLE NO. 3 (LEFT)
300
0
-300
0
10
20
30
40
50
60
70
80
90
100
TIME (SECONDS)
TEST NO. 4
SURFACE TEMPERATURES
THERMOCOUPLE NO. 4 (LEFT)
300
0
-300
0
10
20
30
40
50
60
70
80
90
100
TIME (SECONDS)
TEMPERATURE CHANGE °F

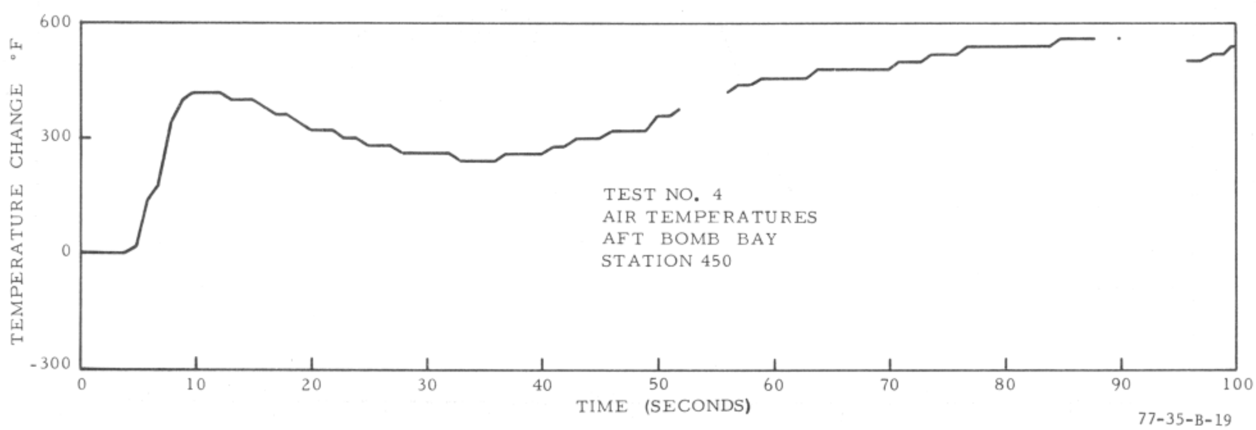
77-35-B-16


TEST NO. 4
SURFACE TEMPERATURES
THERMOCOUPLE NO. 10 (RIGHT)
TEST NO. 4
SURFACE TEMPERATURES
THERMOCOUPLE NO. 2 (LEFT)
TIME (SECONDS)
TEMPERATURE CHANGE °F
600
300
0
-300
0
10
20
30
40
50
60
70
80
90
100
77-35-B-17

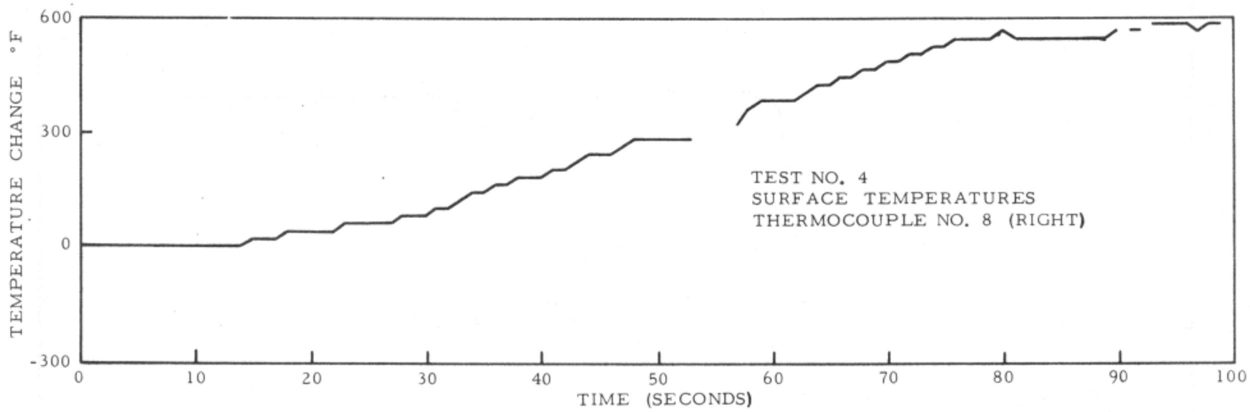

TEST NO. 4
AIR TEMPERATURES
FORWARD BOMB BAY STATION 300
TEMPERATURE CHANGE °F
1500
1200
900
600
300
0
-300
0
10
20
30
40
50
60
70
80
90
100
TIME (SECONDS)

