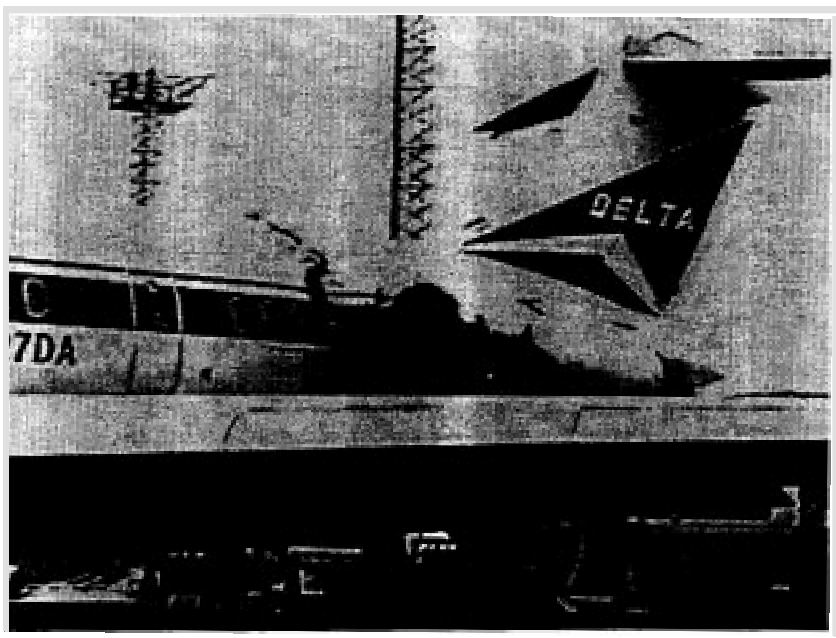
Douglas Aircraft Company Submission

to the

National Transportation Safety Board



Delta Air Lines, Inc. MD-88 Uncontained Engine Failure Accident

Pensacola, Florida, 6 July 1996

DCA96MA068

INTRODUCTION

Pursuant to Title 49 Code of Federal Regulations Chapter VIII, §845.27, the following is respectfully submitted for the National Transportation Safety Board's consideration regarding the Delta Air Lines, Inc. MD-88 uncontained engine failure accident, Pensacola, Florida, 6 July 1996 (DCA96MA068).

OVERVIEW

The purpose of this submission is to provide the National Transportation Safety Board with the Douglas Aircraft Company's (DAC's) analysis and conclusions concerning a hypothetical "worst case" scenario, e.g., this uncontained engine failure event occurring during pressurized flight. The actual damage to the aircraft's systems and structure was taken into consideration, as were the resulting additional demands on the systems and/or structure during flight. The results of this analysis showed that despite the extensive aircraft structural damage, the MD-88 was capable of continued controlled flight to a safe landing.