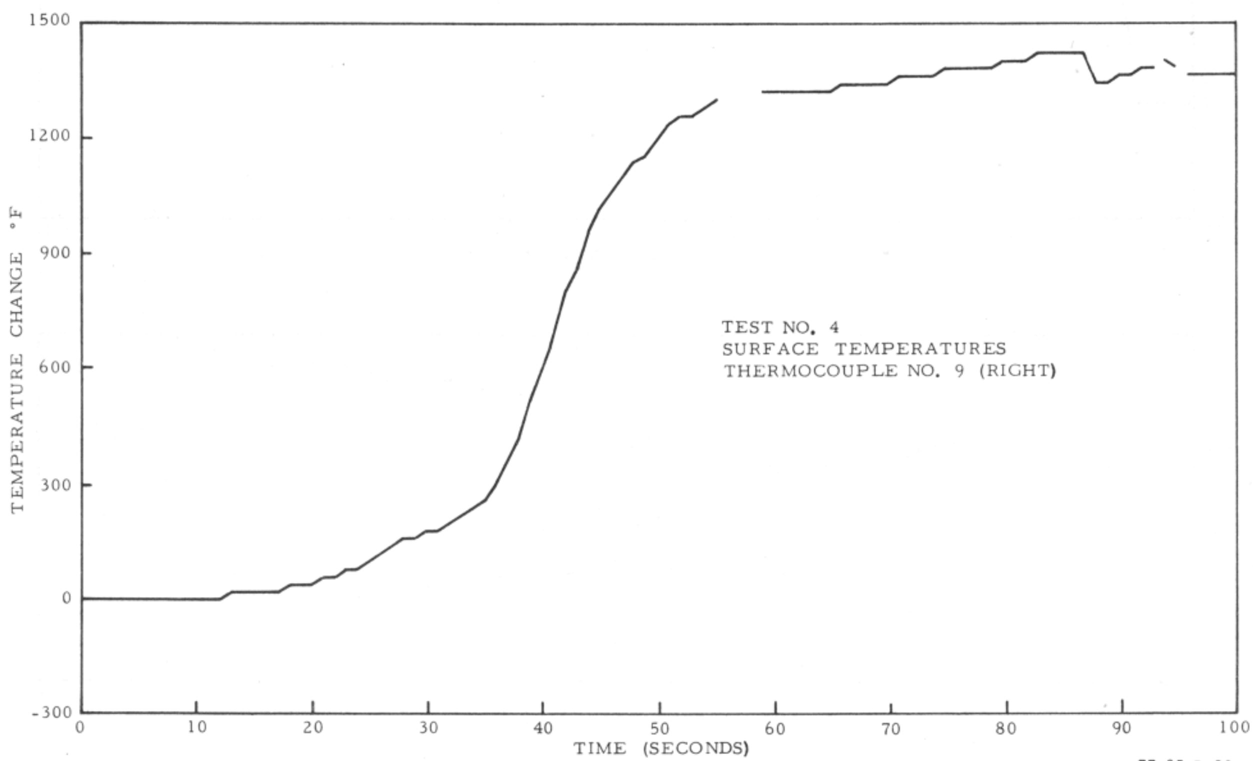

TEST NO. 4
AIR TEMPERATURES
PILOT'S COMPARTMENT STATION 123
TEMPERATURE CHANGE °F
300
0
-300
0
10
20
30
40
50
60
70
80
90
100
TIME (SECONDS)
77-35-B-18


1500
1200
900
600
300
0
-300
TEMPERATURE CHANGE °F
0
10
20
30
40
50
60
70
80
90
100
TIME (SECONDS)
TEST NO. 4
AIR TEMPERATURES
ELECTRONICS COMPARTMENT,
STATION 600


600
300
0
-300
TEMPERATURE CHANGE °F
0
10
20
30
40
50
60
70
80
90
100
TIME (SECONDS)
TEST NO. 4
AIR TEMPERATURES
AFT BOMB BAY
STATION 450