AIRCRAFT AND SYSTEM DAMAGE

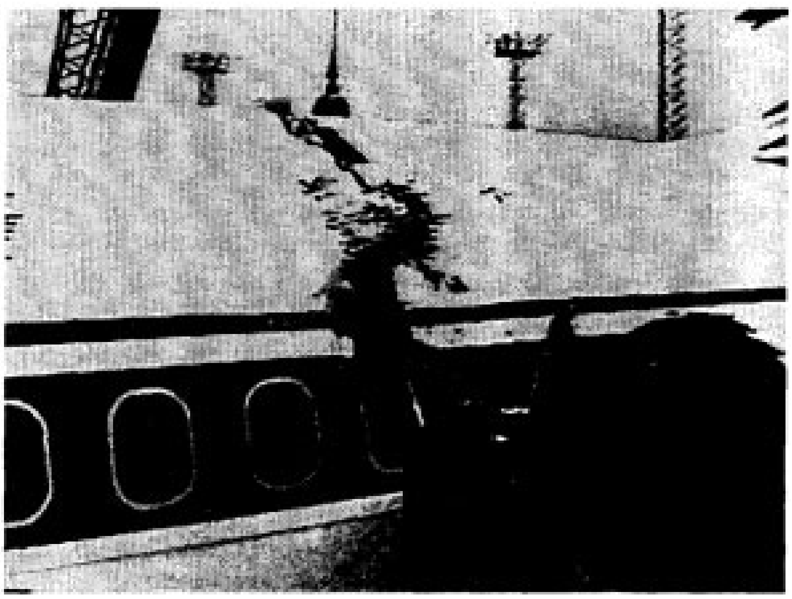


Figure 1. Fuselage Left Side Damage

As described in the *Structures/Systems Group Chairman's Factual Report,* the aft fuselage adjacent to the Number 1 (left) engine sustained puncture holes and tears from the uncontained first stage hub failure (Figures 1 and 2). This damage was confined to above the cabin floor, and generally from the window belt area on the left side over the top of the fuselage structure to the window belt area on the right side. Numerous frames, longerons, and intercostals were severed, buckled, and/or cracked, and a large "gash" was torn in the left fuselage from the window belt to approximately left longeron 2. Most of the wires in a bundle near right side longeron **4,** running fore and aft, were severed, including right generator differential protection loop wires. Damage to the differential circuitry caused the right generator to trip off-line, and the left engine failure brought the left engine generator off line, resulting in the loss of primary AC power (the APU generator was not on line because it was not operating at the time of the accident and would not be operating in flight) until the Emergency Power Switch was selected "ON."

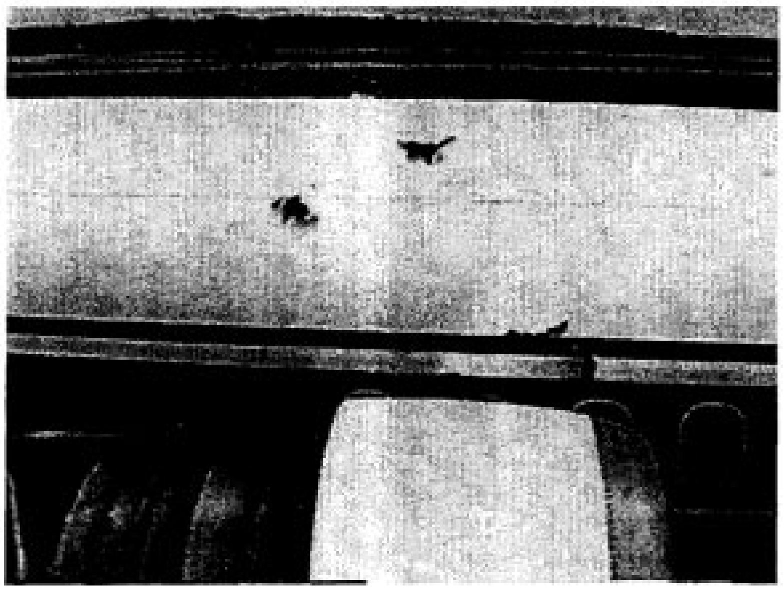


Figure 2. Right Side Fuselage Damage

All flight control cables and hydraulic lines below the floor remained undamaged during this accident.

AIRCRAFT SYSTEMS:

Discussion: The MD-80 series aircraft and the DAC Twinjet fleet as a whole were designed and certified to be safely flown and landed in the event of multiple failures such as the loss of all hydraulic system power and/or the loss of primary electrical power.

Even though the loss of primary electrical power occurred in this accident, the Emergency Power System restored electrical power from the battery to key aircraft systems and flight deck instrumentation per design. After the uncontained engine failure event, the flight crew turned the Emergency Power Switch to the "ON" position' and regained communications with the ATC local controller.[2] Further, post-accident testing of the Passenger Address and Interphone systems using the Emergency Power System indicated normal functioning.[3] Therefore, it was concluded that the Emergency Power System functioned normally during this event, and if the event had occurred in flight the Emergency Power System would have provided instrument guidance and communications with Air Traffic Control, such that a safe landing could have been made.

In regard to hydraulic power, the loss of left engine power would prevent that engine's pump from pressurizing the left hydraulic system. However, if the left system was intact and the hydraulic system switches were configured for takeoff, the left system would be pressurized by the right hydraulic system via a transfer pump, which would remain "ON" even with the loss of primary power. Even in a worst-case condition with no hydraulic power available, the MD-80 series aircraft have been designed for and have successfully demonstrated no-hydraulics approaches and landings during FAA certification flight testing, because the pilot has direct mechanical linkages to control surfaces on all three axis. It should be noted that the right hydraulic system, which was not damaged during this accident, provides power to the landing gear and the rudder, while either the left or the right systems can power the flaps, slats, ground steering, and brakes. Had the hydraulics system failed, the landing gear could have been extended via the emergency system (free fall) and accumulators would have retained pressure for thrust reversing and wheel brakes.

Conclusion: From a systems and flight control standpoint, if the event had occurred after takeoff or later in flight, the MD-80 aircraft, with the damages that were experienced by the uncontained engine failure event, would have been controllable to a safe landing.

AIRCRAFT STRUCTURE:

Discussion: An evaluation of the structural integrity of the accident aircraft was performed. The evaluation addressed two basic issues, (1) the ability of the floor structure to withstand the effects of a significant pressure differential between the cabin and the cargo hold as the result of rapid decompression, and (2) the ability of the damaged fuselage shell to carry required cabin pressure, bending,

[1] EMER PWR Switch - ON is the first item in the "Complete Loss of AC Power" Emergency Checklist.
[2] Operational Factors/Human Performance Group Chairman's Factual Report, Page 6.
[3] Structures/Systems Group Chairman's Factual Report, Page 5.

shear and torsion loads. Flutter characteristics of the damaged aircraft were also investigated.

The initial evaluation was made by comparing the damages documented on the accident aircraft with previous rotorburst finite element analysis provided to the FAA for certification of the MD-87 and MD-90. For the purposes of this submission, it should be noted that the MD-88 was substantiated from the MD-87 analysis because the MD-87 and MD-88 have the same JT8D engine and identical structure in the affected fuselage section, and the MD-87 has higher flight loads. The MD-90 structure is identical to the MD-88 in the damaged area and the assumed damage from the V2500 engine is greater. In addition, the MD-90 analysis uses more recently developed FAA criteria, plus analysis refinements. Cabin pressure was identical for all models.

The actual damages used for comparison with the pre-existing certification analyses were described in the *Structure/Systems Group Chairman's Factual Report,* pages 3 and 4. Based on these damages, the total decompression area is estimated to be approximately 1400 square inches or about 9.72 square feet, entirely above the floor structure. The floor beams are capable of withstanding the effects of rapid decompression from a maximum opening above the floor of 32.5 square feet (and a maximum opening below the floor of 4 square feet),

The integrity of the damaged fuselage shell was evaluated using a finite element approach. The pre-existing rotorburst finite element analyses for the MD-87 and the MD-90 included extensive damage, and a cursory examination indicated that the overall shell stiffness and strength reduction suffered by the accident aircraft was not unlike that already addressed in our certification analysis. The similarities were, in fact, sufficient to judge that adequate flutter margins would have been maintained. For the residual strength assessment, however, the actual damages experienced by Fuselage 1524 were sufficiently different from those analyses that a special model was created. The supplementary model was created from the pre-existing MD-90-30 aft fuselage model. The MD-90 model was used for convenience and because (in this area) the structure is identical to that of the MD-88, and the loads are comparable.

Attachment 1[4] illustrates the idealized damage as it was represented in a finite element model of the aft fuselage shell. This idealization is a conservative[5] simplification of the significant structural damage to the accident MD-88 skin/longeron panels and frames due to puncture and/or severing of the structural members by engine rotor debris. Secondary damage such as dents,

[4] Attachment 1 is a plate diagram that illustrates the fuselage skin, frame, longeron, etc. layout as if the fuselage had been severed along the length of its underside ("belly"), then laid out flat.
[5] Longer arc length, to Longeron 19L (corresponds to the bottom of the window belt panel), and additional damage area from Longeron 7L-9L, stations 1271-1287.

gouges, skin wrinkling, and fastener “pull-through” is not included in this idealization unless it is believed to affect the load carrying capability of the residual primary structure. The damage to the right side of Fuselage 1524 (see Figure 2 above) consisted of two cut frames plus several widely scattered cuts. None of these cuts were considered individually significant (this assumption is supported by pre-existing MD-90 studies which showed that small cuts opposite a large cut have an insignificant effect on the overall fuselage stiffness and strength), and the cuts were sufficiently scattered such that it is believed that they would not have coalesced into a single (critical) crack.