77-35-B-19

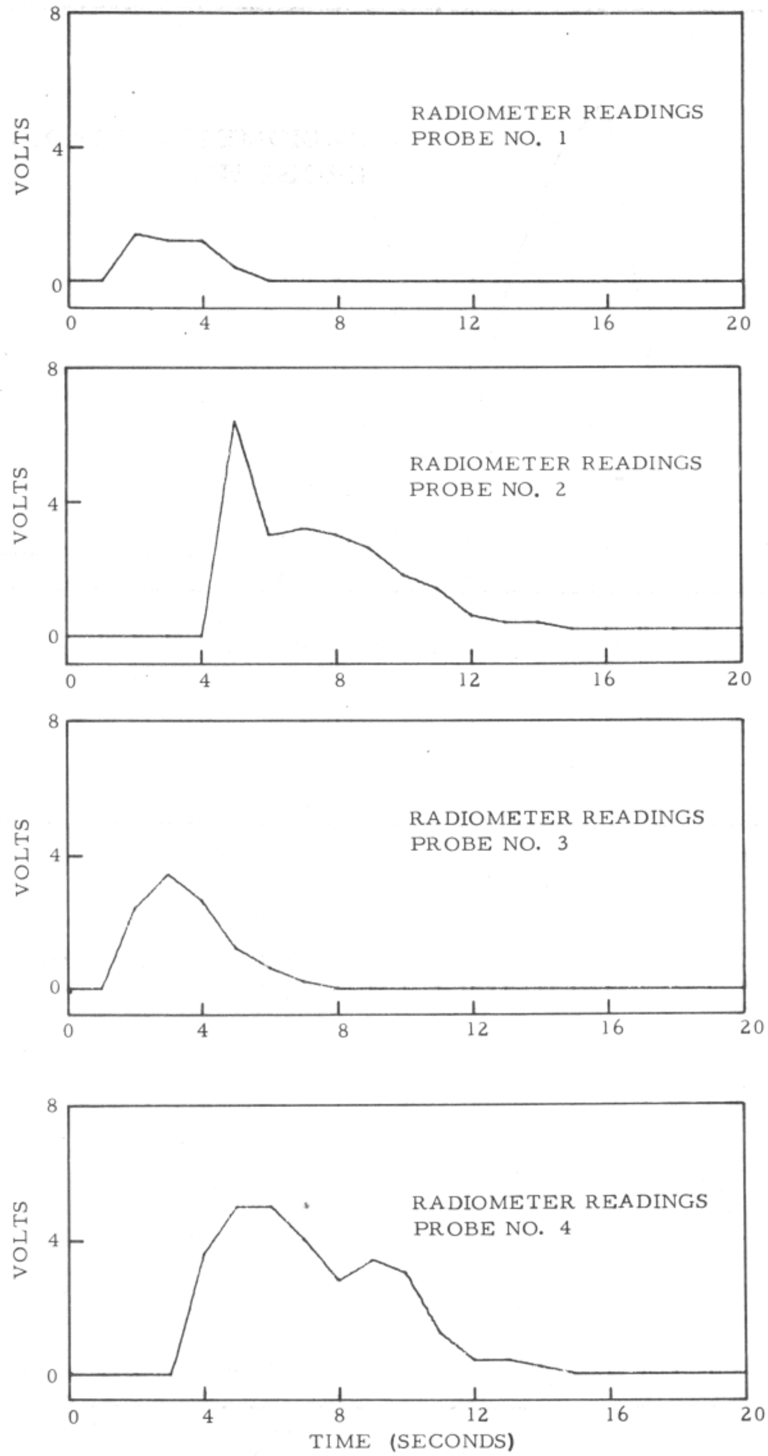

600
300
0
-300
TEMPERATURE CHANGE °F
0
10
20
30
40
50
60
70
80
90
100
TIME (SECONDS)
TEST NO. 4
SURFACE TEMPERATURES
THERMOCOUPLE NO. 8 (RIGHT)


1500
1200
900
600
300
0
-300
TEMPERATURE CHANGE °F
0
10
20
30
40
50
60
70
80
90
100
TIME (SECONDS)
TEST NO. 4
SURFACE TEMPERATURES
THERMOCOUPLE NO. 9 (RIGHT)