The new MD-90 idealization includes a 96 inch cut to represent the MD-88 damage illustrated in Attachment 1. The results of this analysis are shown in Figure 3 below. It should be noted that the conditions generating the lowest Margins of Safety are illustrated in Figure 3. Single-engine continued takeoff, single engine landing, and other structural loading conditions were considered in the analysis but generated larger Margins of Safety.

	ANALYSIS DAMAGE ZONE			
	Arc Length	**Margin of Safety**	**Failure Mode**	**Design Cond.** (Note **1**)
Finite Element Analysis Cond. 1	**96** in. plus frame **cut**	+.07	(panel shear + tension)	**1.**0 g level flight + 1**10%** cabin pressure differential
Finite Element Analysis Cond. 2	96 in. plus frame cut	+.09	(panel shear + tension)	70% (2.5 **g** balanced maneuver, engine out), no pressure
MD-88 Fus 1524 Actual	75 in., plus 14 in. window and cut frame	greater than or equal to **+.07 by** comparison with Conds. 1 and **2** above		

Note 1: Ultimate loading conditi

Figure 3: Comparison Summary Chart

Results: As the Figure 3 chart illustrates, the Margin of Safety (M.S.) equals at least +.07 for the conservative MD-90 finite element analysis[6]. This, in combination with comparisons against the pre-existing certification analyses, indicates that Fuselage **1524** would have been structurally capable of continued safe flight to landing with the damages that were experienced by the uncontained engine failure event.

SUMMARY CONCLUSIONS

The foregoing analyses indicate that the aircraft would have been capable of continued controlled flight to a safe landing had this uncontained engine failure event occurred after takeoff and/or during pressurized flight.

SAFETY RECOMMENDATIONS

The Douglas Aircraft Company recommends that current and future industry rotorburst event studies and/or test activities, together with regulatory review processes, continue to take the data from this accident and other rotorburst events into consideration in efforts to meet the dual goals of preventing uncontained engine failures and, should they occur, of minimizing their effect on airframe integrity and airworthiness.

[6] Because the Margin of Safety for Design Condition **1**, Figure 3, is less than the Margin of Safety for Design Condition 2, Condition **1** is the more critical.

ATTACHMENT I

;EMBLY

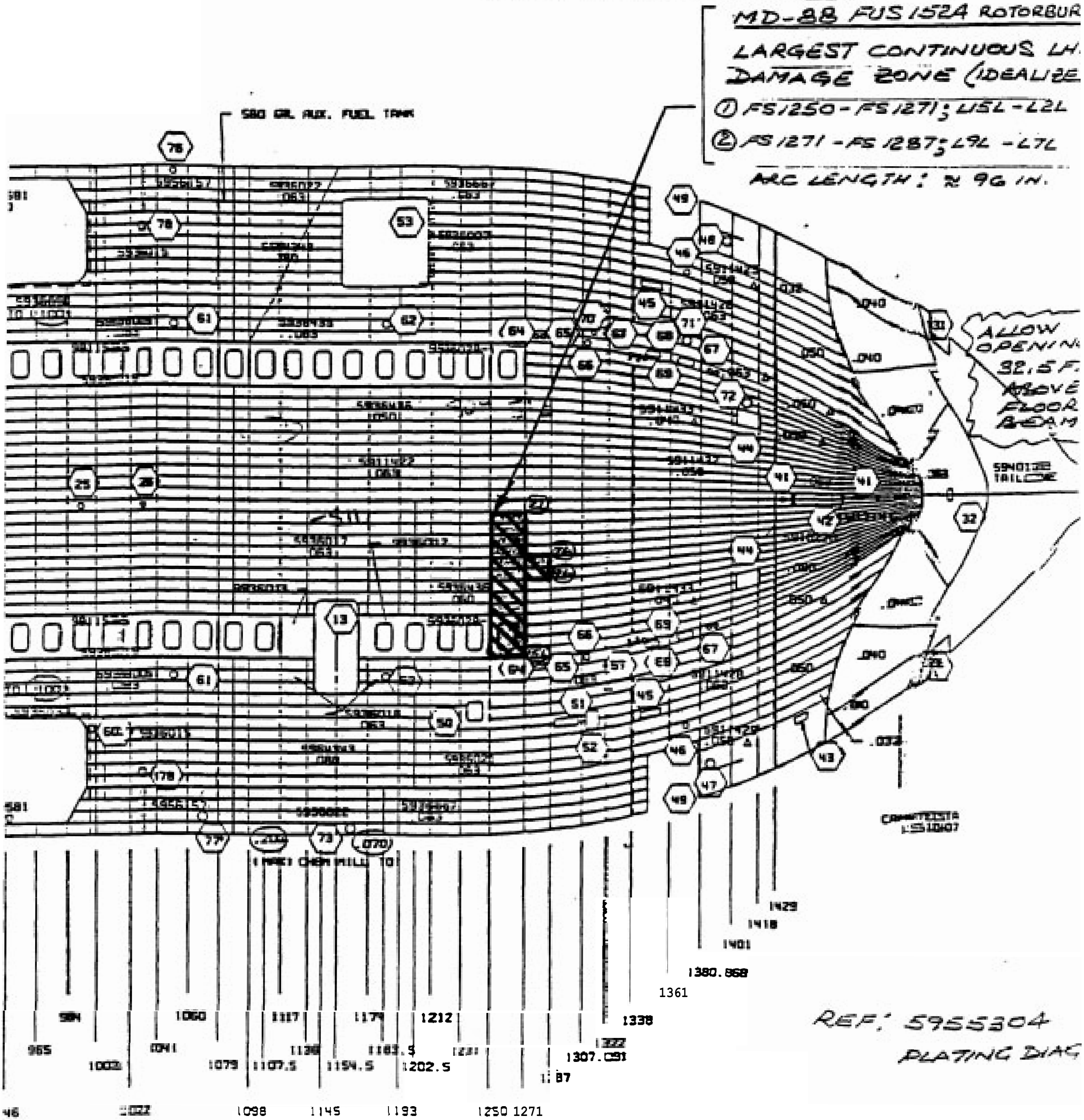
G DIAGRAM

PREPARED BY M. Gadeski
CHECKED BY
DATE
TITLE FUSELAGE PRIMARY STRUCTURAL STRENGTH

MCDONNELL DOUGLAS

PAGE
MODEL
REPORT NO

McDonnell Douglas Corporation—This information is subject to data rights legends on title page or first page



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PLATING DIAG

ATTACHMENT II

in showing compliance with § 25.571(e), Damage-tolerance (discrete source) evaluation. The intent of these guidelines is to define load conditions that will not be exceeded with a satisfactory level of confidence on the flight during which the specified incident of § 25.571(e) occurs. In defining these load conditions, consideration has been given to the expected damage to the airplane, the anticipated response of the pilot at the time of the incident, and the actions of the pilot to avoid severe load environments for the remainder of the flight consistent with his knowledge that the airplane may be in a damaged state. With these considerations in mind, the following ultimate loading conditions should be used to establish residual strength of the damaged structure.

b. The maximum extent of immediately obvious damage from discrete sources (§ 25.571(e)) should be determined and the remaining structure shown, with an acceptable level of confidence, to have static strength for the maximum load (considered as ultimate load) expected during completion of the flight.

c. The ultimate loading conditions should not be less than those developed from the following conditions:

(1) At the *time* of the incident:

(i) The maximum normal operating differential pressure, multiplied by a 1.1 factor, plus the expected external aerodynamic pressures during 1 g level flight, combined with 1 g flight loads.

(ii) The airplane, assumed to be in 1 g level flight, should be shown to be able to survive any maneuver or any other flight path deviation caused by the specified incident of § 25.571(e), taking into account any likely damage to the flight controls and pilot normal corrective action.

(2) Following the incident:

(i) Seventy percent (70%) limit flight maneuver loads and, separately, 40 percent of the limit gust velocity (vertical or lateral) at the specified speeds, each combined with the maximum appropriate cabin differential pressure (including the expected external aerodynamic pressure).

(ii) The airplane must be shown by analysis to be free from flutter up to V_D/M_D with any change in structural stiffness resulting from the incident.

[redacted]

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