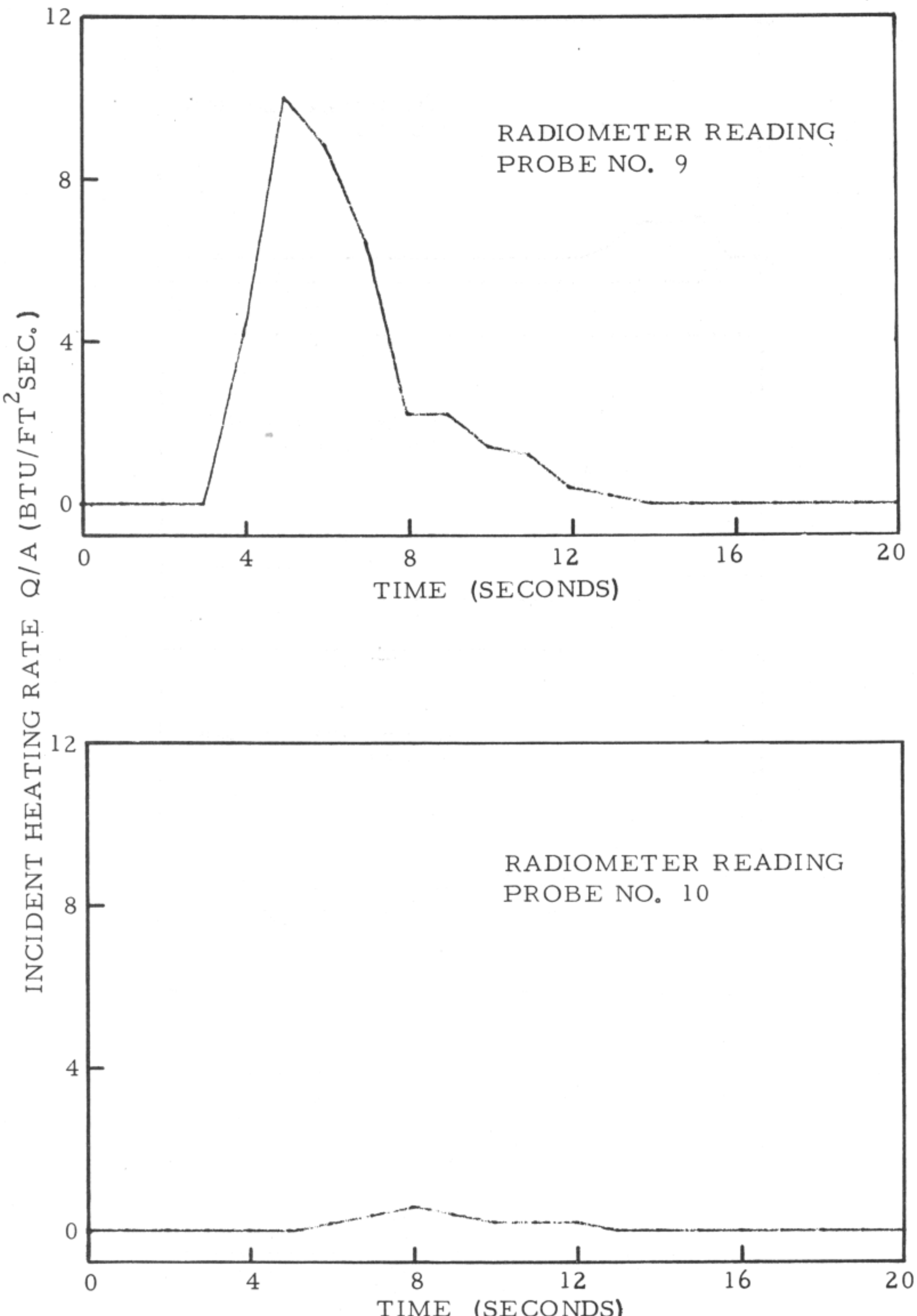
77-35-B-20

APPENDIX C

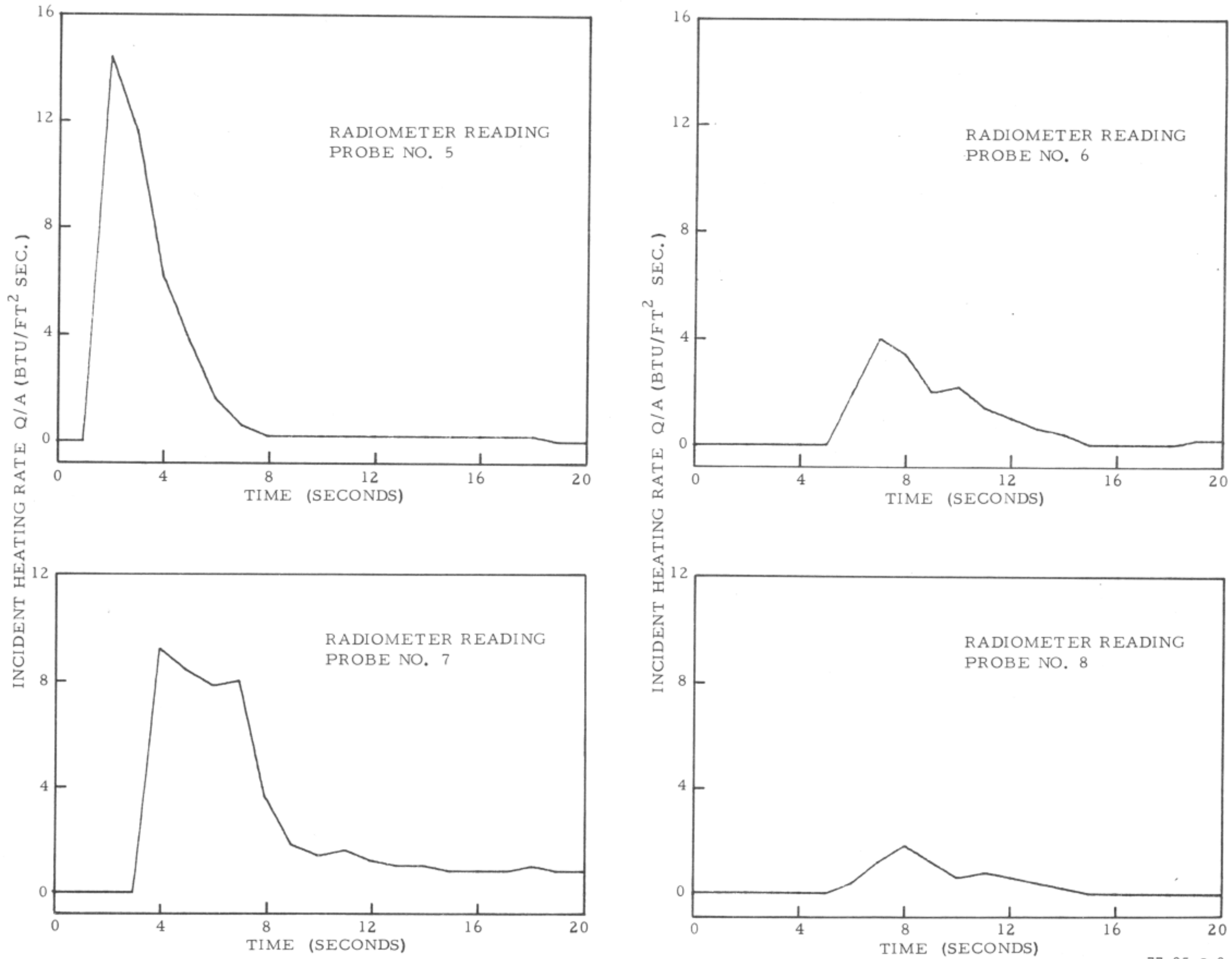
RADIOMETER PLOTS
SECOND RB66 TEST
(TEST NO. 4)


RADIOMETER READINGS
PROBE NO. 1
RADIOMETER READINGS
PROBE NO. 2
RADIOMETER READINGS
PROBE NO. 3
RADIOMETER READINGS
PROBE NO. 4
VOLTS
0
4
8
0
4
8
12
16
20
TIME (SECONDS)

77-35-C-1


INCIDENT HEATING RATE Q/A (BTU/FT²SEC.)
RADIOMETER READING
PROBE NO. 9
TIME (SECONDS)
RADIOMETER READING
PROBE NO. 10
TIME (SECONDS)
0
4
8
12
16
20

77-35-C-2



INCIDENT HEATING RATE Q/A (BTU/FT² SEC.)
TIME (SECONDS)
RADIOMETER READING
PROBE NO. 5
RADIOMETER READING
PROBE NO. 6
RADIOMETER READING
PROBE NO. 7
RADIOMETER READING
PROBE NO. 8
0
4
8
12
16
20

77-35-C-3